AW139 Helicopter



AVIONIC SYSTEMS MAINTENANCE TYPE TRAINING COURSE TRAINING MANUAL (ASMTTC)

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LIST OF EFFECTIVE DATA MODULES AND RELEVANT PAGES

The listed Data Modules are included in issue 001, dated 2010-01-01, of this Training Information Set. C = Changed DM; N = New DM.
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HIGHLIGHTS

Record of Issues:
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DATA MODULE CODE	DATA MODULE

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LIST OF ABBREVIATIONS

AB	Agusta Bell
AC	Alternating Current; Advisory Circular; Autopilot Controller
A/C	Aircraft
ACCB	Air Conditioning Control Box
ACP	Audio Control Panel
ACT	Actuator
ADA	Air Data Application
ADC	Ampére Direct Current
ADI	Attitude Director Indicator
ADF	Automatic Direction Finder
ADM	Air Data Module
ADS	Air Data System
ADV	Advance
AEO	All Engine Operative
AFCS	Automatic Flight Control System
AGB	Accessory Gear Box
AGL	Above Ground Level
Ah	Ampere hour
AHRS	Attitude And Heading Reference System
AHRU	Attitude Heading Reference Unit
AIOP	Actuator Input/Output Processor (Module)
ALT	Altitude; Barometric Altitude
ALTA	Altitude Acquire
ALTN	Alternate
ALT SEL	Altitude Select
ALVL	Autolevel
AM	Amplitude Modulation
AMLCD	Active Matrix Liquid Crystal Display
AMM	Air Management Module
AMP	Ampére
ANT	Antenna
AOA	Angle of Attack
AP	Autopilot
APM	Aircraft Personality Module
APP	Approach
APPR	Approach
ARINC	Aeronautical Radio Inc.
ASCB	Avionic Standard Communication Bus
ASCB-D	Avionics Standard Communication Bus ver.D
ASEL	Altitude Preselect
ASSUR	Assurance
ATC	Air Traffic Control

ATT	Attitude; Attitude Retention Module
AUD	Audio
AUTO	Automatic
AUX	Auxiliary
AW	AgustaWestland
AWG	Aural Warning Generator
BAG	Baggage
BATT	Battery
BC	Back Course
BDGW	Basic Design Gross Weight
BFO	Beat Frequency Oscillator
BIT	Built In Test
BKUP	Backup
BL	Buttock Line
BLT	Belt
BOD	Bottom Of Descent
BOV	Bleed Valve
BOW	Basic Operating Weight
BRG	Bearing
BRK	Breaker
BRT	Bright ; Brightness
BTL	Bottle
C	Collective
°C	Celsius / Centigrade(degree)
CAA	Civil Aviation Authority
CAB	Cabin
CAN	Control Area Network (bus)
CAPTS	Captions
CAS	Crew Alerting System; Calibrated Air Speed
CAT	Category
CB	Circuit Breaker
C/B	Circuit Breaker
CCD	Cursor Control Device
CCP	Cockpit Control Panel
CCU	Cockpit Control Unit
CCW	Counter Clock-Wise
CFUR	Critical Failure (ECC)
CG	Center of Gravity
CHM	Chime
CHRG	Charge
CIO	Control Input/Output (MAU Module)
CKPT	Cockpit
CLP	Collective Lever Pitch

CLPR	Caliper
CLR	Clear
CLTV	Collective
CMC	Central Maintenance Computer
CMC-RT	Central Maintenance Computer Remote Terminal
CMS	Central Maintenance System
COLL	Collective; Collision
COM	Communication
COMM	Communication
COND	Conditioning
CONTR	Control
COV	Crossover Valve
CP	Control Panel
CPLT	Copilot
CPL	Coupled
CRS	Course
CRZ	Cruise
CSIO	Custom Input/Output (MAU Module)
CSL	Console
CT	Current Transformer; Compressor Turbine
CVR	Cockpit Voice Recorder
CW	Clock-Wise
CWS	Central Warning System
DAU	Digital Acquisition Unit
DBM	Data Base Module
DC	Display Controller; Direct Current; Down Counter
DCL	Deceleration
DCM	Detachable Configuration Module
DCU	Data Collection Unit
DEL	Delete
DEOS	Digital Engine Operating System (MAU)
DET	Detector
DETR	Detector
DF	Directional Finder
DFDAU	Digital Flight Data Acquisition Unit
DG	Directional Gyro
DGPS	Differential Global Positioning System
DGR	Degraded
DH	Decision Height
DIM	Dimmer / Dimmed / Dimming
DIR	Direct / Direct to
DITS	Digital Information Transmission System
DME	Distance Measuring Equipment
DMG	Digital Map Generator

DN	Down
DOP	Dilution of Precision
DOTS	Duct Overheat Temperature Switch
DPST	Double Pole Single Throw
DR	Dead Reckoning
DSCWD	Discrete Worth (EEC)
DTD	Data Transfer Device
DTK	Desired Trak
DU	Display Unit
DVS	Doppler Velocity System
DWS	Debris Warning System
EAPS	Engine Air Particle Separator
EASA	European Aviation Safety Agency
ECL	Engine Control Lever
ECP	Engine Control Panel
ECS	Environmental Control System
ECU	Engine Control Unit
EDS	Electronic Display System
EEC	Electronic Engine Control
EGPWS	Enhanced Ground Proximity Warning System
EICAS	Engine Instruments and Crew Alerting System
ELEC	Electric / Electrical
ELT	Emergency Locator System
EMC	Electromagnetic Compatibility
EMERG	Emergency
EMF	Electro-Motive Force
EMI	Electromagnetic interference
EMS	Emergency Medical Service
ENAC	Ente Nazionale Aviazione Civile
ENBL	Enable / Enabled
ENG	Engine
EP	Estimated Position; Electric Pump
EPGDS	Electrical Power Generation And Distribution System
EPU	Estimated Position Uncertainty
ESIS	Electronic Stand-by Instrument System
ESS	Essential
ET	Elapsed Time
ETA	Estimated Time of Arrival
ETE	Estimated Time En-route
EXT	External
EXTG	Extinguish / Extinguisher
EXTN	External
F/C	Flight Control
F/C SOV	Flight Control Shut-off Valve

FAA	Federal Aviation Administration
FAF	Final Approach Fix
FCOC	Fuel Cooled Oil Cooler
FCP	Fuel Control Panel
FCU	Fuel Computer Unit
FD	Flight Director
FDAU	Flight Data Acquisition Unit
FDR	Flight Data Recorder
FF	Fuel Flow
FH	Flying Hours; Flight Hours
FHDB	Fault History Data Base
FL	Flight Level; Fuel
FLIR	Forward Looking Infra Red
FLT	Flight
FLT DIR	Flight Director
FMCW	Frequency Modulated Continuous Wave
FMM	Fuel Management Module
FMS	Flight Management System
FMV	Fuel Management Valve
FOD	Foreign Object Damage
FOG	Fiber Optic Gyro
FOM	Figure of Merit
FPL	
FTR	Force Trim Release
FV	Flapper Valve
FWD	Forward
GA	Go-Around
GBS	Groun Base Software
GC	Guidance Controller
GCU	Generation Control Unit
GEN	General; Generator
GFE	Government Furnished Equipment
GLS	GPS Landing System
GOV	Governor (Engine)
GND	Ground
GPS	Global Positioning System
GR	Gear
GS	Glide Slope; Ground Speed
GSE	Ground Support Equipment
GW	Gross Weight
HCB	Heating Control Box
Hd	Density Altitude
HDG	Heading
HDPH	Head Phone

HDST	Headset
HECTM	Helicopter Engine Condition Trend Monitoring
HF	High Frequency
HIL	Horizontal Integrity Limit (GPS)
HOV	Hover
Hp	Pressure Altitude
HP	High Pressure; Horse Power
HPA	hecto Pascal (hPa)
HPS	Hydraulic Power Supply
HPU	Hydraulic Power Unit
HSI	Horizontal Situation Indicator
HT	Height
HTL	Heater
HUMS	Health and Usage Monitoring System
HVR	Hover
H-V	Height - Velocity
HYD	Hydraulic
IAF	Initial Approach Fix
IAS	Indicated Air Speed
ICS	Intercommunication System
ID	Ident / Identifier
IETP	Interactive Electronic Technical Publication
IFR	Instrument Flight Rules
IGB	Intermediate Gearbox
IGE	In Ground Effect
IGN	Ignition
IHBT	Inhibit
ILS	Instrument Landing System
INPH	Interphone
IR	Infra Red
ISA	International Standard Atmosphere
ITT	Inter-Turbine Temperature
JAA	Joint Aviation Authority
JAR	Joint Airworthiness Regulations
KIAS	Knots Indicated Air Speed
KPH	Kilograms per Hour
KTS	Knots
L	Left
LAN	Local Area Network
LAT	Lateral
LCD	Liquid Crystal Display
LCF	Low Cycle Fatigue

LD	Load
LDG	Landing
LDG GR	Landing Gear
LDI	Load Diagnostic Information
LED	Light Emitting Diode
LG	Landing Gear
LGCL	Landing Gear Control Lever
LGCP	Landing Gear Control Panel
LGCV	Landing Gear Control Valve
LH	Left Hand
LIM	Limiter / Limiting
LLD	Low Level Detection
LLDM	Low Level Detection Mode
LLLTV	Low Light Level Television
LLS	Lightning Sensor System
LOC	Localizer
LOS	Line Of Sight
LONG	Longitude; Longitudinal
LNAV	Lateral Navigation
LP	Low Pressure
LRM	Line Replaceable Module
LRU	Line Replaceable Unit
LSK	Line Select Key
LSS	Lightning Sensor System
LT	Left; Light; Local Time
LVDT	Linear Variable Differential Transducer
MAG	Magnetic
MAN	Manual
MAP	Missed Approach Point
MAU	Modular Avionics Unit
MAX	Maximum
MB	Marker Beacon
MCDU	Multifunction Control Display Unit
MCL	Master Caution Light
MCP	Maximum Continuous Power; Miscellaneous Control Panel
MDA	Minimum Descent Altitude
MEM	Memory
MFD	Multi-Function Display
MGB	Main Gear Box
MIC	Microphone
MIL-SPEC	Military Specification
MIL-STD	Military Standard
MIN	Minimum
MISC	Miscellaneous
MKR BCN	Marker Beacon

MLG	Main Landing Gear
MM	Maintenance Manual
MMEL	Master Minimum Equipment List
MMH	Maintenance Man Hour
MMI	Man-Machine Interface
MOT	Mark On Target
MPFDR	Multipurpose Flight Data Recorder
MPOG	Minimum Pitch On Ground
MR	Main Rotor
MRA	Main Rotor Actuator
MRC	Modular Radio Cabinet
MRDS	Main Rotor Drive System
MSG	Message
MSL	Mean Sea Level
MSTR	Master
MSU	Magnetic Sensor Unit
MTBAF	Mean Time Between Attributable Fault
MTBF	Mean Time Between Failures
MTBO	Mean Time Between Overhaul
MTBR	Mean Time Between Removal
MTTR	Maintenance Time to Replace
MWF	Monitor Warning Function
MWL	Master Warning Light
MWS	Monitor Warning System
N/A	Not Applicable, Not Available
NATO	North Atlantic Treaty Organization
NAV	Navigation
NC	Normally Closed
NCFUR	Non-Critical Failure (EEC)
N-ESS	Non Essential
Nf or NF	Engine free turbine speed
Nf/Npt	Engine free turbine speed
Ng	Engine gas generator Turbine/Compressor Speed
Ng or NG	Engine gas generator speed
NIC	Network Interface Controller
NIM	Network Interface Module
NLG	Nose Landing Gear
NM	Nautical Mile
NO	Normally Open
NORM	Normal
NO SMK	No Smoking
Nr	Rotor rpm
Nr or NR	Rotor speed
N-S	Non Essential
NVG	Night Vision Goggle

NVM	Non Volatile Memory
OAT	Outer Air Temperature
OC	Over Current
OEI	One Engine Inoperative
OGE	Out of Ground Effect
OS	Over Station
OVHT	Overheat
OVTQ	Overtorque
P	Pitch
PA	Passenger Address; Pubblic Address
PAX	Passengers
PCI	Peripheral Component Interconnect
PCM	Power Control Module
PDP	Power Distribution Panel
PERF	Performance
PFD	Primary Flight Display
PI	Power Index
PLA	Power Lever Angle
PLT	Pilot
PMA	Permanent Magnent Alternator
PMS	Power Management Switch
POSN	Position
P/N	Part Number
PPH	Pounds per Hour
PRAIM	Predictive RAIM
PREV	Previous
PRI	Primary
PROG	Progress
PROM	Programmable Read Only Memory
PRV	Preview; Pressure Relief Valve
P/R	Partial Retraction
PSM	Power Supply Module
PSU	Passenger Service Unit
PT	Power Turbine
PTT	Press to Talk; Press to Transmit
PWR	Power
QRH	Quick Reference Handbook
R	Right; Roll
RA	Rectract Actuator
R/A	Rectract Actuator
RAD	Radio
RAD ALT	Radio Altitude / Radio Altimeter

RAIM	Receiver Autonomous Integrity Monitor
RAM	Random Access Memory; Reliability; Availablility and Maintainability
RB	Rotor Brake
RBA	Rotor Brake Assembly
RBAA	Rotor Brake Actuation Assembly
RBCL	Rotor Brake Control Lever
RBCM	Rotor Brake Control Module
RBPI	Rotor Brake Pressure Indicator
RBRB	Rotor Brake Relays Box
RCB	Radio Control Bus
RCP	Reversion Control Panel
RCRD	Record
RDR	Radar
REC	Received; Record
REL	Release
RETR	Retract / Retraction
RFM	Rotorcraft Flight Manual
RH	Right Hand
RHT	Radar Altitude Hold
RIB	Remote Images Bus
RIC	Remote Instrument Controller
RNAV	Area Navigation
RNP	Required Navigatio Performance
ROC	Rate of Climb
RPM	Revolution Per Minute
RSB	Radio System Bus
RST	Reset
RT	Right
RTD	Resistance Temperature Device
RTN	Return
RTR	Rotor
SA	Shortening Actuator
SAR	Search and Rescue
SAS	Stability Augmentation System
SATCOM	Satellite Communication
SCMS	Software Configuration Management Module
SEAT BLT	Seat Belt
SEL	Select
SELCAL	Selective Call
SHP	Shaft Horse Power
SHT DN	Shut Down
SID	Standard Instrument Departure
SL	Sea Level
S/N	Serial Number

SOV	Shut Off Valve
SP	Space
SPD	Speed
SPKR	Speaker
SQ	Squech
STA	Station (line)
STAR	Standard Terminal Arrival Route
STBY	Stand-By
ST/SP	Start / Stop
SSM	Sign Status Matrix
SW	Stopwatch
S/W	Software
SYS	System
TA	Traffic Advisory
TACAN	Tactical Air Navigation
TAS	True Air Speed
TAWS	Terrain Awareness and Warning System
TBD	To Be Defined
TBO	Time Between Overhaul
TCAS	Traffic Alert and Collision Avoidance System
TCPS	Temperature Compensated Pressure Switch
TCV	Temperature Control Valve
TD	Transition Down
TEMP	Temporary
TERR	Terrain
TEV	Thermostatic Espansion Valve
TGB	Tail Gear Box
TNG	Training
TO	Take Off
T/O	Take Off
TOC	Table of Contents; Top Of Climb
TOD	Top Of Descent
TOP	Take Off Power
TQ	Engine Torque
TR	Tail Rotor
TRA	Tail Rotor Actuator
TRANS	Transmission
TRDS	Tail Rotor Drive System
TRSOV	Tail Rotor Shut-Off Valve
TSO	Time Since Overhaul
TU	Transition Up
TWD	Threat Warning Display
UCPL	Uncoupled
UHF	Ultra High Frequency

ULB	Underwater Locator Beacon
UNLK	Unlock
UTC	Universal Time Coordinates
UTIL	Utility
UTIL SOV	Utility Shut-Off Valve
UV	Ultra-Violet
VAPP	VOR Approach
VCE	Voice
VDR	VHF Data Radio
VEL	Velocity
VENT	Ventilation
VFR	Visual Flight Rules
VGP	Vertical Glide Path
VHF	Very High Frequency
VIDL	VOR/ILS Data Link
VIL	Vertical Integrity Limit (GPS)
VIP	Very Important Person
VLO	Maximum landing gear operating speed
VLE	Maximum landing gear extended speed
VNE	Never Exceed speed
V/L	VOR/LOC
VNAV	Vertical Navigation
VNE	Velocity Never Exceed
VOL	Volume
VOR	VHF Omnidirectional Range
VOX	Voice Operated Switch / Voice Operated Circuit
VPATH	Vertical Flight Path
VR	Take Off Rotation Speed
V REF	Reference Speed
VS	versus
V S	Vertical Speed
VTA	Vertical Track Alert
V/UHF	VHF and UHF
WHL	Wheel
WL	Water Line
WOW	Weight-On-Wheels
WT	Weight
VTR	Winch-man Trim
WX	Weather
WXR	Weather Radar
XFD	Crossfeed
XFEED	Crossfeed
XMIT	Transmit

XMSN	Transmission
XPDR	Transponder
Y	Yaw
Z	Zulu (time)

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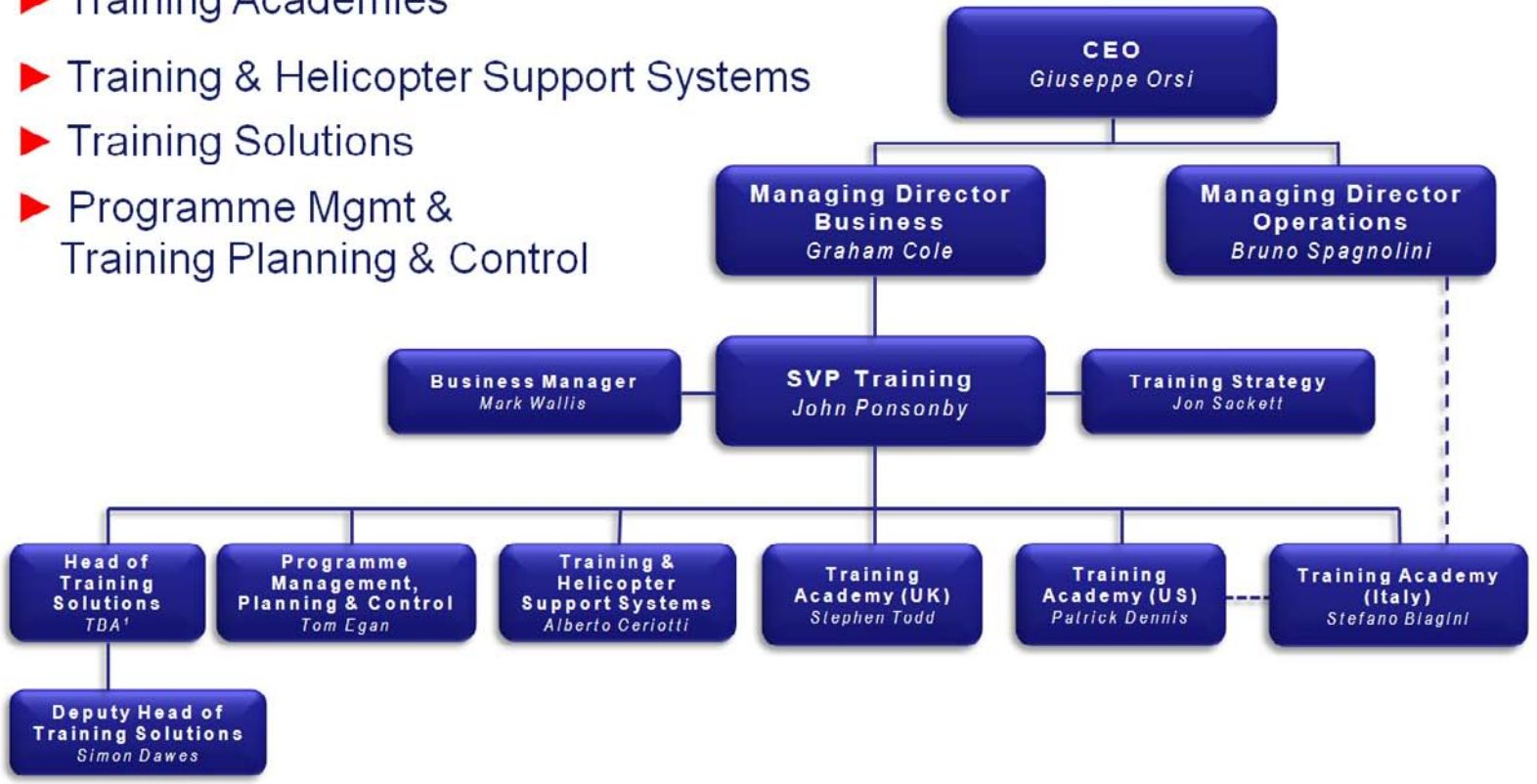
CHAPTER
00
HELICOPTER - GENERAL

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SECTION 00 - AW TRAINING ACADEMY ITALY “A. MARCHETTI”

AW Training is organised against four primary functional areas:

- Training Academies
- Training & Helicopter Support Systems
- Training Solutions
- Programme Mgmt & Training Planning & Control



ICN-3E-A-000000-G-A0126-00018-A-01-1

AW TRAINING ACADEMIES

Value to the Customer; Value to the Company

Priorities

- Ensure all customers have access to world class, value for money OEM training
- Improve the breadth of training solutions available to military and civil customers
- Improve the market penetration of these capabilities through consideration of a regional training footprint

Training Academies delivers training in Italy, UK, US and worldwide. Each Academy is responsible for assuring the delivery of the assigned training contracts.

The AW Training Academy Italy and USA are an international approved Organization which deliver Basic and Type Rating courses to Pilots and Technicians for Agusta design Helicopters.

AWTA UK is the Training delivery center for Westland design Helicopters, including EH101.



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AW TRAINING ACADEMIES

The AWTA Italy "Alessandro Marchetti" is located in Sesto Calende, next to Lake Maggiore in northern Italy some 60 kilometres from Milan and just south of the Italian Alps that tower over the region.



The Centre was officially opened in July 2006, relocated to the old historical "SIAI Marchetti" plant in Sesto Calende.

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AWTA ITALY, SESTO CALENDE

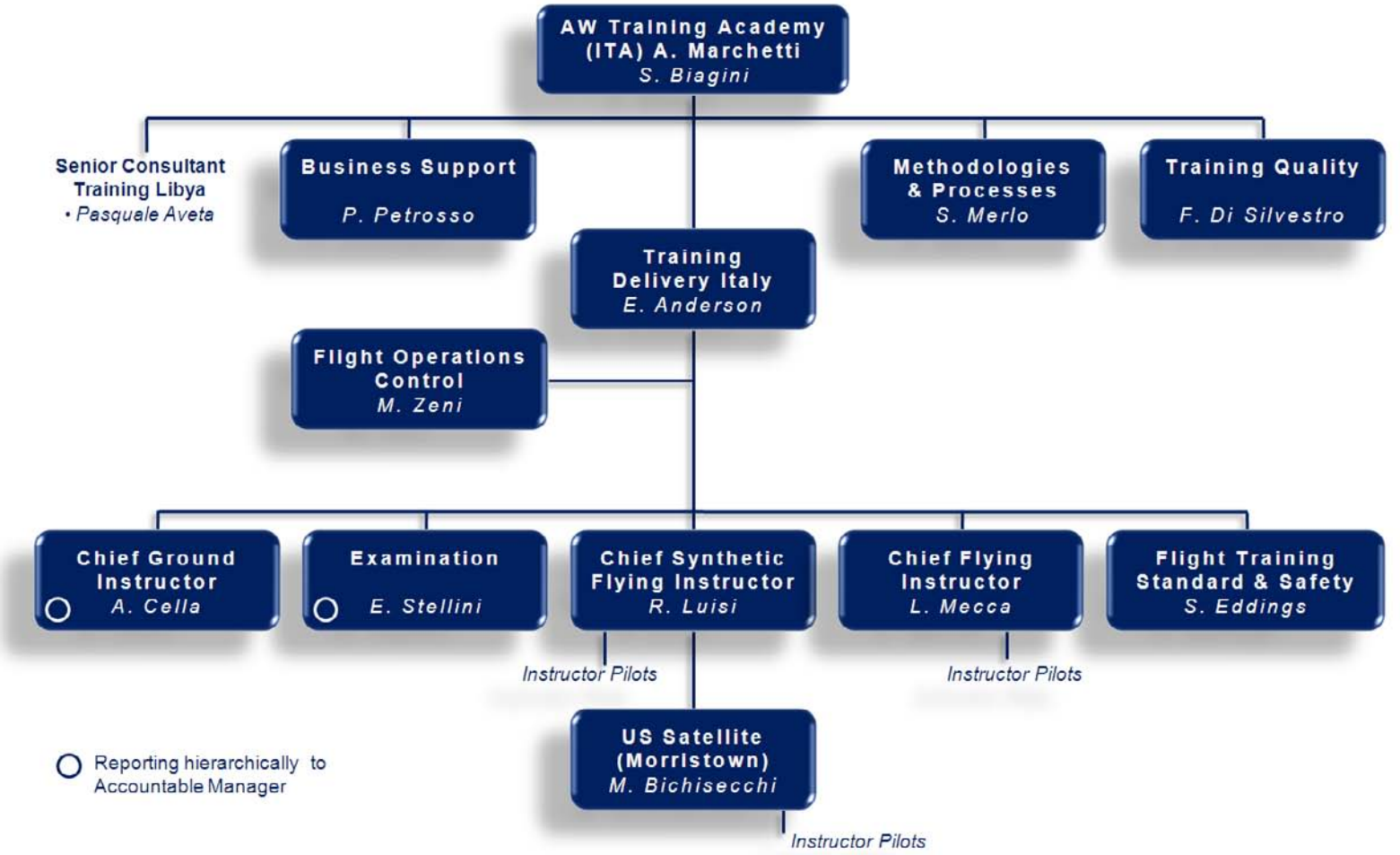
AW Training Academy Sesto Calende plant, the historical SIAI Marchetti facility.

The Società Idrovolanti Alta Italia (SIAI) was founded in 1915 in Sesto Calende. In 1922, Alessandro Marchetti took responsibility as Technical Director and co-owner, and under his guide SIAI (in the meantime named SIAI Marchetti) launched first-class aircraft production, such as S51 (Schneider trophy), S55 (1933: a fleet of 24 aircraft crossing Atlantic), S.79 (Damned Hunchback), etc. which made aviation history.



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AWTA ITALY, SESTO CALENDE



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AWTA ITALY ORGANISATION



General:

- EN 9100:2003 (UNI EN 9100:2005)
- UNI EN ISO 9001:2000
- AER-Q-2110 ; AQAP 2110; AQAP 2210

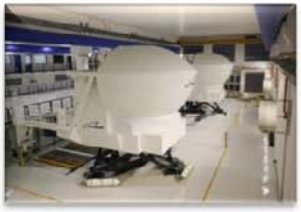
Surveilled by:
UNAVIAcert
Italian Military Authority (DGAA-Direzione Generale Armamenti Aeronautici)



Pilots:

- Approved FTO/TRTO as JAR FCL 2 for ab-initio pilots and VFR and IFR Conversion-to-Type.
- Approved FAA Training Centre according to FAA Part 142 since June 2007 for AB/AW139.

Surveilled by:
ENAC (Ente Nazionale Aviazione Civile – Italy)
Federal Aviation Administration (USA)



Simulators:

- Approved JAR-STD-1H Level D since 2006.
- Approved FAA Level D (as per AC120-63) since 2007.

Surveilled by:
Federal Aviation Administration (USA)
ENAC (Ente Nazionale Aviazione Civile – Italy)



Technicians:

- Approved MTO as per EASA Part 147 for Basic and Type Training.
- Approved MTO as per Canadian Aviation Authority Standard 566.
- Approved MTO as per CASA Australia CAR 30 and 100.66.

Surveilled by:
ENAC (Ente Nazionale Aviazione Civile – Italy) on behalf of EASA
TCCA (Transport Canada – Canada)
CASA - Australia

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AWTA MAIN CERTIFICATIONS

The AWTA offers a complete range of courses to meet all the customer needs in terms of training.



All teaching materials used during the courses are developed by applying state-of-the-art technologies and teaching methodologies.

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AWTA TRAINING PROCESS

Courses for Pilots

The Training Academy is an organisation certified by JAR, the Italian Civil Aviation Authority (E.N.A.C.) in accordance with JARFCL2) and FAA (as a Training Center Part 142).

With its different types of helicopters and the level of professionalism of its instructors and pilot instructors, the Training Academy offers the highest range of services which run from ab-initio training to pilot and basic VFR and IFR training to the highest level of qualification required by the complexity of modern military helicopters. All courses comply with the standards established by JAR-FCL2 and/or 14CFR Part 142.

Course Catalogue and Calendar available on http://www.agustawestland.com/content/customer-training

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TRAINING COURSES

Technical Courses

The Training Academy has been approved as a maintenance-training centre and certified by ENAC in accordance with regulation EASA Part147.
The technician courses are arranged to support the complete Agusta Helicopter fleet: AB212, AB412, CH47, A119/AW119Ke, A109 Series, A129, AW139, EH101 and NH90.

Basic courses are available for technical personnel without experience and Type Rating courses are available for personnel who have already done technical training.
All courses comply with the standards established by JAR 66/147. Here follows a list of the different levels of training which can be undertaken at our "Academy":

- **Category A:** line mechanic certificate (turbine helicopters)
- **Category B1.3:** line technician certificate (turbine helicopters)
- **Category B2**: line technician avionics certificate
- **Category C**: basic maintenance certificate

Course Catalogue and Calendar available on http://www.agustawestland.com/content/customer-training

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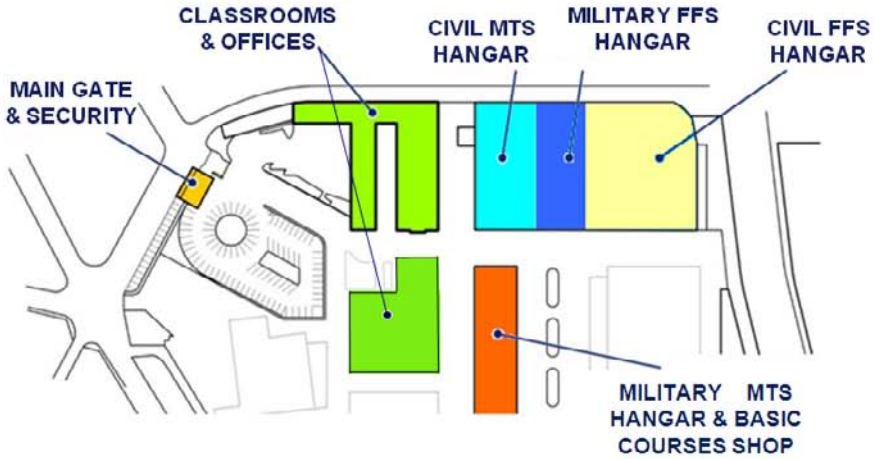
TRAINING COURSES

Special Courses

• HUMS	1 week

On demand …

• FLOATS & RAFTS (AEROSEKUR)	1 week
• STRUCTURAL REPAIR	2 weeks
• BLADES REPAIR	1 week



AW139 & AW109E Full Flight Simulators (FFS)





AW139 & AW109E Maintenance Training Simulators



Classrooms for Ground Courses & Theorethical Elements

ICN-3E-A-000000-G-A0126-00029-A-01-1

AWTA ITALY FACILITY

AW139 and AW109E Maintenance Training Simulators (MTS) offer practical and efficient alternatives to the real aircraft for mechanical and avionics technicians and include features that enhance learning.



Practical maintenance activities are performed in accordance with Interactive Electronic Technical Publications (IETP) available on workstations.

ICN-3E-A-000000-G-A0126-00030-A-01-1

MAINTENANCE TRAINING SIMULATORS (MTS)

Rotorsim is jointly owned by AW and CAE. It began offering comprehensive training for AW109 operators in 2006, followed by a comprehensive training program for AW139 operators. The Rotorsim area features three CAE-built full-mission simulators.



AW109E Power Simulator
JAA Certification (through the Italian Civil Authority - ENAC) JAR-STD-1H Level D - June 2006
FAA Qualification - Level D (as per AC120-63) - Feb 2007

AW139 Simulator
JAA Certification (through the Italian Civil Authority - ENAC) JAR-STD-1H Level D - Dec 2006
FAA Qualification - Level D (as per AC120-63) - Feb 2007

AW109LUHS & AW109N Simulator
Military / Para-Military
Full Flight Simulator Level D

ICN-3E-A-000000-G-A0126-00031-A-01-1

ROTORSIM - FULL FLIGHT SIMULATORS (FFS)

Flight Training activity with highly qualified Instructor Pilots is performed in the AW Airfields of Vergiate and Cascina Costa.



ICN-3E-A-000000-G-A0126-00032-A-01-1

FLIGHT TRAINING

The AWTA has classrooms with video-projectors, real helicopter components and Interactive Whiteboards.
School rooms and workshops are furnished with training aids, CBT, technical mock-ups and Electronic Training Book (eTB)



The AWTA boasts a large team of pilots and theory instructors (mechanics – avionics) who cover the huge range of courses on offer. All the staff is highly qualified and has in-depth experience in the aeronautical sector.

ICN-3E-A-000000-G-A0126-00033-A-01-1

CLASSROOM AND MULTIMEDIA INSTRUCTION

- Civil certified type rating courses
  - AW109
  - AW139
  - AW119



- Significant investment being made to extend the courses available for the AW139 with new Rotorsim Full Flight Simulator
- Will expand further to meet growing demand in the USA



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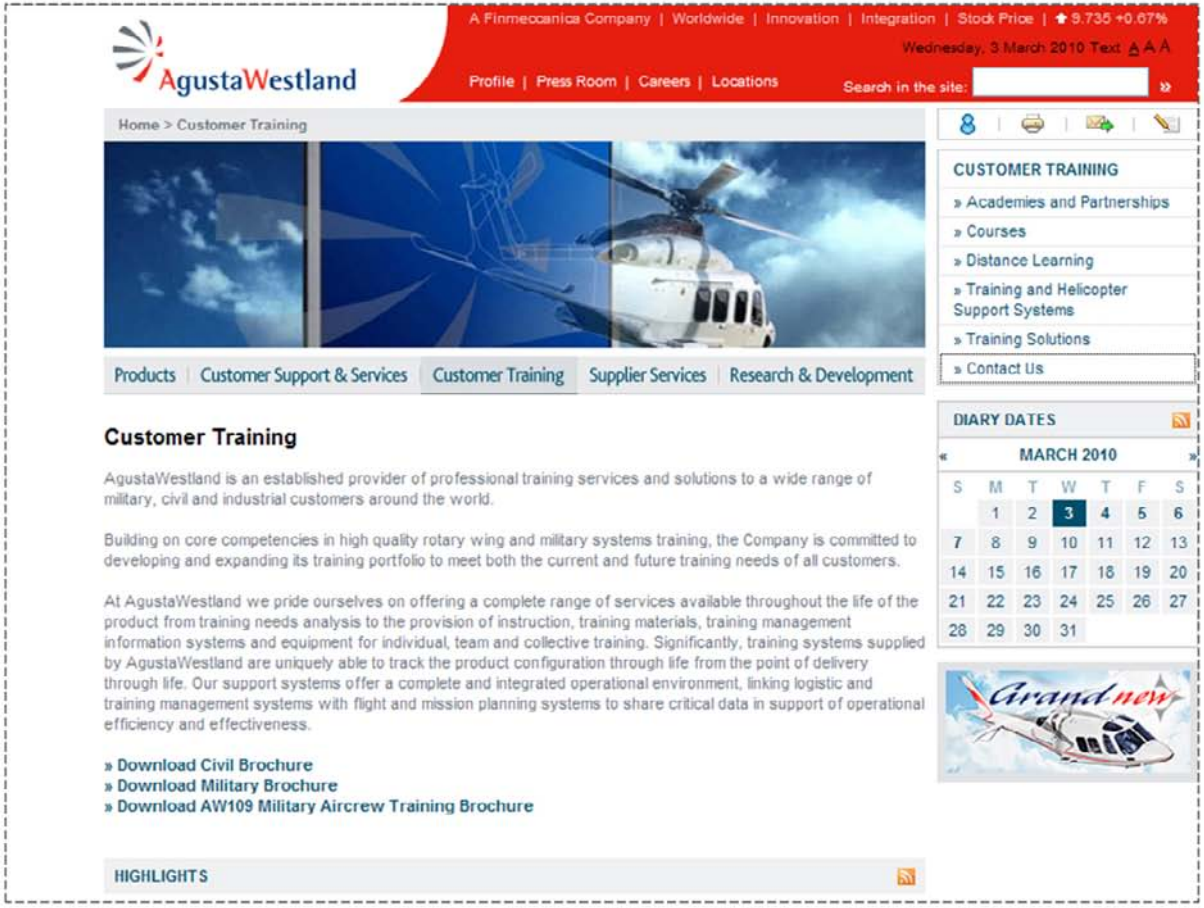
US TRAINING ACADEMY - PHILADELPHIA

- FAA Level D AW139 Simulator #2
- 4-Axis Auto Pilot
- Type Rating, Recurrent, and Dry Training capabilities
- Rotorsim USA operates under AgustaWestland TRTO in accordance with FAA Part 142 and JAR-FCL2 regulations
- Located within CAE's North East Training Center in Morristown, NJ, USA
- Contact **AgustaWestland Training Academy** to arrange Rotorsim USA training



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ROTORSIM USA



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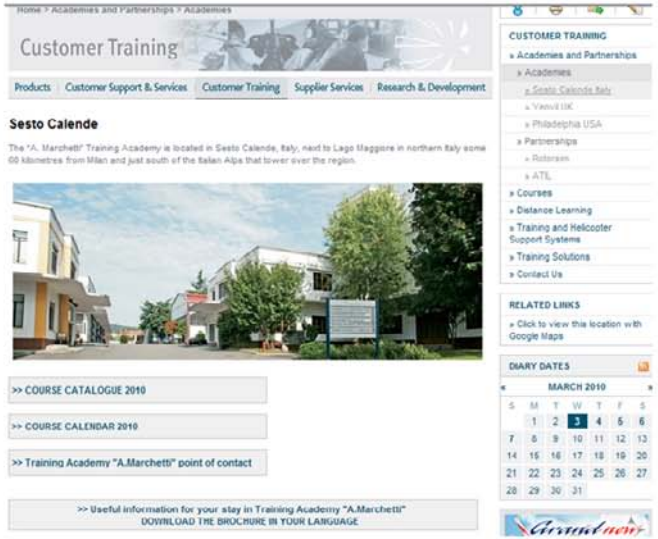
AW CUSTOMER TRAINING WEBSITE

- Course Catalogue
- Course Calendar
- Registration & Points of contact
- General Terms & Conditions
- Useful information for stay in Training Academy "A. Marchetti" – Brochure in eight languages
- Simulator availability

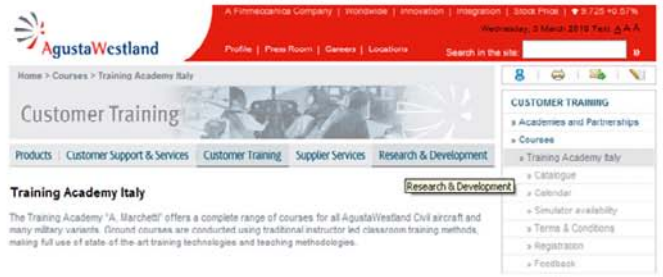
Under development

- Ground courses availability

http://www.agustawestland.com/content/sesto-calende



http://www.agustawestland.com/content/training-academy-italy-0



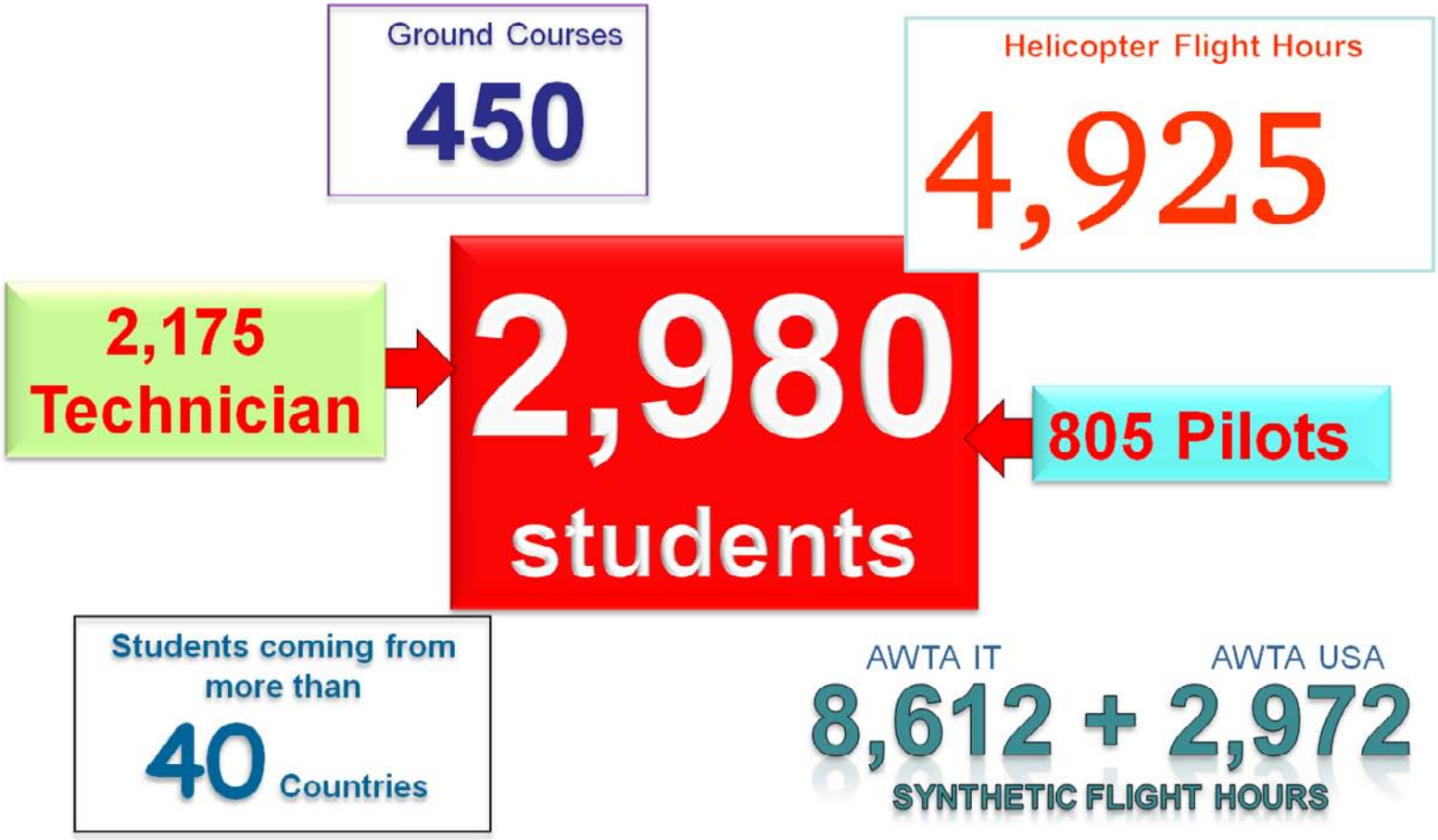
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TRAINING ACADEMY ITALY WEBSITE

Training Resources	Availabilities AWTA IT	Availabilities AWTA USA
Ground Instructors	**48** (Basic Modules, TA3, TB1.3 & TB2)	**10**
Classrooms	**18** (6/10/18 seats - 230 students capacity)	**7**
Maintenance Training Simulators (MTS)	**2** (A109E & AW139)	**1** (AW139)
Full Flight Simulators (FFS)	**4** (A109E, AW139, A109LUHS & A109N – In addition, next October 2010 will be operative the NH90 FFS)	**1** (AW139)
Pilot Instructors	**39** (both real and simulated flight activity)	**15** (both real and simulated flight activity)
Helicopters	**5** (N° 2 AW139, N°1 A109E, N°1 A109S and N°1 AB206, located at Vergiate & Cascina Costa airfields)	**1** (N°1 AW119 at Philadelphia)

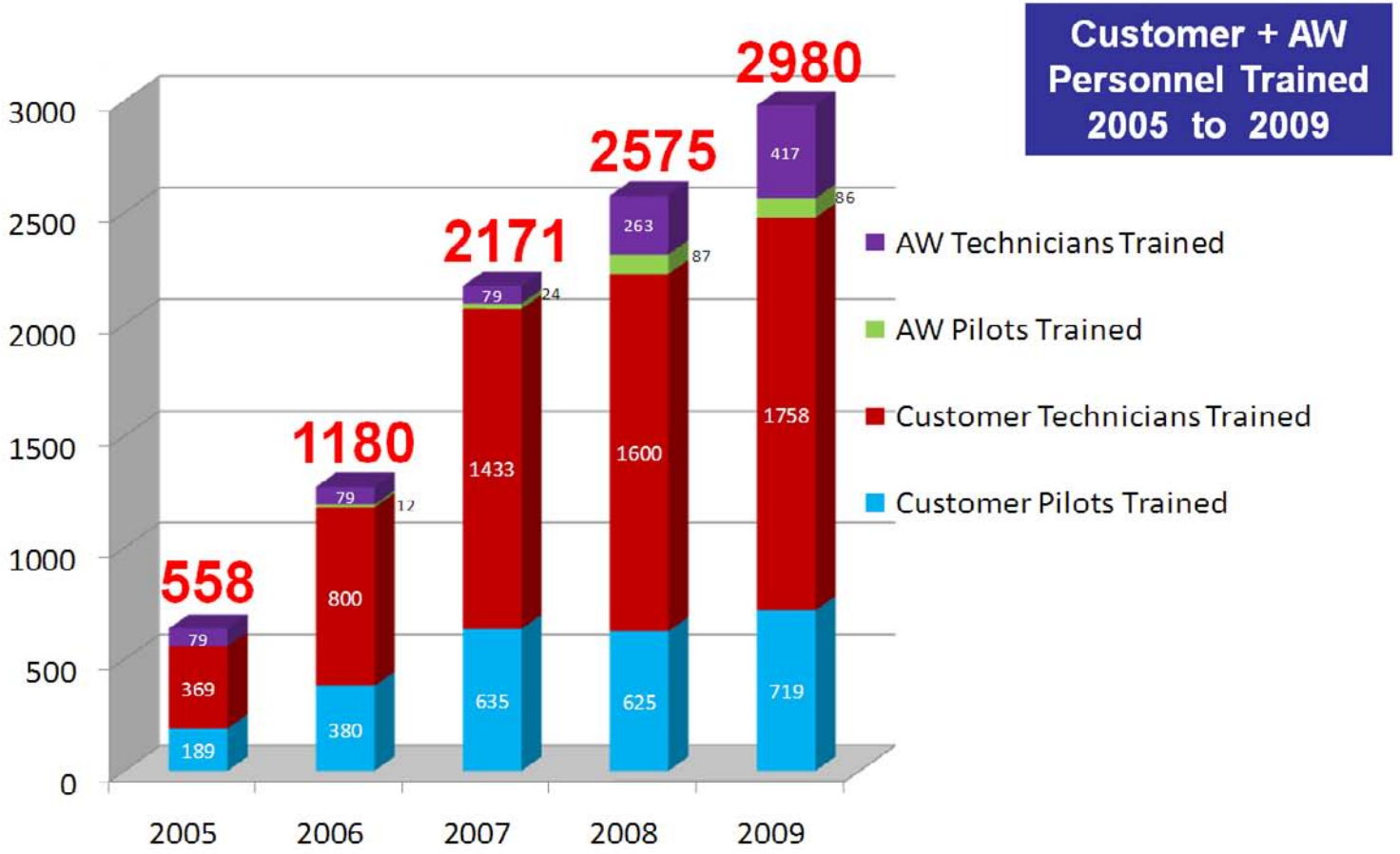
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AWTA: THE NUMBERS



ICN-3E-A-000000-G-A0126-00039-A-01-1

AWTA: THE NUMBERS (YEAR 2009)

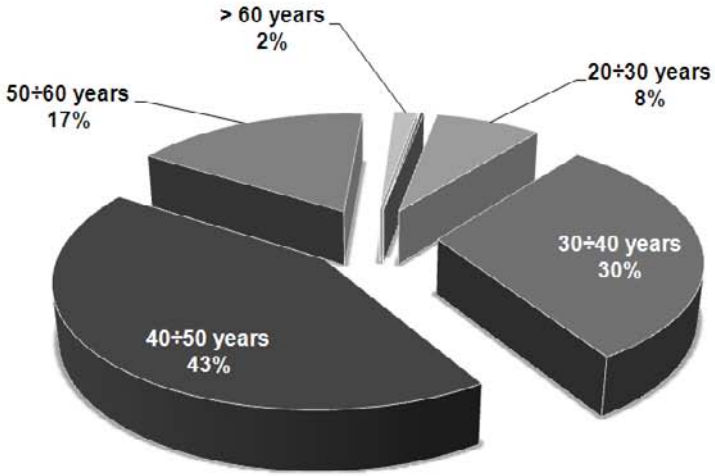


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AWTA: THE NUMBERS

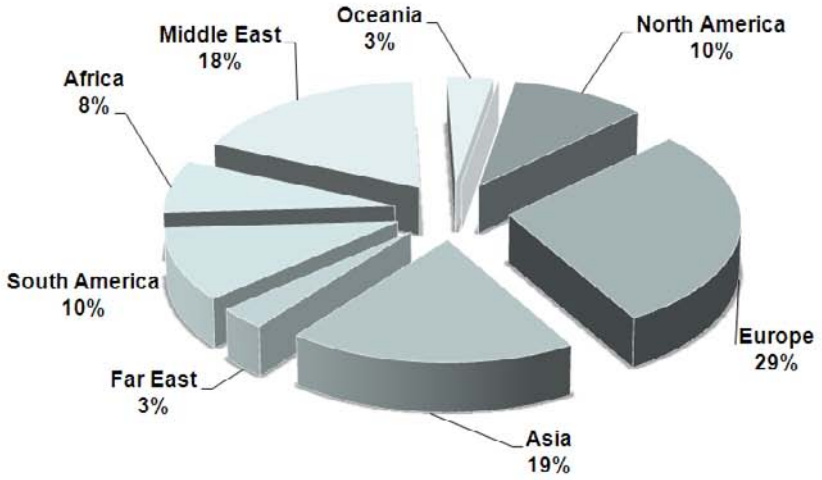
Sample population of students

Period of Sample	...	Oct 2009	Nov 2009	Dec 2009	Jan 2010	Total
Evaluation Forms collected	...	88	44	50	52	380



Age

World region of provenience



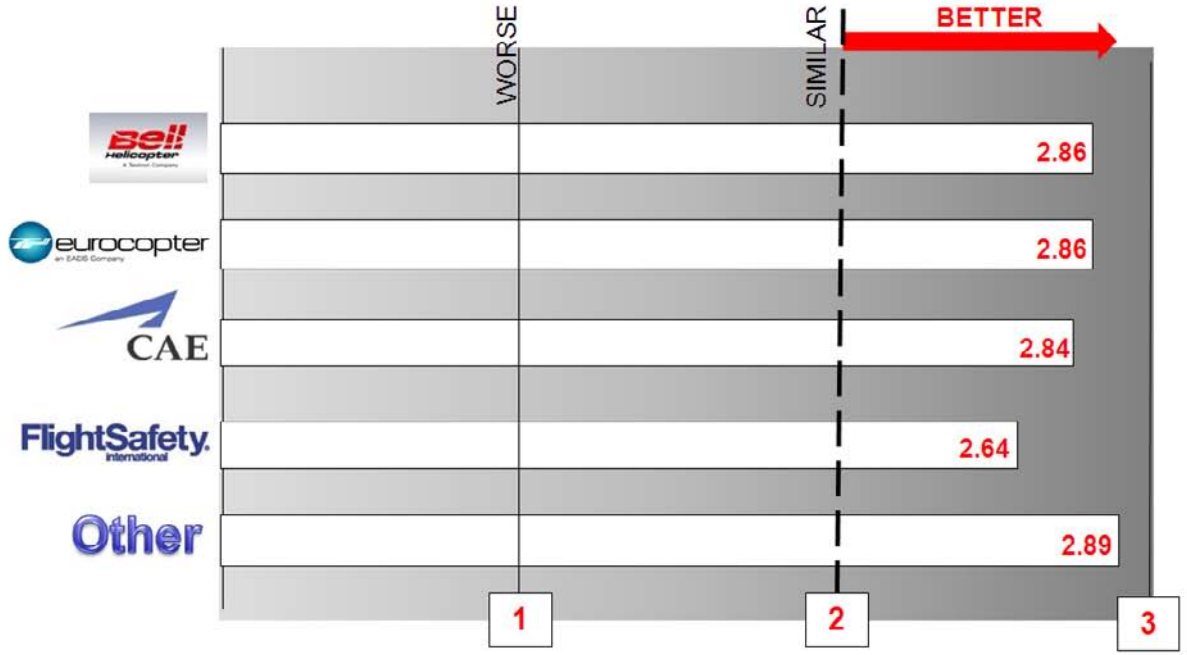
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AWTA: THE NUMBERS (YEAR 2009)

Competitors comparison

Based on the Evaluation Forms filled at the end of AWTA Courses, the graphic shows how students compare AW with Training received by its competitors.

Level **1** = WORSE; Level **2** = SIMILAR; More than **2** = BETTER



ICN-3E-A-000000-G-A0126-00042-A-01-1

AWTA: THE NUMBERS (YEAR 2009)

SECTION 00 - HELICOPTER - GENERAL

GENERAL DESCRIPTION

The AW139 is a large twin turbo-shaft engines multi-purpose helicopter designed for the following:

PRIMARY ROLES

- transport of passengers
- transport of material
- emergency Medical Service (EMS) air ambulance

SECONDARY ROLES

- air taxi
- transport of underslung loads
- SAR - Search And Rescue (over land, sea, mountain)
- off-shore
- fire-fighting
- ecologic, radiologic, surveillance
- civil protection
- law enforcement
- paramilitary

NOTE A

AW139 abd AB139 are two names for the same product. They identify two batches of aircraft manufactured in conformity with a unique Type Certificate Data Sheet.
AB139: up to s/n 31054
AW139: from s/n 31055 onward

JAR CERTIFICATION

The AW139 is certificated by European Aviation Safety Agency (EASA) in accordance with JAR 29 for Large Rotorcraft Categor A (Appendix B - IFR, and Appendix C - Icing conditions) and Category B
Certification includes:

- Taxiing from prepared surfaces
- Cat. 1 or 2 ILS approaches
- Overwater operations (no amphibian operations)
- Carriage of external loads
- Cold weather operations
- Hot and high operations.
- IFR single pilot operations
- VFR night single pilot operations

Installation of dedicate optional kits may be necessary to perform some of the certified operations.

STRUCTURE

The cockpit, the front part of the fuselage, include the pilot (RH) and co-pilot or passenger (LH) side-by-side seating positions, adjustable crashworthy seats, the instrument panel and a forward opening hinged door on each side with emergency release.
The middle part of the fuselage is composed by cabin and rear fuselage.
The cabin include the passenger seating area, a sliding door on each side, the fuel tanks housing, the main gearbox and engine attachments, main landing gear attachments and housing.
The rear part of the fuselage includes the baggage bay.
Cabin size is adequate for the following internal arrangements:

- **Normal density civil transport** 2 +11 passengers (all looking forward) in crashworthy seats.
- **Normal density civil transport** 2 + 12 passengers
- **High density civil transport** 2 +15 passengers.
- **Civil EMS** 6 litter patients, 2 non-litter patients and 2 medical attendants.

The tail boom, the final part of the fuselage, shall include the stabilizer, the fin, the intermediate and tail gearbox attachments.
The parts mentioned above are considered primary structure.
The remaining parts of the fuselage, are considered secondary structure and include:

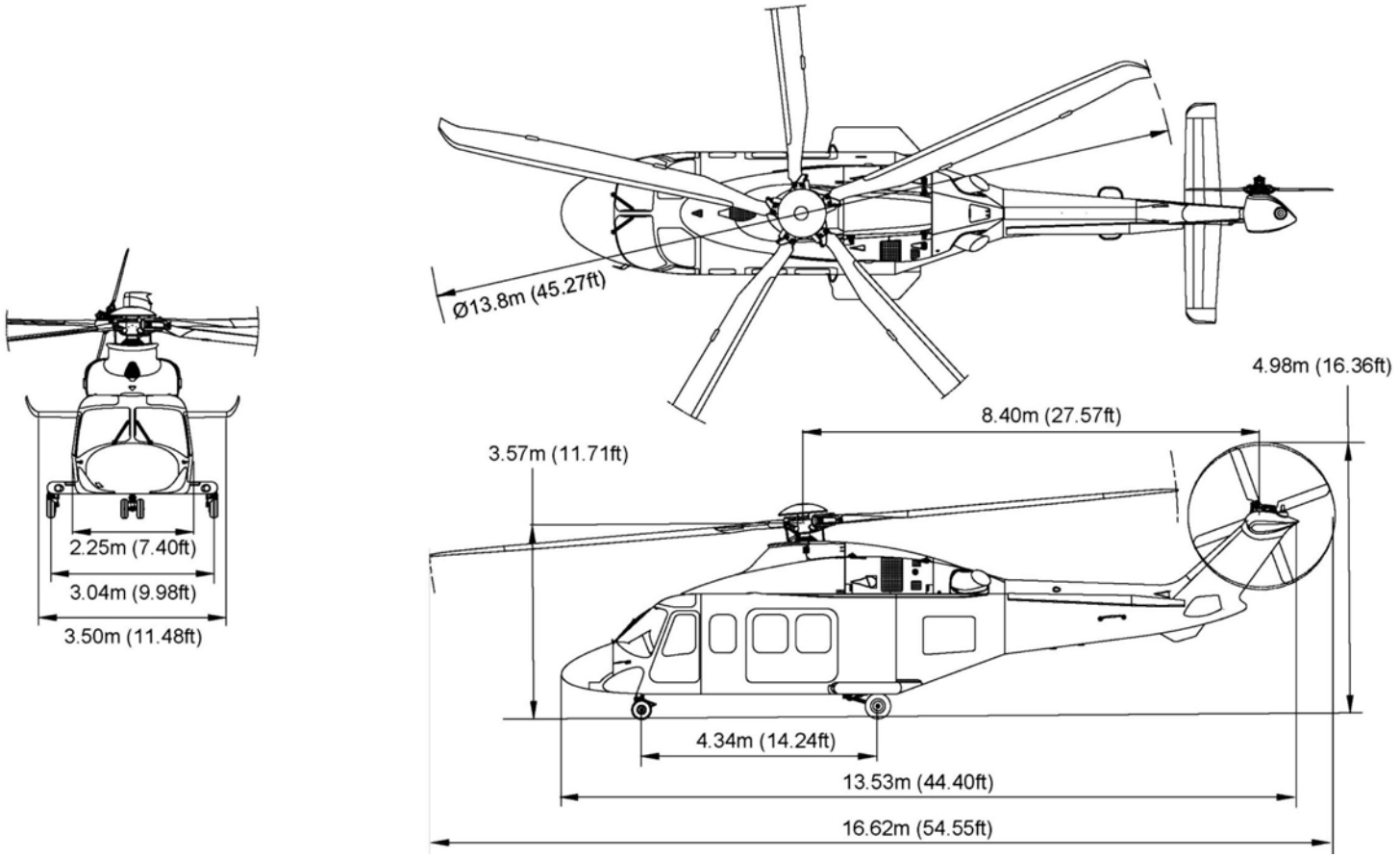
- pilot doors
- passenger doors
- inspection doors
- transparencies
- main gearbox cowlings
- fairings
- radomes.

OPERATING PERFORMANCE

- Vne (AEO) – Up to Hd = 6000 ft..... 167 KIAS
- Rate of Climb (@ AEO, MCP, 6400 Kg)..... 2150 fpm
- Service Ceiling (Hp or Hd) ..... 20000 ft
- Hovering Ceiling (IGE & OGE, @ MCP, AEO)..... Up to 18000 ft
- Takeoff & Landing Ceiling (Hp or Hd)..... Up to 14000 ft
- Operating Temperature..... -40°C to ISA + 35°C

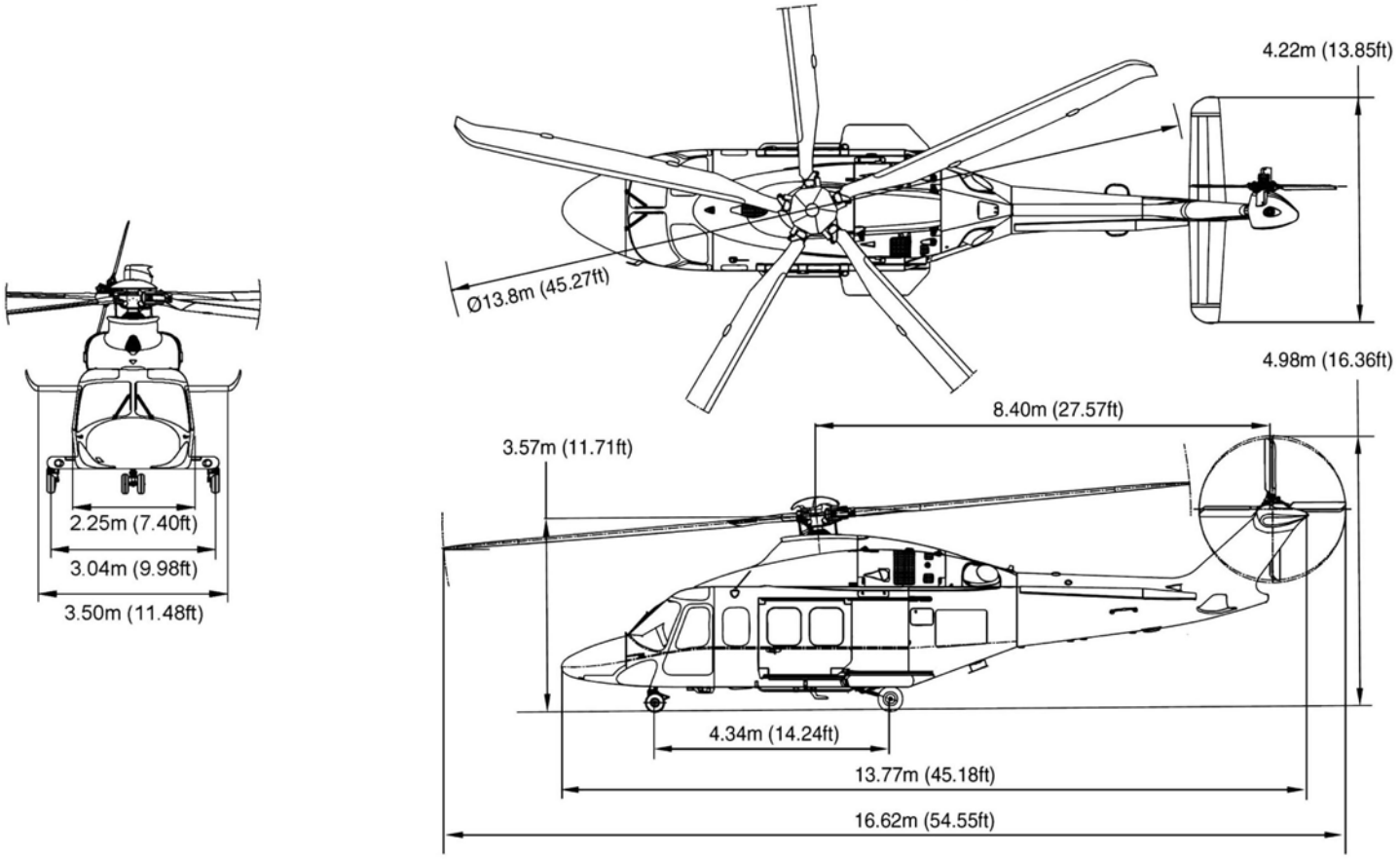
WEIGHT LIMITS

- Maximum gross weight for CAT B take-off/landing..... 6400 Kg
- Increased Gross Weight (Kit P/N 4G0000F00111 – see RFM Suppl-50)..... 6800 Kg
- Minimum flight/rotor-turning gross weight..... 4400 Kg



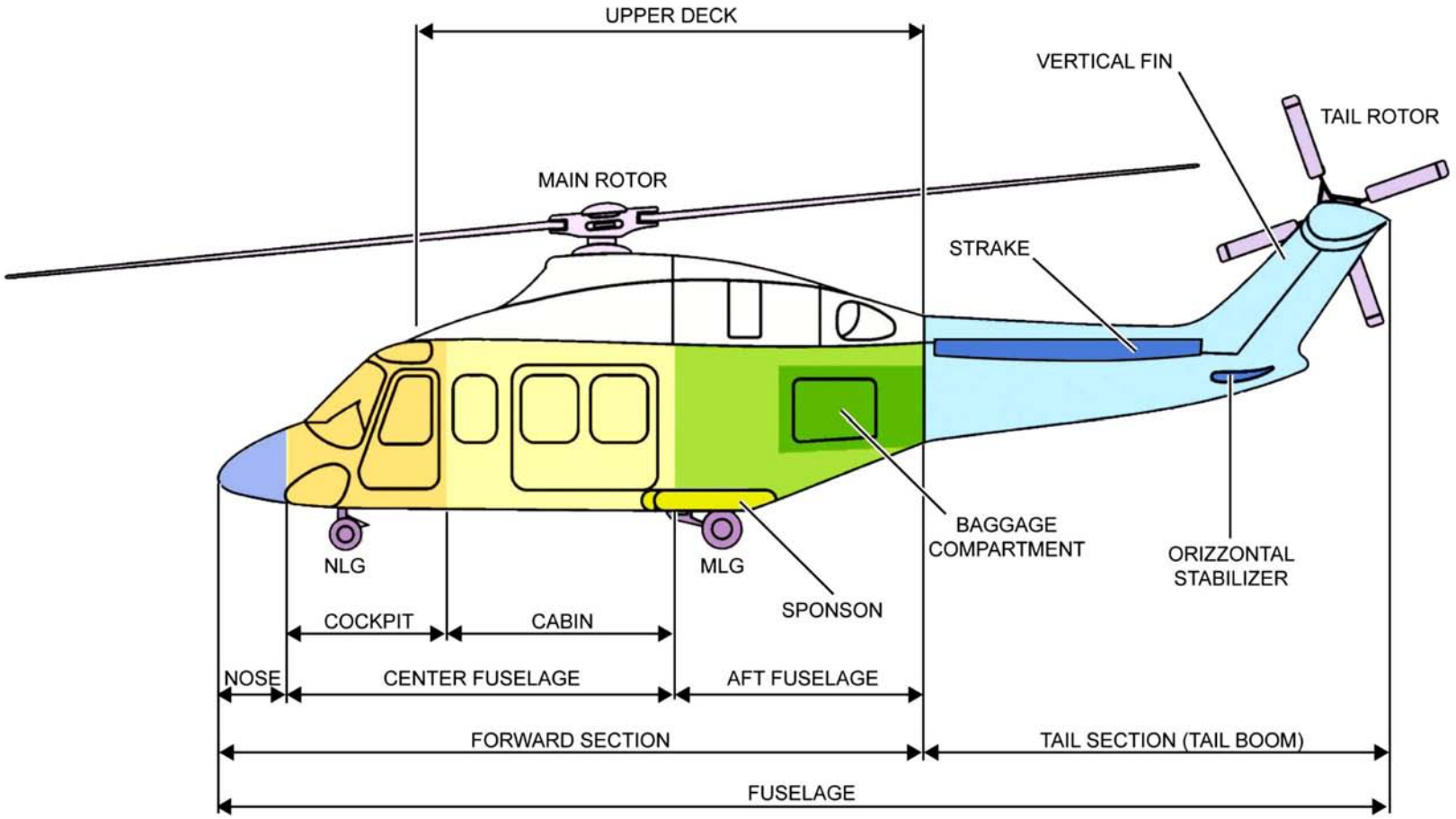
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PRINCIPAL DIMENSIONS (SHORT NOSE VERSION)



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PRINCIPAL DIMENSIONS (LONG NOSE VERSION)



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AW139 HELICOPTER - MAJOR ZONES



CABIN SLIDING DOOR

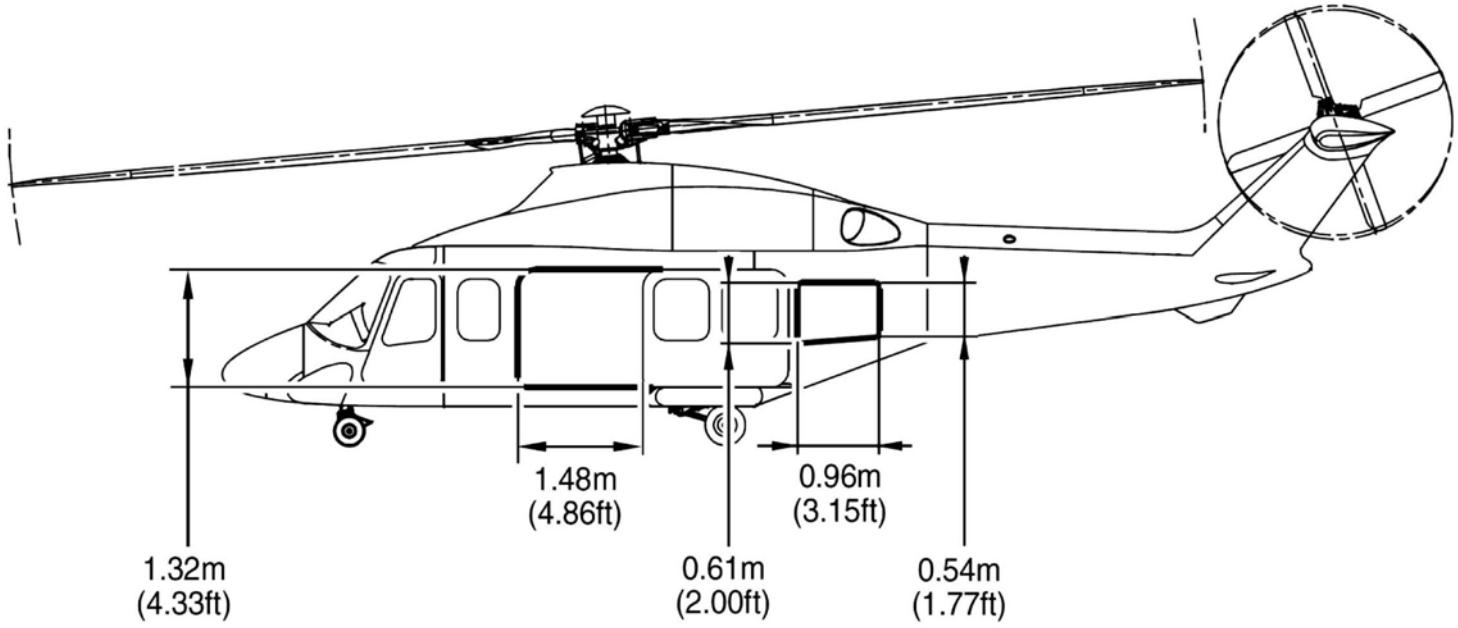


PILOT DOOR

NOTE: Same arrangement on both sides

ICN-39-A-000000-G-A0126-00400-A-01-1

ACCESS TO COCKPIT AND CABIN



ICN-39-A-000000-G-A0126-01600-A-01-1

CABIN AND BAGGAGE DOOR DIMENSIONS

EMERGENCY EXITS

Four emergency exits per side are provided by jettison on the following windows:

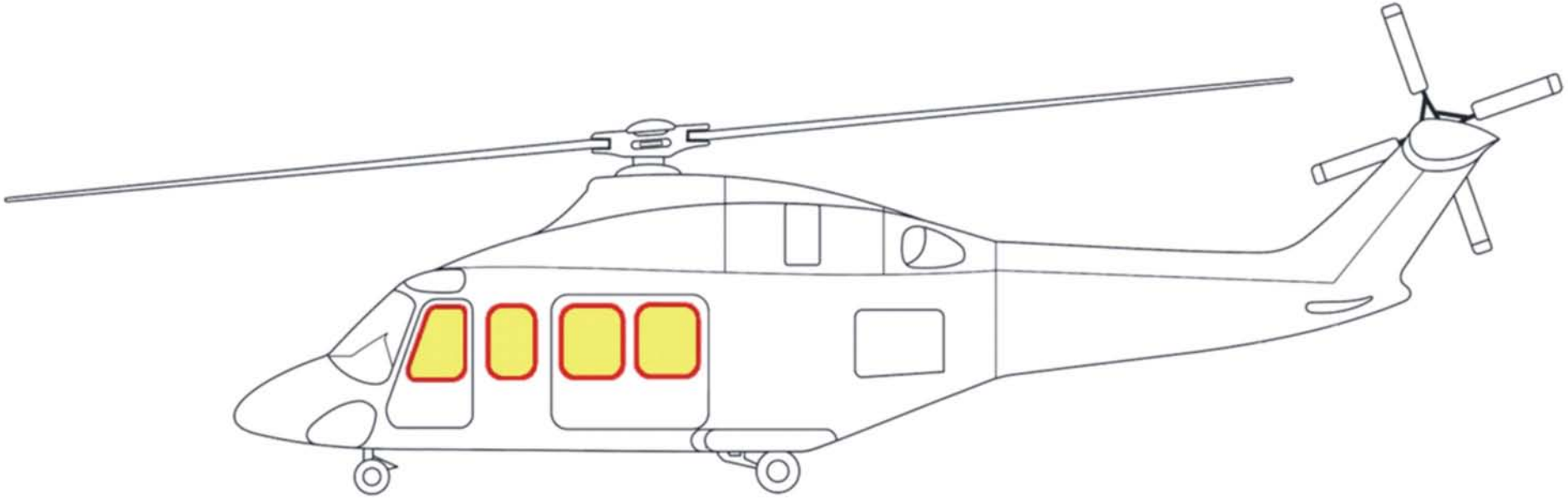
- Cockpit Door Window (1 per side)
- Cabin Window (1 per side, on the fixed frame)
- Cabin Door Windows (2 per side)

A red strap in the internal bottom side of each emergency exit window permits removal of the rubber cord that holds the window transparent panel in place. The emergency exit is made free by pushing the transparent panel out acting on its bottom left corners.

OPTIONAL EQUIPMENT

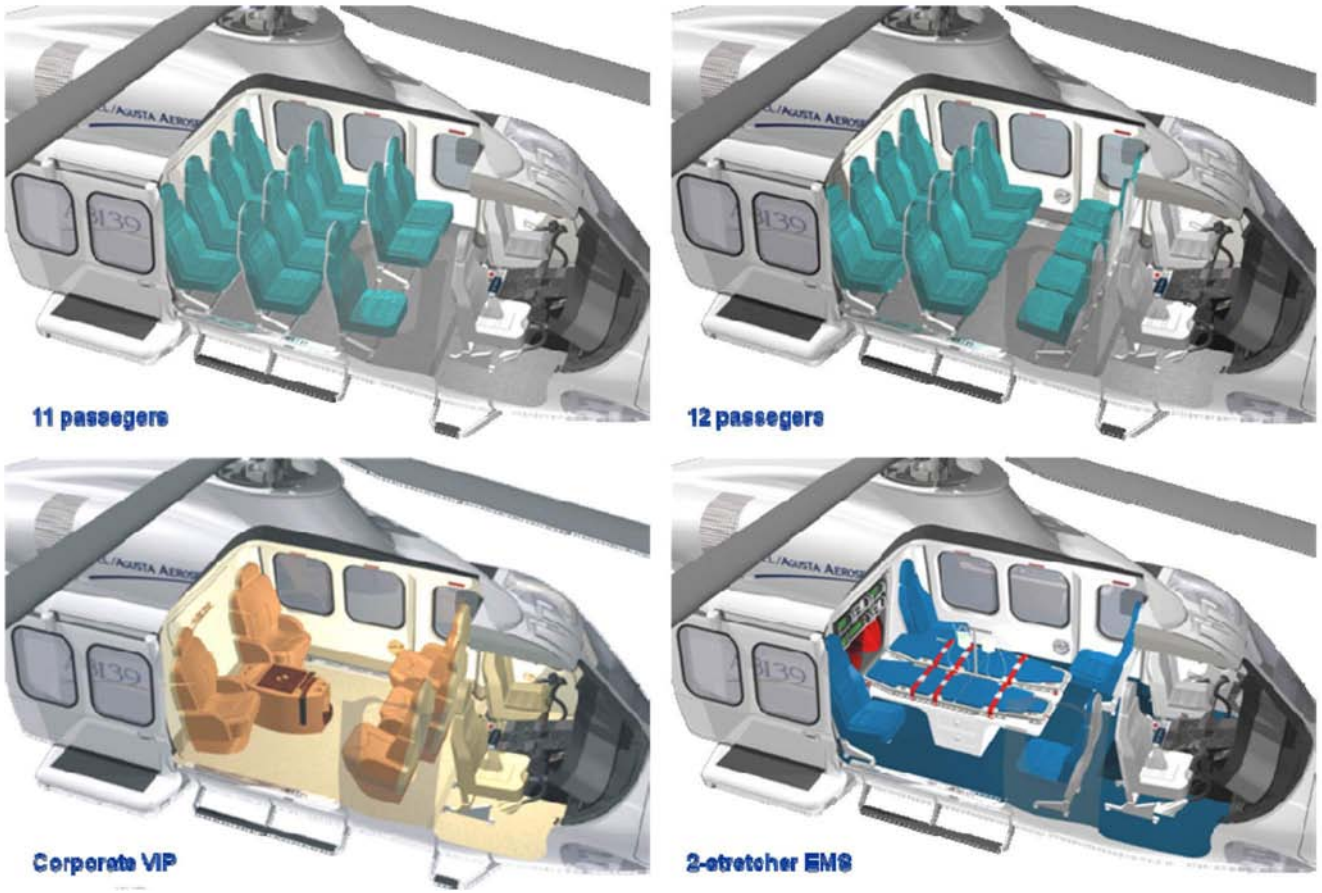
A wide selection of optional kits can be installed on the AW139 helicopter. Some of the available optional kits are listed below:

- Rotor Brake
- Air Conditioning System
- 4-Axis Flight Director (Basic, Enhanced, SAR versions)
- Engine Air Particle Separator (EAPS)
- HF Radio
- V/UHF Radio
- Weather Radar (P660, P701, RDR-1550B)
- Traffic Collision Avoidance System (TCAS)
- Enhanced Ground Proximity Warning System (EGPWS)
- Emergency Floats and Liferafts (Ditching configuration)
- Deployable ELT
- Hoist
- Cargo Hook
- Auxiliary Fuel Tank
- Cabin Bubble Window
- Cabin Sliding Windows
- Heated Windshield
- External Public Address System (External Loudspeakers)
- Second Radar Altimeter
- Snow Skids
- Slump Pads (soft terrain pads)
- Hinged passenger doors
- Digital Map System (Sky Force Observer Mark II, EURONAV IV, EURONAV V)
- Nightsun Searchlight (SX-16, SX-5, II XP)
- FLIR (SEA FLIR II, Star Safire III, Ultraforce 350, TALON)



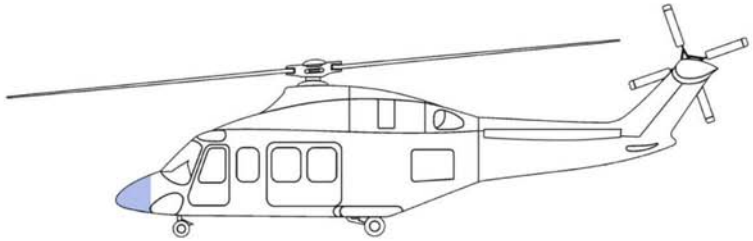
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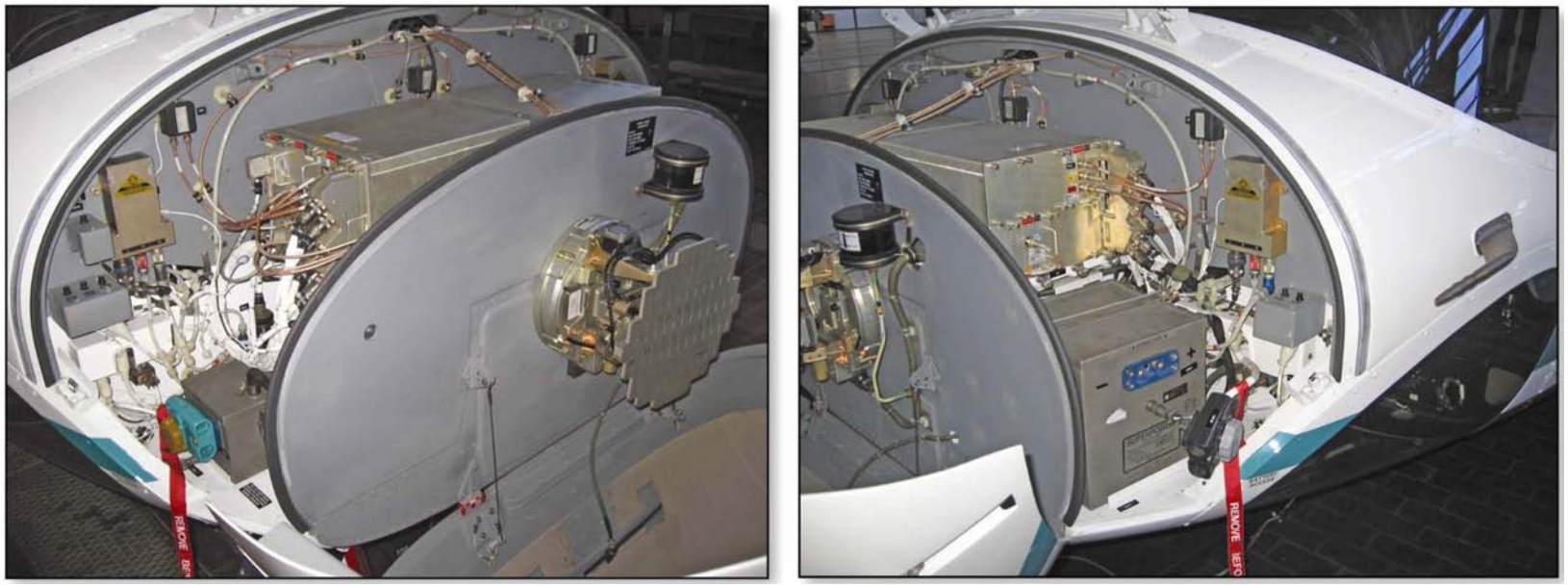
EMERGENCY EXITS



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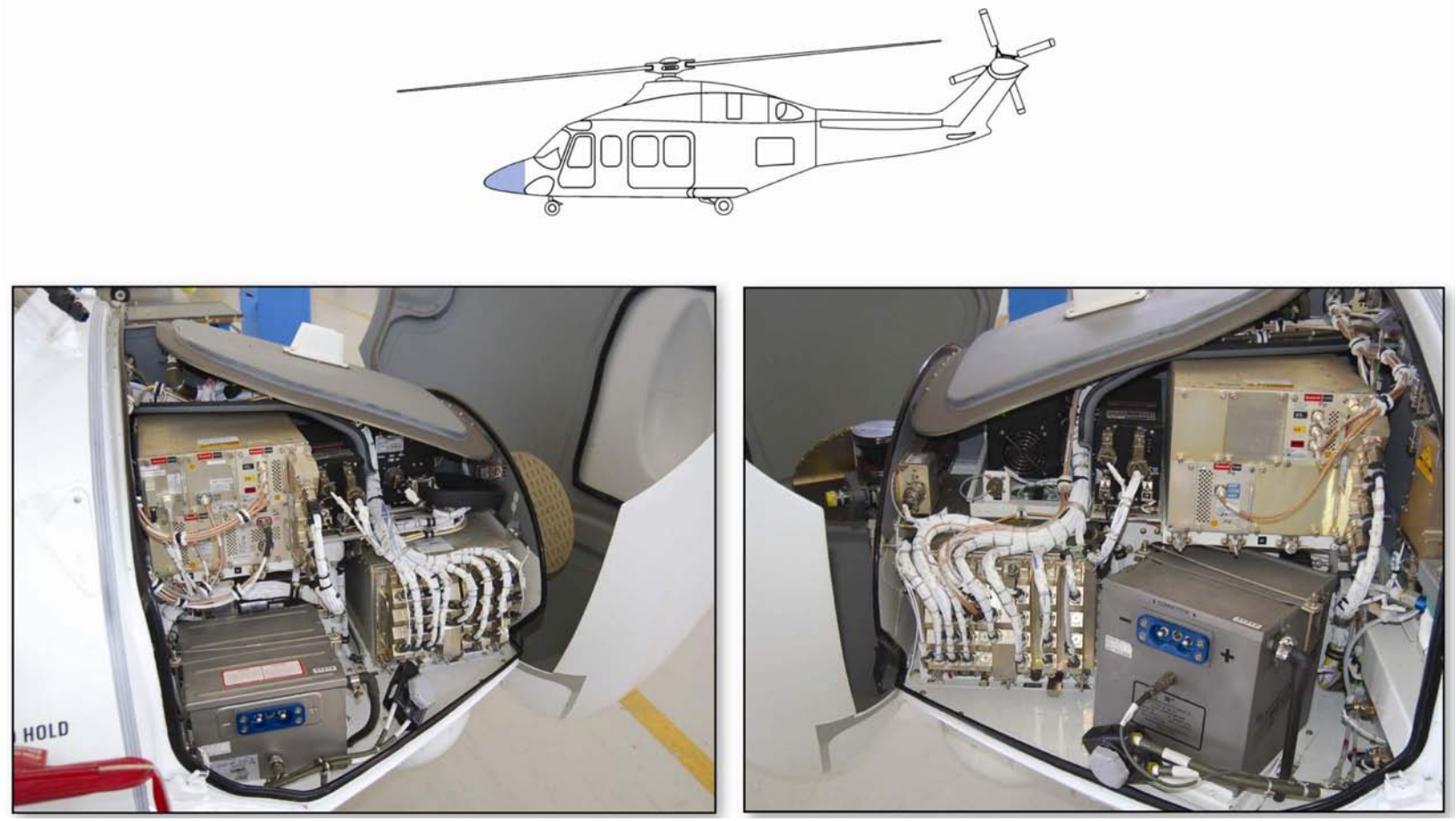
CABIN - CONFIGURATIONS





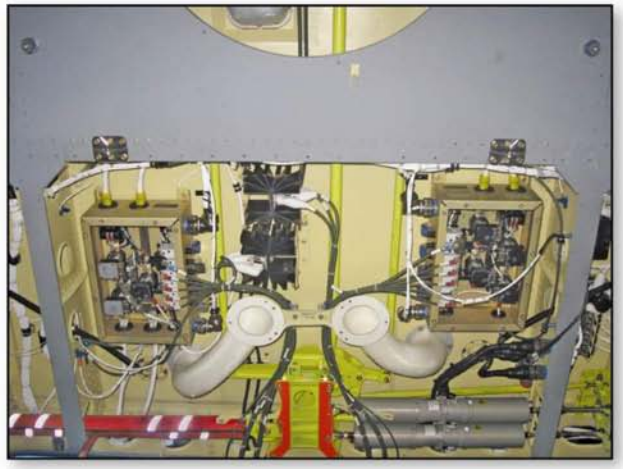
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ELECTRICAL / AVIONIC BAY – SHORT NOSE CONFIGURATION



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ELECTRICAL / AVIONIC BAY – LONG NOSE CONFIGURATION



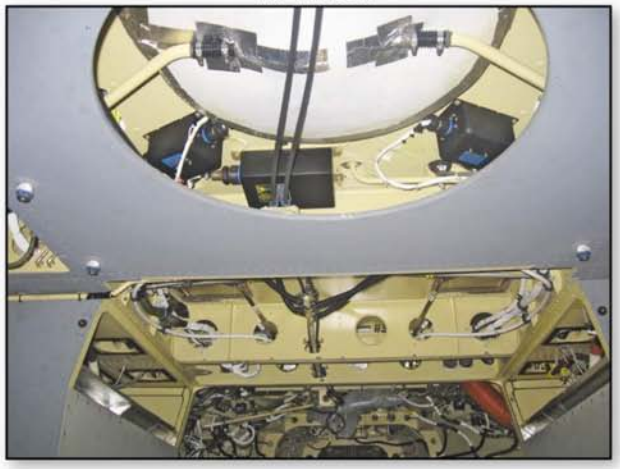
FORWARD AREA





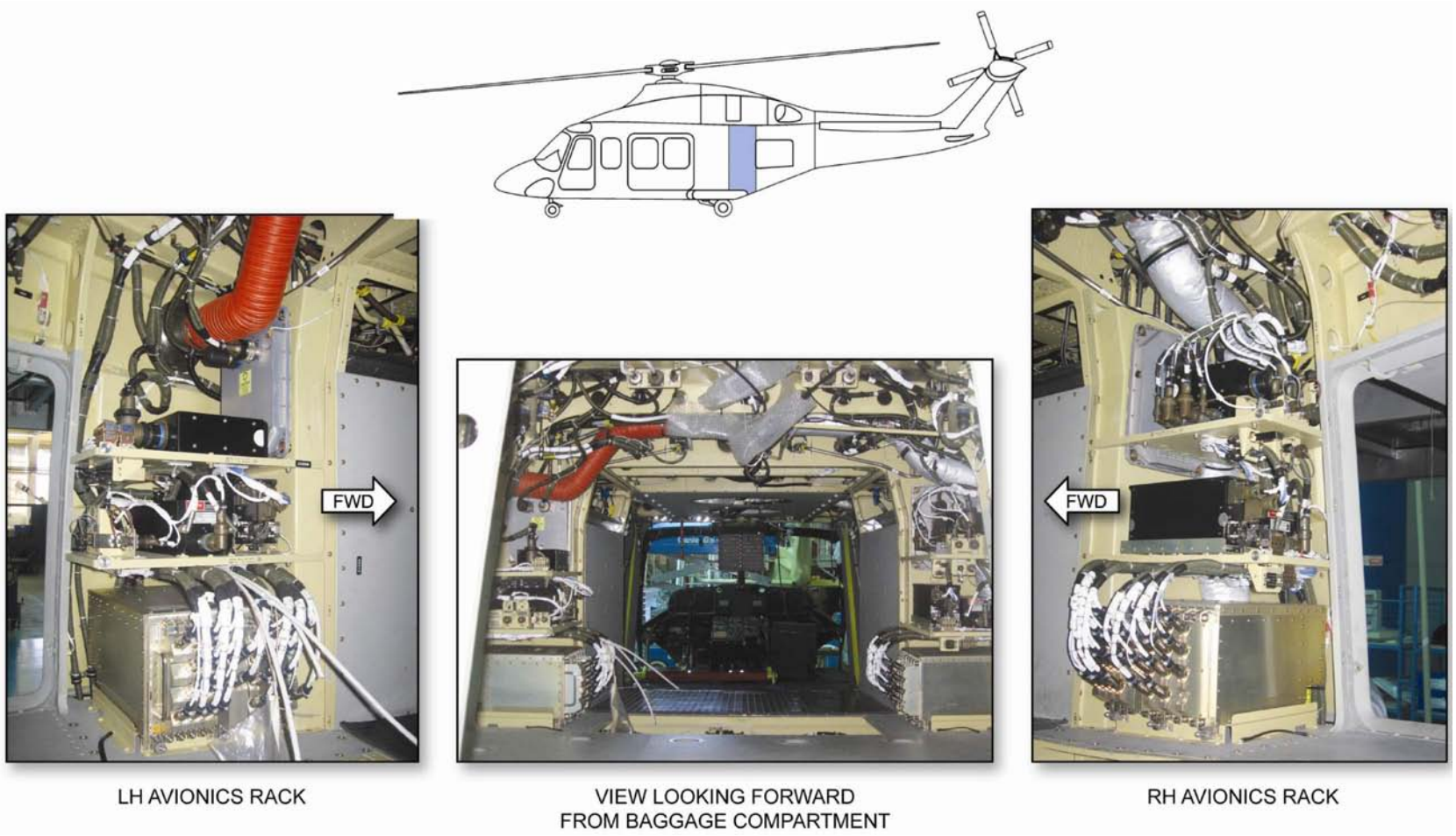
CENTRAL AREA

AFT AREA



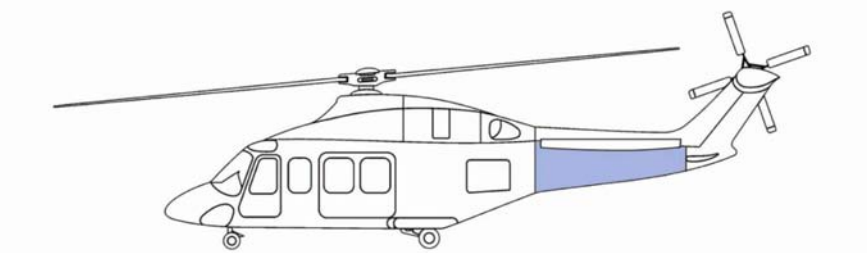
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ELECTRICAL / AVIONIC BAY – CABIN CEILING CONFIGURATION



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ELECTRICAL / AVIONIC RACKS CONFIGURATION





VIEW LOOKING AFT FROM BAGGAGE COMPARTMENT

ICN-39-A-000000-G-A0126-01614-A-01-1

ELECTRICAL /AVIONIC BAY TAILBOOM CONFIGURATION

SYSTEMS OVERVIEW

POWERPLANT

The power plant shall comprise: engines and related installation, fire detection and extinguishing system.

Engines

The helicopter is powered by two PT6C-67C turboshaft engines. Each engine is installed in a separated fireproof area above the cabin roof and supplies power to the drive system by means of a rotating shaft. The engines are connected to the airframe by means of two attachment points on the engine body and to the main gearbox by means of a tube and a gimbal joint.
Air is supplied to the engine via individual, side facing air inlets. The engines are started by a DC starter-generator.
Engine control is achieved via a control panel located in the cockpit and manual back-up of the engine control via pushpull cables.
The engines are provided with torque sensing and matching.

Fire Detection / Extinguishing

The fire detection system consists of a continuous wire detector installed in the powerplant fire zones, routed in a way that allows coverage of all critical areas.
The fire extinguishing system consists of directional flow valves which allow discharging the contents of one bottle while sealing the connection to the other bottle and the subsequent discharge of the second bottle in the same bay if required.

DRIVE SYSTEM

The drive system consists of the Main Rotor Drive System and the Tail Rotor Drive System.

Main Rotor Drive System

The Main Rotor Drive System mainly consists of the Main Gearbox (MGB) that is mounted on the roof of the cabin by means of four struts and an anti-torque device.
The MGB has three stages of reduction and includes a duplicated oil lubrication system. It provides the attachment points for the rotor brake coaxial with the tail rotor drive output.
The MGB drives three hydraulic pumps and other accessories.

Tail Rotor Drive System

The tail rotor drive system consists of three drive shafts driven by the MGB, the Intermediate Gearbox IGB and the Tail Gearbox TGB oil splash lubricated.
The main rotor is a five blades, fully articulated rotor.
The tail rotor is a four blades, fully articulated rotor.

ROTORS

The rotor system consists a main rotor (MR) and tail rotor (TR).
The main rotor is a five blades, fully articulated rotor.
The tail rotor is a four blades, fully articulated rotor.

HYDRAULIC POWER

The hydraulic power system consists of two independent circuits that supply hydraulic fluid at a nominal working pressure of 3000 psi (207 bar). Both hydraulic power system circuits supply the hydraulic power necessary to operate the flight control servo-actuators and the landing gear.

For ground test and malfunction condition, a shut-off valve allows to shut-off the flight controls circuits.
For safety reason, another shut-off valve allows to shut the landing gear circuits off, increasing the survivability of the flight control functions. Hydraulic power to the landing gear actuators is used to extend and retract the main and the nose landing gears.
The main and nose landing gear are maintained in UP position by the hydraulic pressure (no mechanical uplocks are provided).
The extended DOWN position is maintained with a mechanical locks in the main and nose gear actuators.

LANDING GEAR

The AW139 helicopter is provided with a fully retractable, nose tricycle landing gear equipped with shock absorbers.
Hydraulic actuators provide for extension and retraction of the landing gear. The operation is electrically controlled and hydraulically operated, both in normal and emergency conditions.
Two main landing gear (MLG) legs are installed in the sponsons: each MLG leg is provided with a single wheel and weight-on-wheel (WOW) microswitches.
An independent disk type braking system permits differential braking of the main wheels.
The nose landing gear (NLG) leg is free to rotate 360° when on the ground. An automatic mechanical centering device recenters the nosewheel at lift off if it is within ±115° from forward direction. The nosewheel can also be held centered by an electro-mechanic centering system.

FUEL

The fuel system includes crashworthy fuel tanks located in the rear area of the cabin. Each tank contains a booster pump, an engine feed line, a fuel and water drain valve. Filling with fuel is achieved by gravity.
The fuel selector manifold allows fuel to supply each engine separately or fuel both engines in a cross-feed condition.
The fuel quantity gauging system is composed by four capacity probes, a Fuel Computer Unit (FCU) and a fuel low level sensor for each tank. The fuel venting system consists of pipes for each tank designed to prevent fuel leakage in case of helicopter roll-over after crash landing.

ELECTRICAL POWER

DC power is generated by two 30V, 300A DC air cooled generators. Two batteries provides a back-up source of emergency power in the event that both generators fail and power for autonomous ground operations and engine starting.
The electrical power is delivered to aircraft systems by a dual system of distribution bus bars consisting of the Main (MAIN), Essential (ESS) and Non-Essential (NON ESS) busses.
Power from a DC external power source can also be connected to the aircraft busses.

LIGHTING

The lighting system includes interior and exterior lights.
The interior lighting subsystem consists of instruments lighting, panels lighting, overhead panels lighting and cockpit utility lighting.
The exterior lighting subsystem consists of anti collision lights, position lights and landing lights.
NVG compatibility can be provided at customer option.

COCKPIT AND CABIN VENTILATION SYSTEM

The ventilation system consists of two separate subsystems for cockpit and cabin ventilation. Heating and cooling systems are provided as optional kits.

AVIONICS

The PRIMUS EPIC® system is an integrated avionics system that includes the following sub-systems necessary to operate:

- Auto-Pilot
- Flight Management System
- Communications
- Indicating and Recording Systems
- Aural Warning Generator
- Navigation
- Crew Alerting System
- Central Maintenance Systems (CMS)

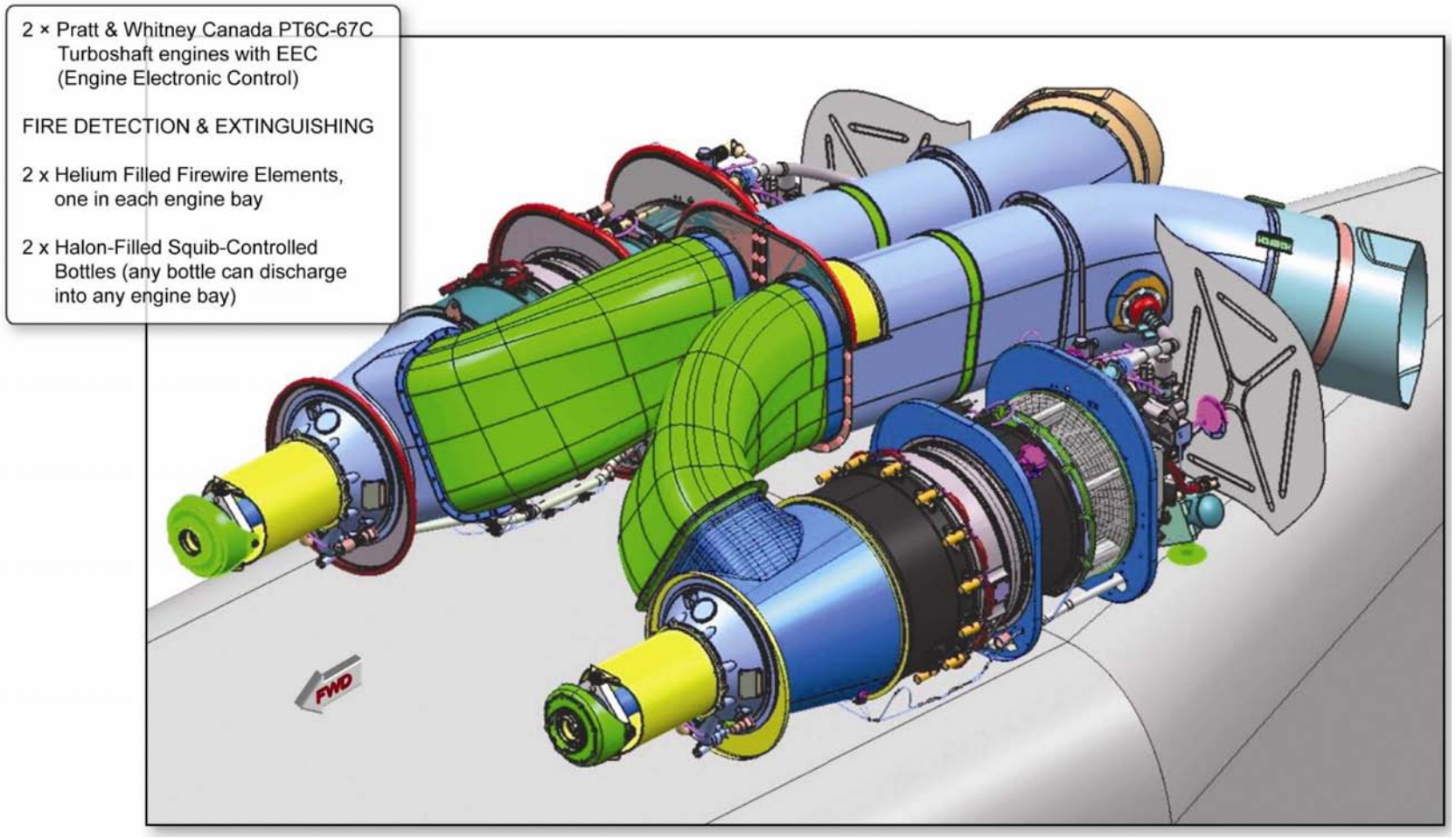
The PRIMUS EPIC® system is integrated into:

- two Modular Avionics Units (MAU)
- four flat panel color LCD Display Units (DU) to show data in the cockpit
- two Modular Radio Cabinets (MRC) that include the following radios:
  - VHF-COMM
  - VOR/ILS
  - ADF
  - DME
  - Transponder (XPDR)

The MAUs, the DUs and the MRCs are directly connected to each other via a bi-directional digital data bus named Avionic Standard Communication Bus-D (ASCB-D).
A LAN digital bus also interconnects the same units for maintenance purposes.
In addition to the PRIMUS EPIC® integrated avionics system, the following systems are also part of the aircraft standard avionics configuration:

- Standby Instrument
- Emergency Locator Transmitter (ELT)
- Flight Data Recorder / Cockpit Voice Recorder FDR/CVR)

A great variety of optional systems is available to fulfil Customer's operational requirements.



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POWER PLANT



MAIN ROTOR



MAIN ROTOR DRIVE SYSTEM



TAIL ROTOR



TAIL ROTOR DRIVE SYSTEM

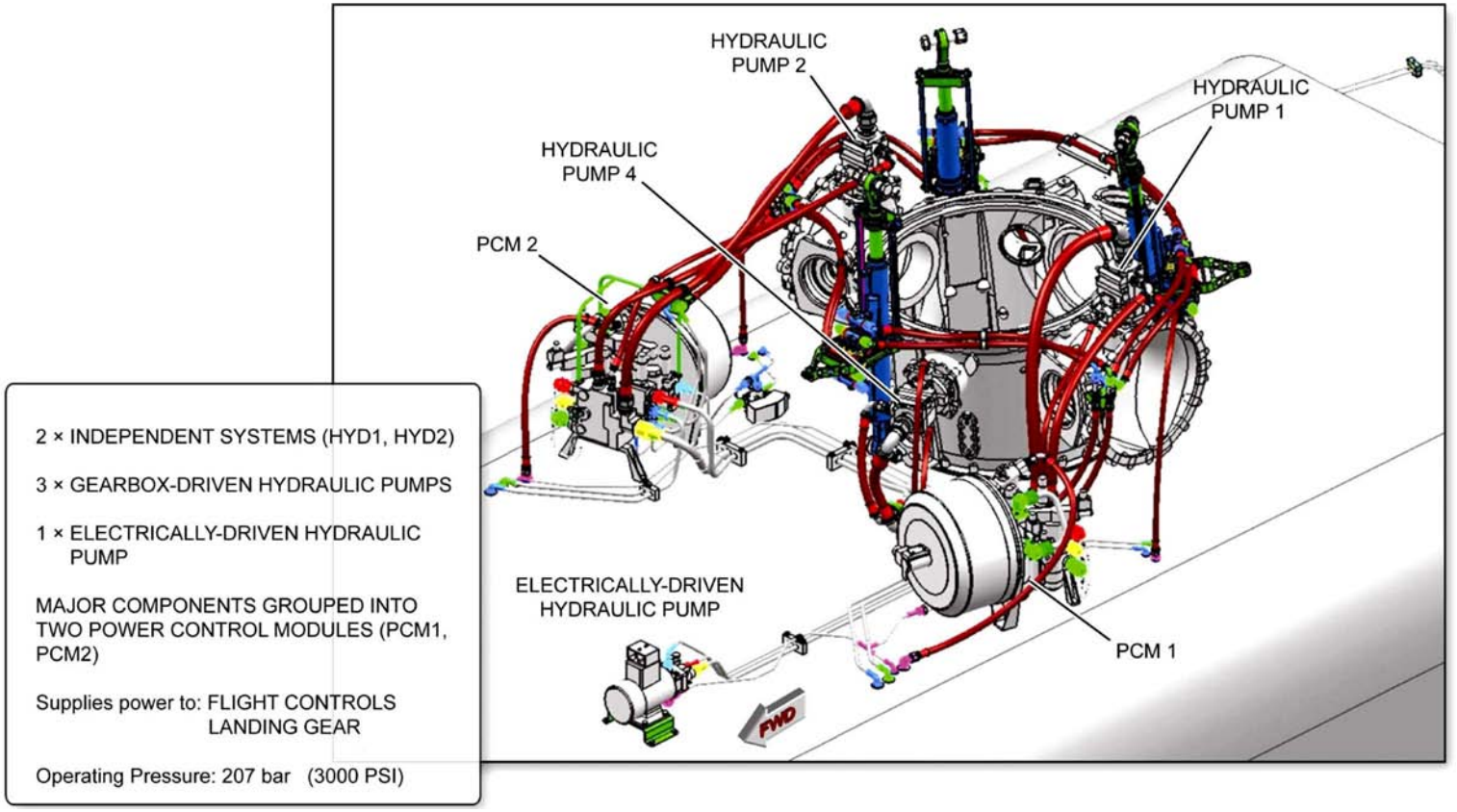
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ROTORS AND ROTOR DRIVE SYSTEMS



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FLIGHT CONTROLS



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HYDRAULIC POWER

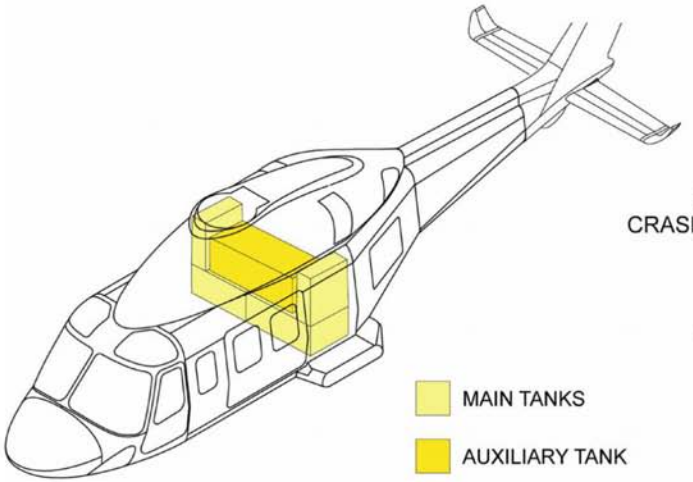


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LANDING GEAR

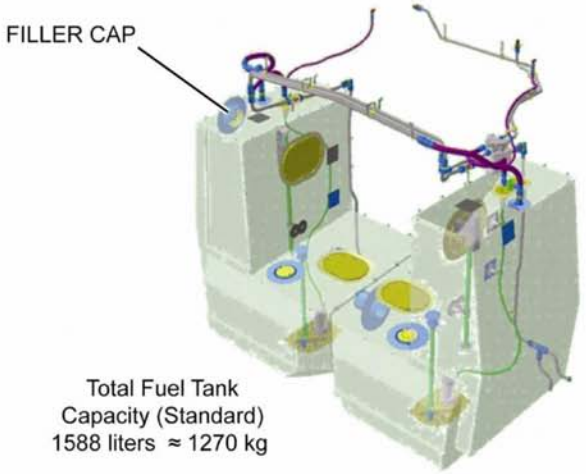


1 × 500 liters (≈ 400 kg)
OPTIONAL
AUXILIARY FUEL TANK



2 × INTERCONNECTED
CRASH-RESISTANT BLADDER-TYPE
FUEL TANKS

SINGLE-POINT
GRAVITY REFUELLING

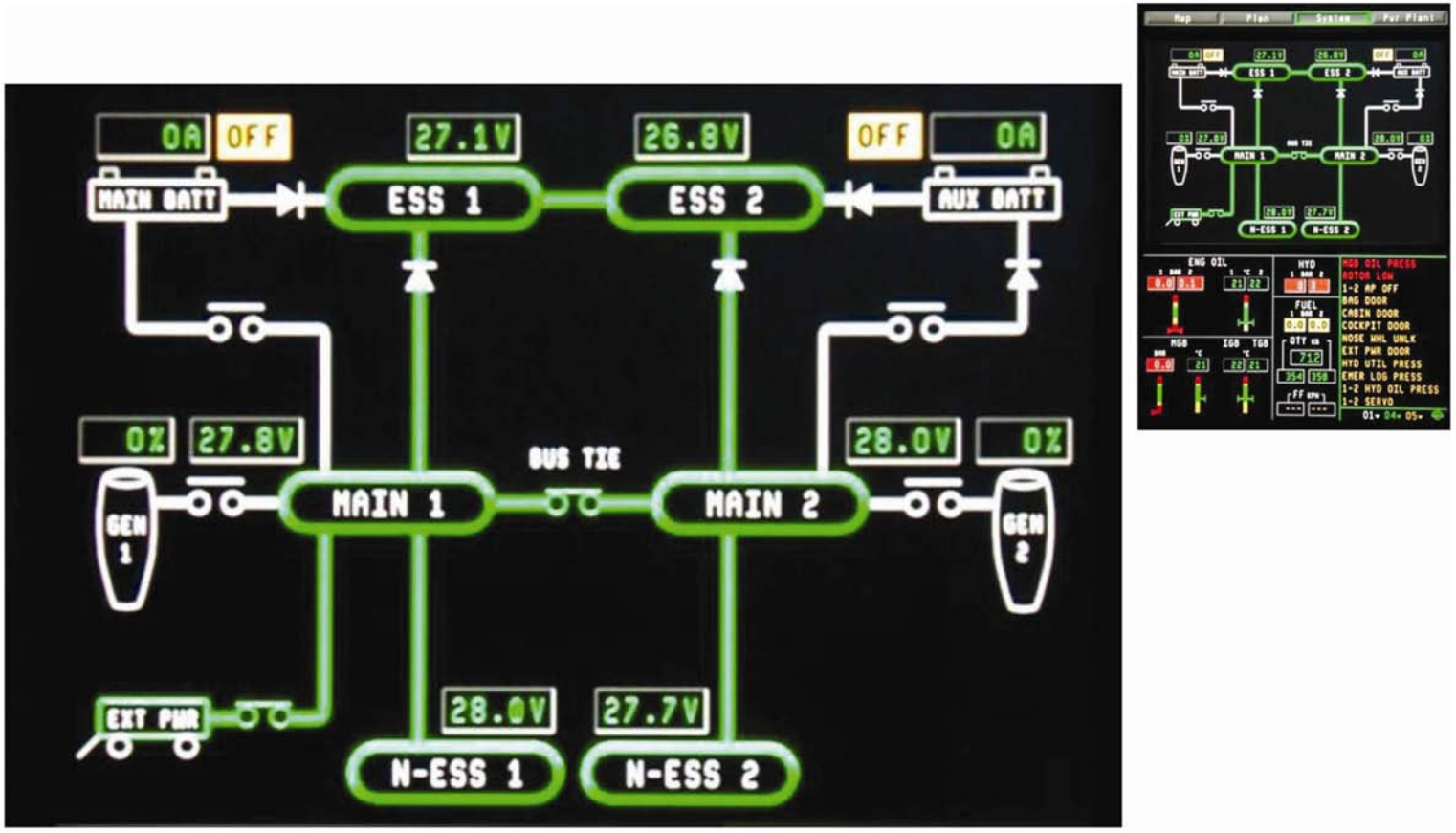


Total Fuel Tank
Capacity (Standard)
1588 liters ≈ 1270 kg



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FUEL



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ELECTRICAL POWER



INTERIOR LIGHTS

COCKPIT
INSTRUMENT, CONSOLE, OVERHEAD
2 x UTILITY
2 x ANTI STORM
1 x DOME

CABIN
6 x FLUORESCENT
12 x READING
2 x EMERGENCY DOME

BAGGAGE
3 x DOME

ANTI COLLISION
1 x TAIL TOP

NAVIGATION
2 x SIDE
1 x TAIL

LANDING
2 x LANDING (Fixed)
1 x SECONDARY LANDING (Searchlight)

EMERGENCY EGRESS
2 x SIDE (Fixed)



EXTERIOR LIGHTS

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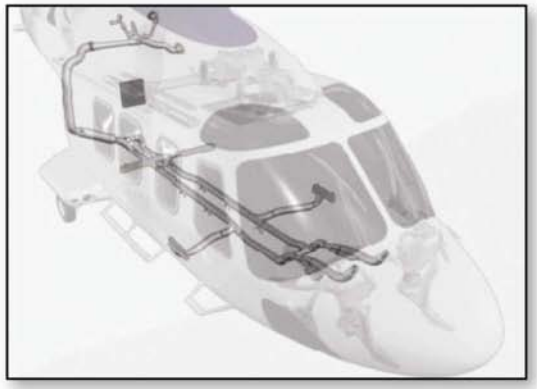
LIGHTING



VENTILATION

THREE SEPARATE SYSTEMS:
- PILOT VENTILATION
- COPILOT VE NTILATION
- CABIN VENTILATION

DYNAMIC OR FORCED AIR VENTILATION



HEATING

SINGLE SYSTEM SUPPLIED BY BOTH ENGINES

AUTOMATIC TEMPERATURE CONTROL



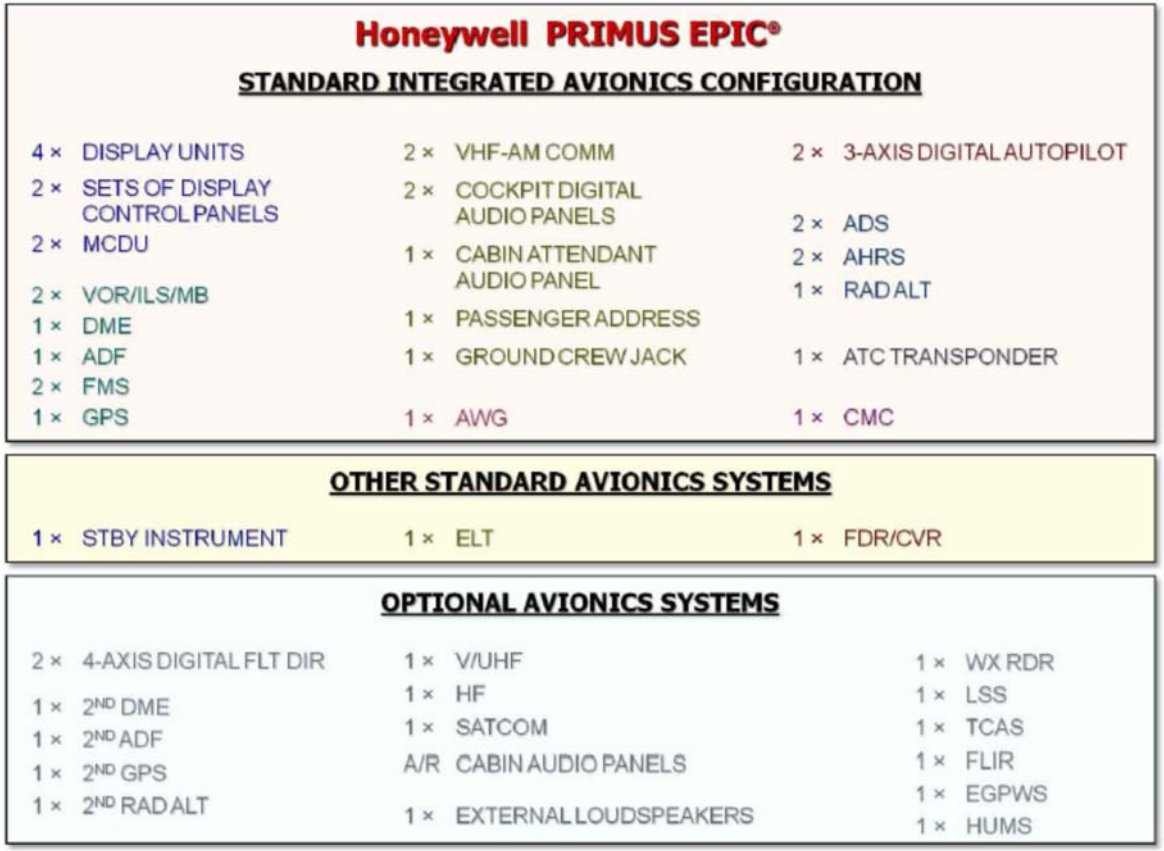
AIR CONDITIONING (Optional)

TWO SEPARATE VAPOUR-CYCLE SYSTEMS:
- COCKPIT AIR CONDITIONING
- CABIN AIR CONDITIONING

MAIN GEARBOX-DRIVEN COMPRESSORS

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ENVIRONMENTAL CONTROL SYSTEM



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AVIONIC SYSTEM CONFIGURATION

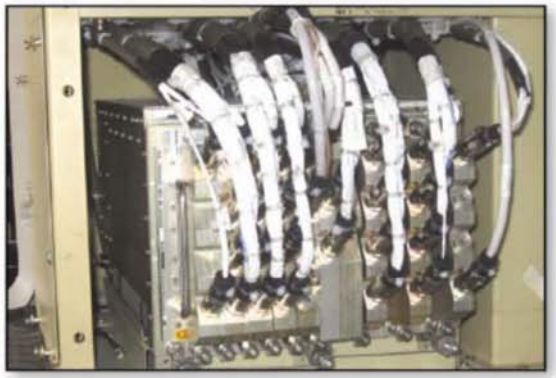
Honeywell PRIMUS EPIC®
INTEGRATED AVIONICS

Integrates systems and subsystems for:

- NAVIGATION
- COMMUNICATIONS
- AUTOFLIGHT
- INDICATING/RECORDING
- CENTRAL MAINTENANCE



DU (Display Unit)



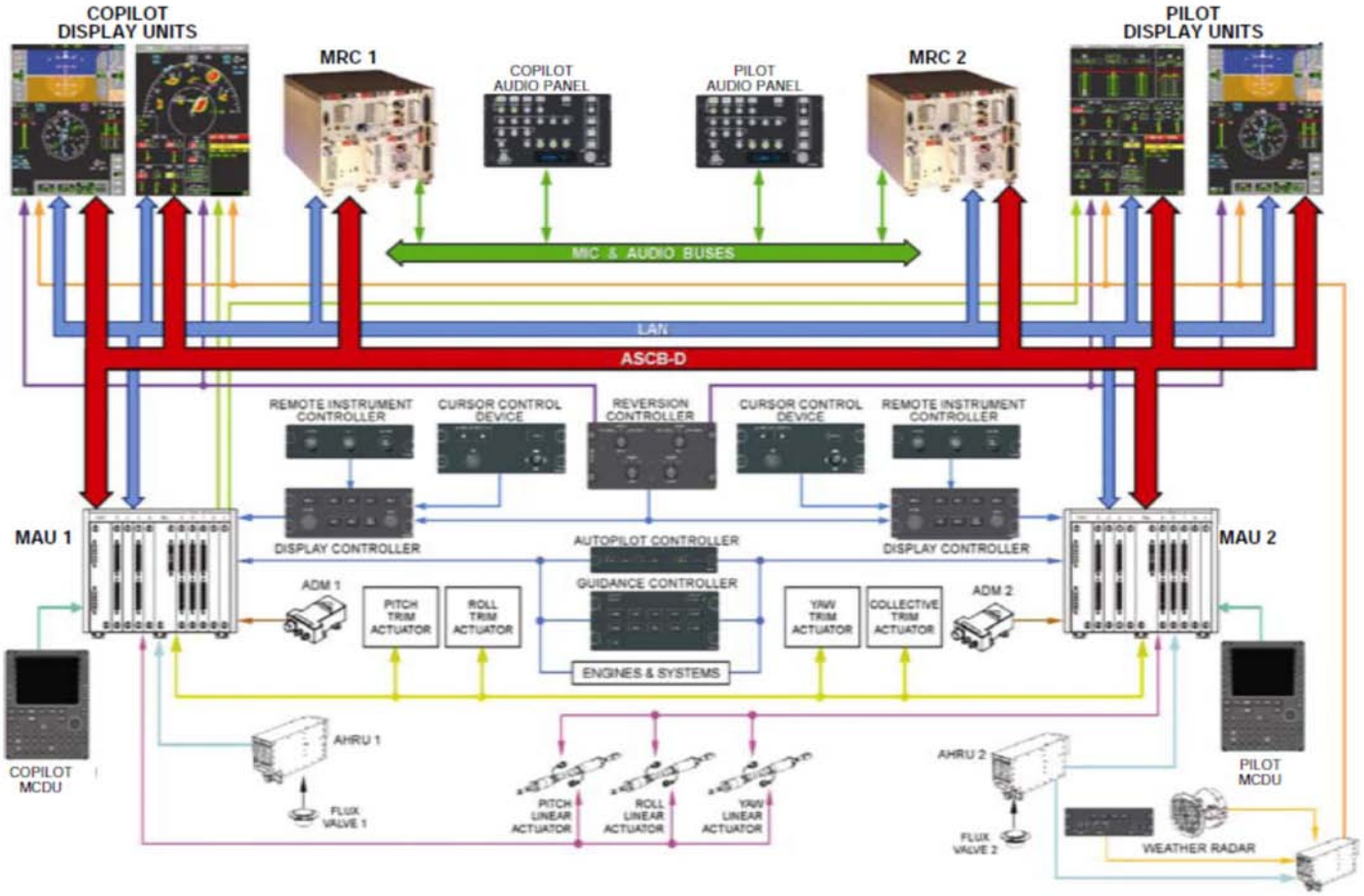
MAU (Modular Avionic Unit)



MRC (Modular Radio Cabinet)

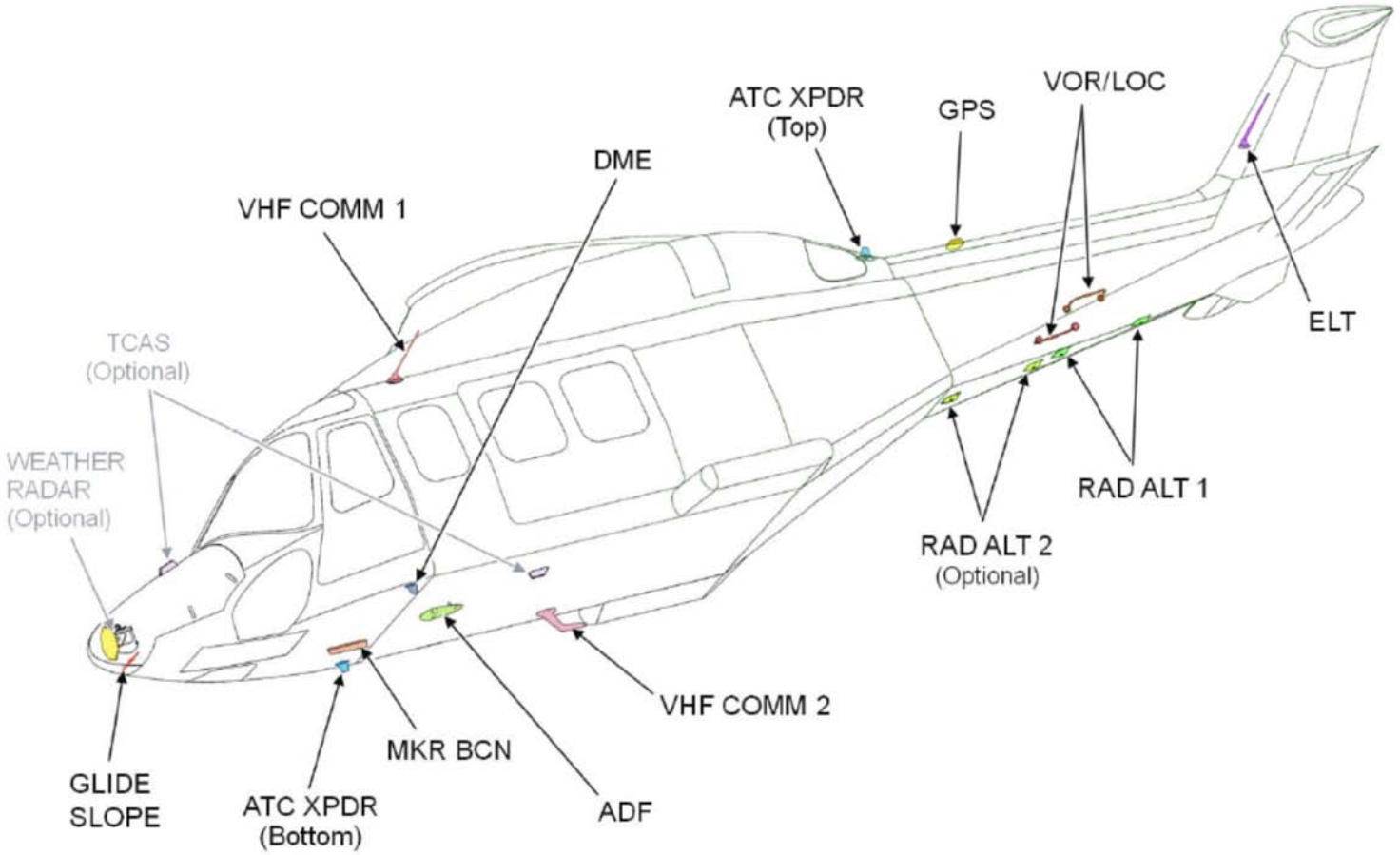
ICN-39-A-000000-G-A0126-01625-A-01-1

AVIONICS SYSTEM - HIGHLIGHTS



ICN-39-A-000000-G-A0126-01626-A-01-1

PRIMUS EPIC ® GENERAL ARRANGEMENT



ICN-39-A-000000-G-A0126-01627-A-01-1

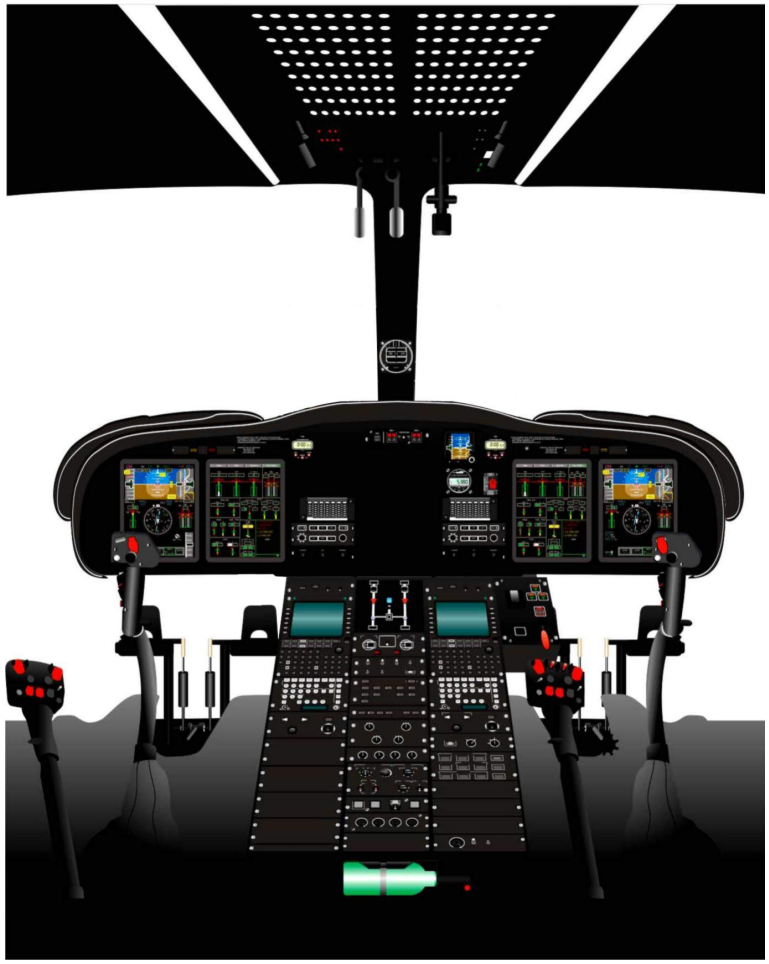
ANTENNA LAYOUT (TYPICAL)

NOISE CHARACTERISTICS

The following noise levels comply with ICAO Annex 16, Chapter 8 Amdt 7, “NOISE requirement”.

Model: AW139 Engine Pratt and Whitney PT6C-67C			Gross Weight 6400 Kg		
Configuration	Level Flyover ENPL (EPNdB)	Take Off EPNL (EPNdB)		Approach EPNL (EPNdB)	
Clean aircraft No external kits installed	100% NR	100% NR	102% NR	100% NR	102% NR
	89.8	90.5	91.0	93.0	93.5

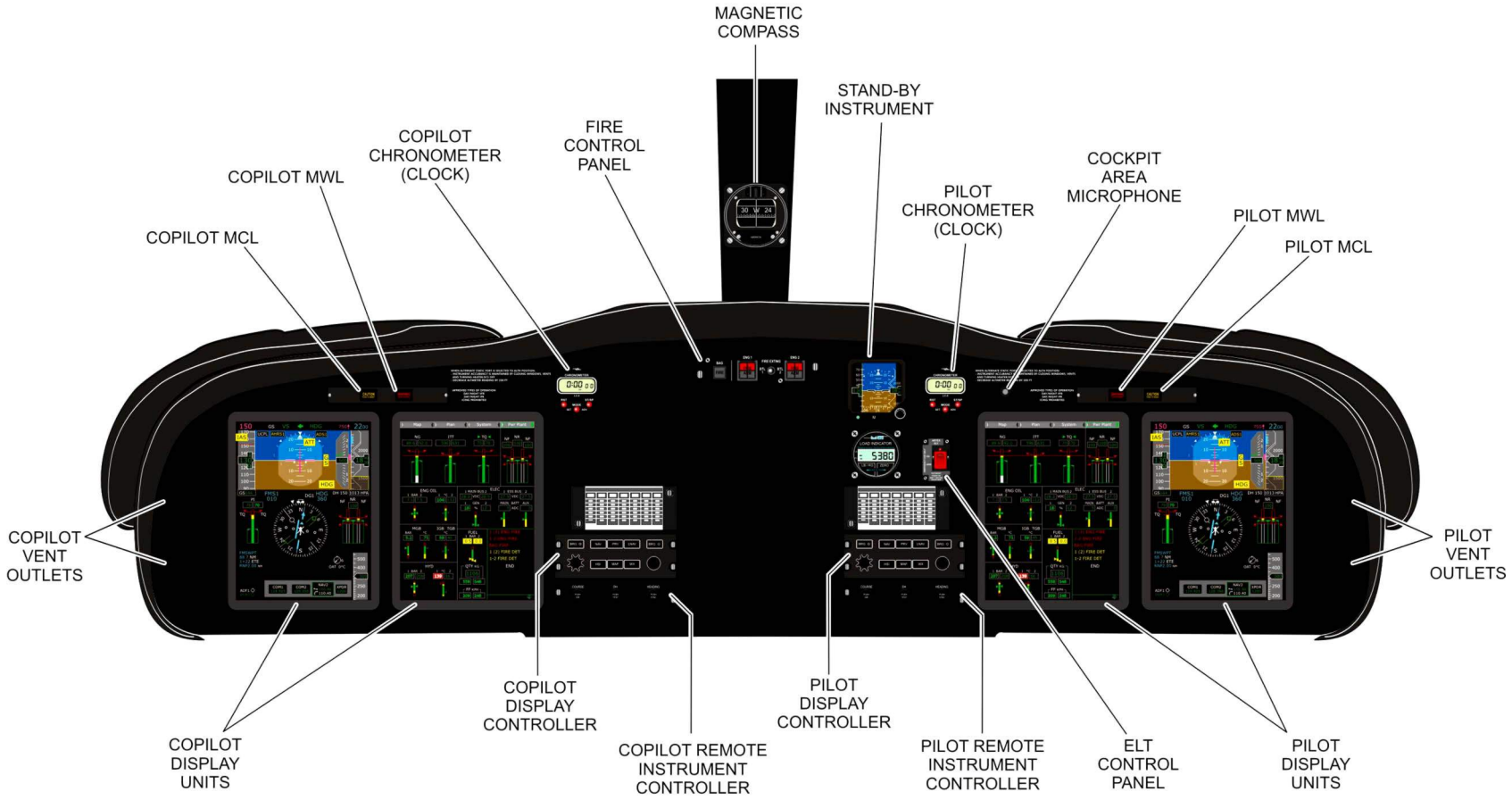
SECTION 03 - COCKPIT LAYOUT



ICN-39-A-000300-G-A0126-00430-A-01-1

AW139 HELICOPTER - COCKPIT (TYPICAL)

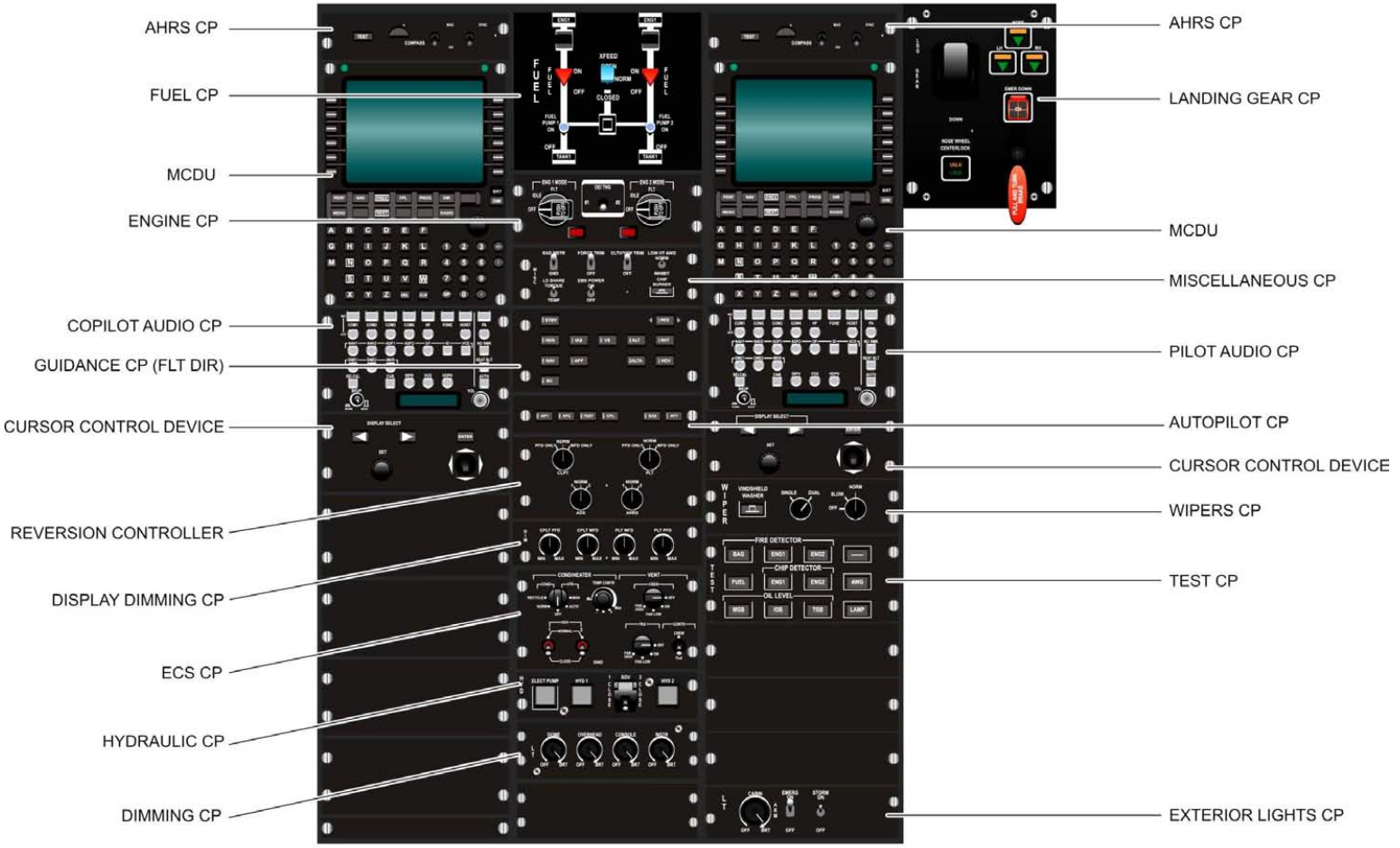
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ICN-39-A-000300-G-A0126-00431-A-01-1

AW139 HELICOPTER - INSTRUMENT PANEL (TYPICAL)

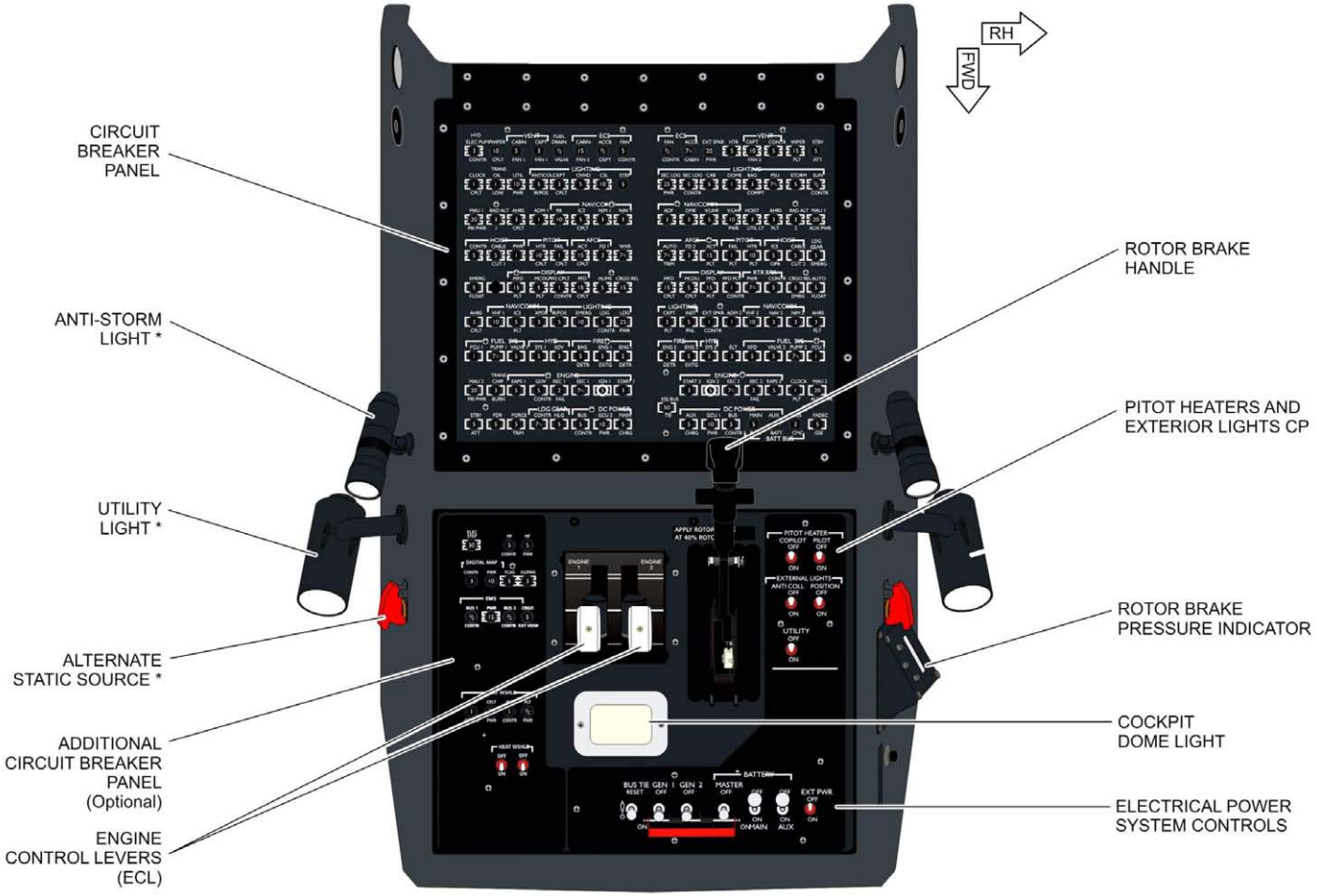
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ICN-39-A-000300-G-A0126-00432-A-01-1

AW139 HELICOPTER - CENTRAL CONSOLE (TYPICAL)

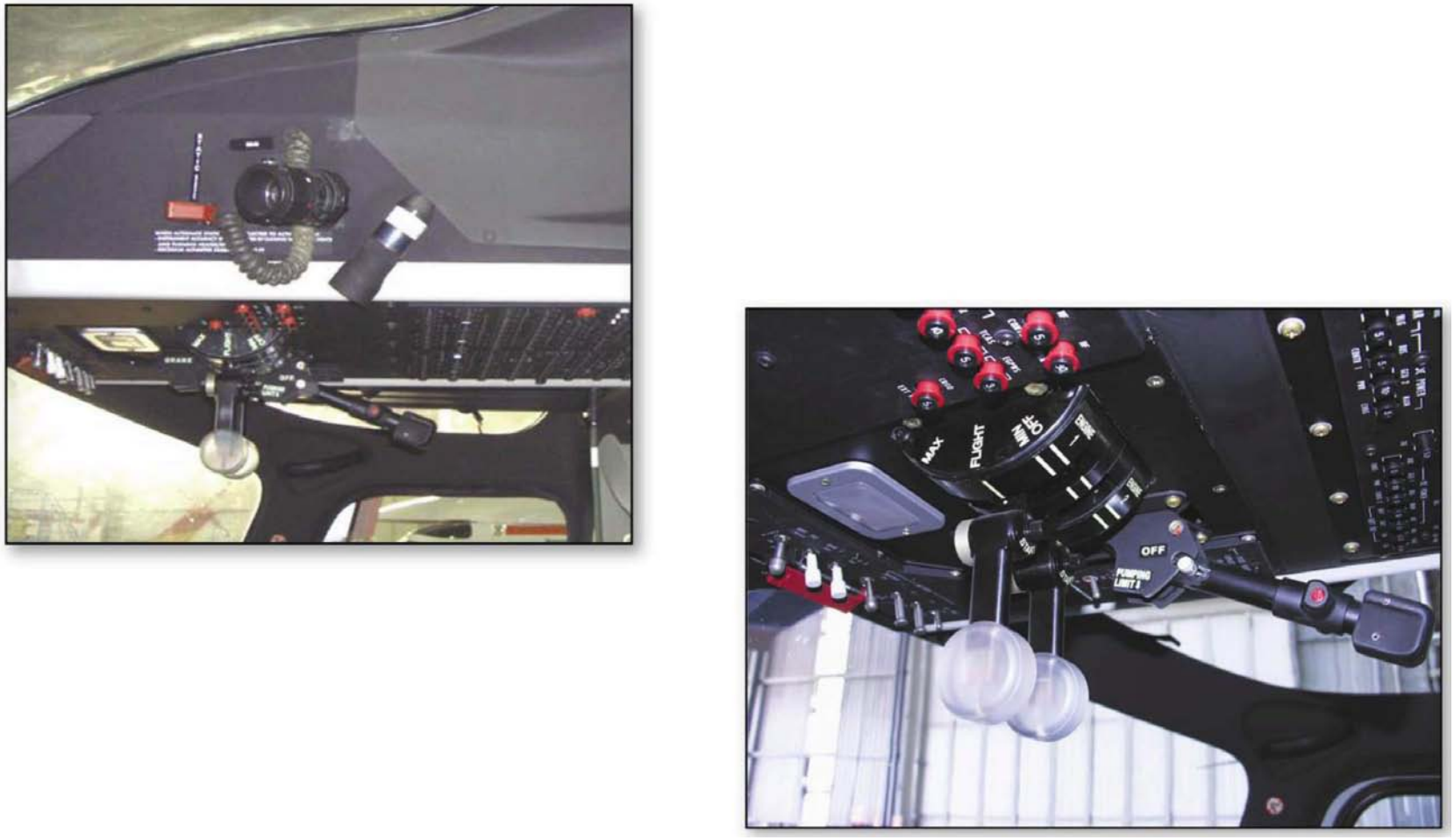
(Printed A3 format at the end of data module)



ICN-39-A-000300-G-A0126-00433-A-01-1

AW139 HELICOPTER - OVERHEAD CONSOLE (TYPICAL)

(Printed A3 format at the end of data module)



ICN-39-A-000300-G-A0126-01637-A-01-1

OVERHEAD CONSOLE – COPILOT SIDE WALL – AND ENGINE CONTROL LEVERS (ECL)



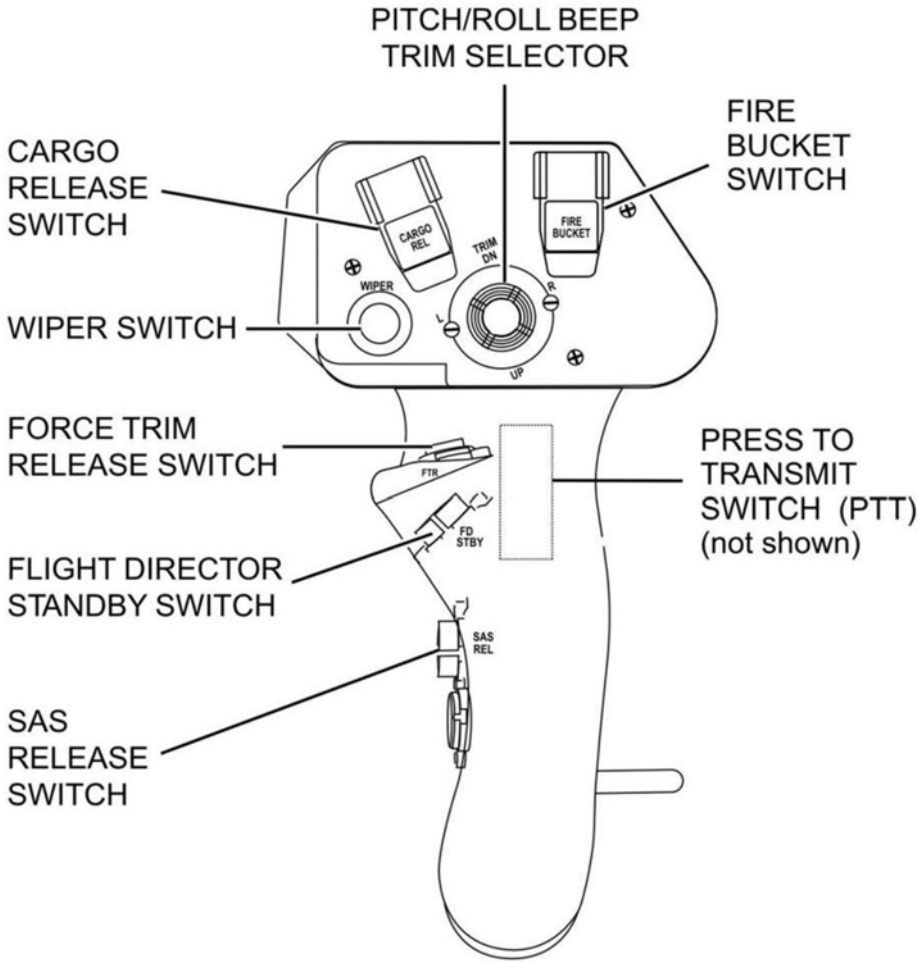
ICN-39-A-000300-G-A0126-01635-A-01-1

OVERHEAD CONSOLE – PILOT SIDE WALL



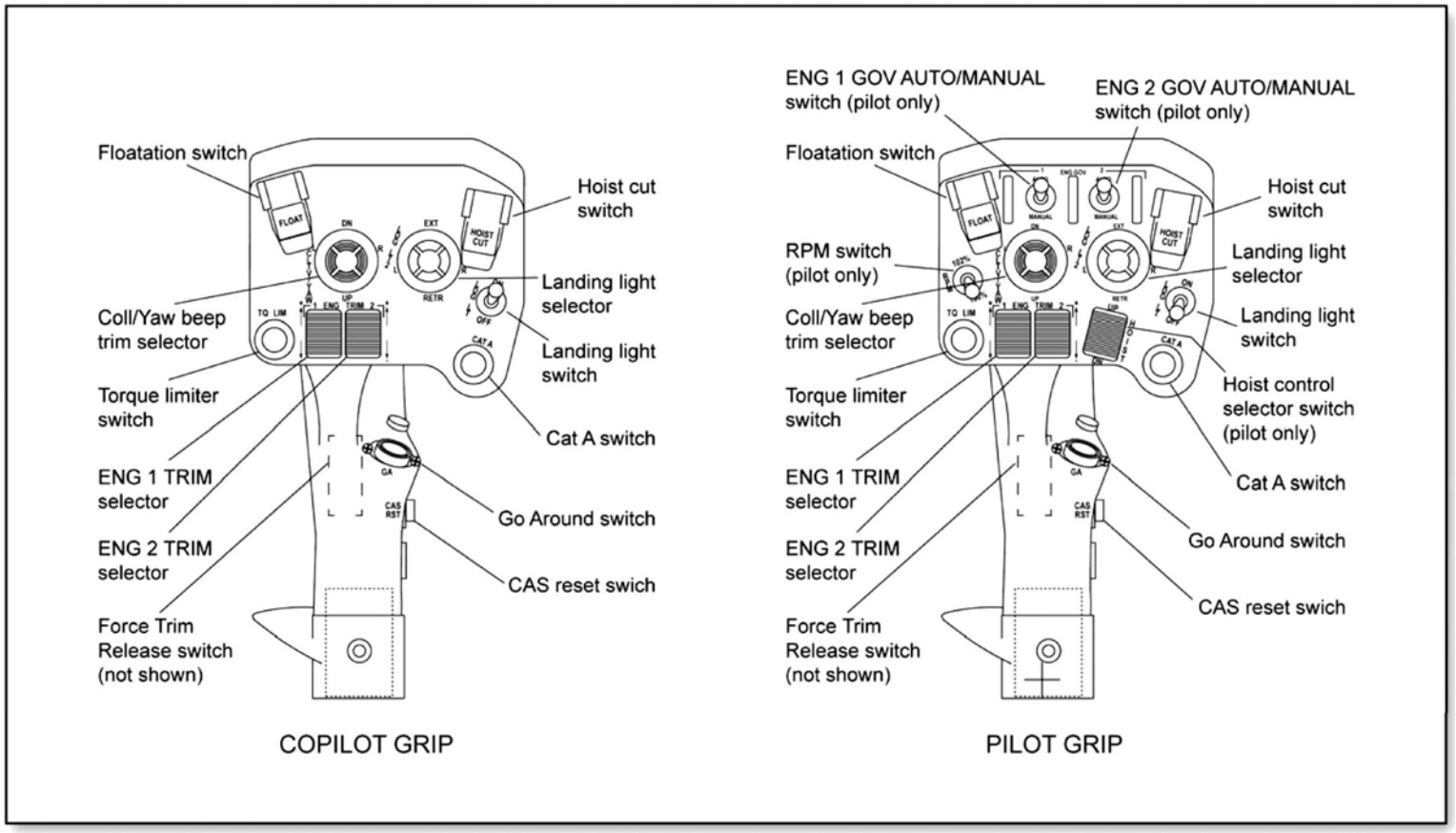
ICN-39-A-000300-G-A0126-01636-A-01-1

STANDBY OUTSIDE AIR TEMPERATURE (OAT) INDICATOR



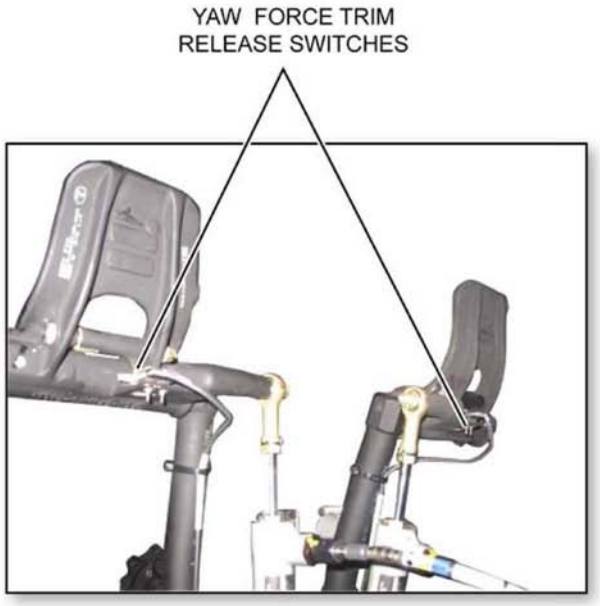
ICN-39-A-000300-G-A0126-03231-A-01-1

PILOT / COPILOT CYCLIC STICK GRIP



ICN-39-A-000300-G-A0126-01640-A-01-1

PILOT / COPILOT COLLECTIVE LEVER GRIP



NOTE: Copilot station arranged symmetrically



ICN-39-A-000300-G-A0126-01641-A-01-1

PEDALS AND FOOT SWITCH

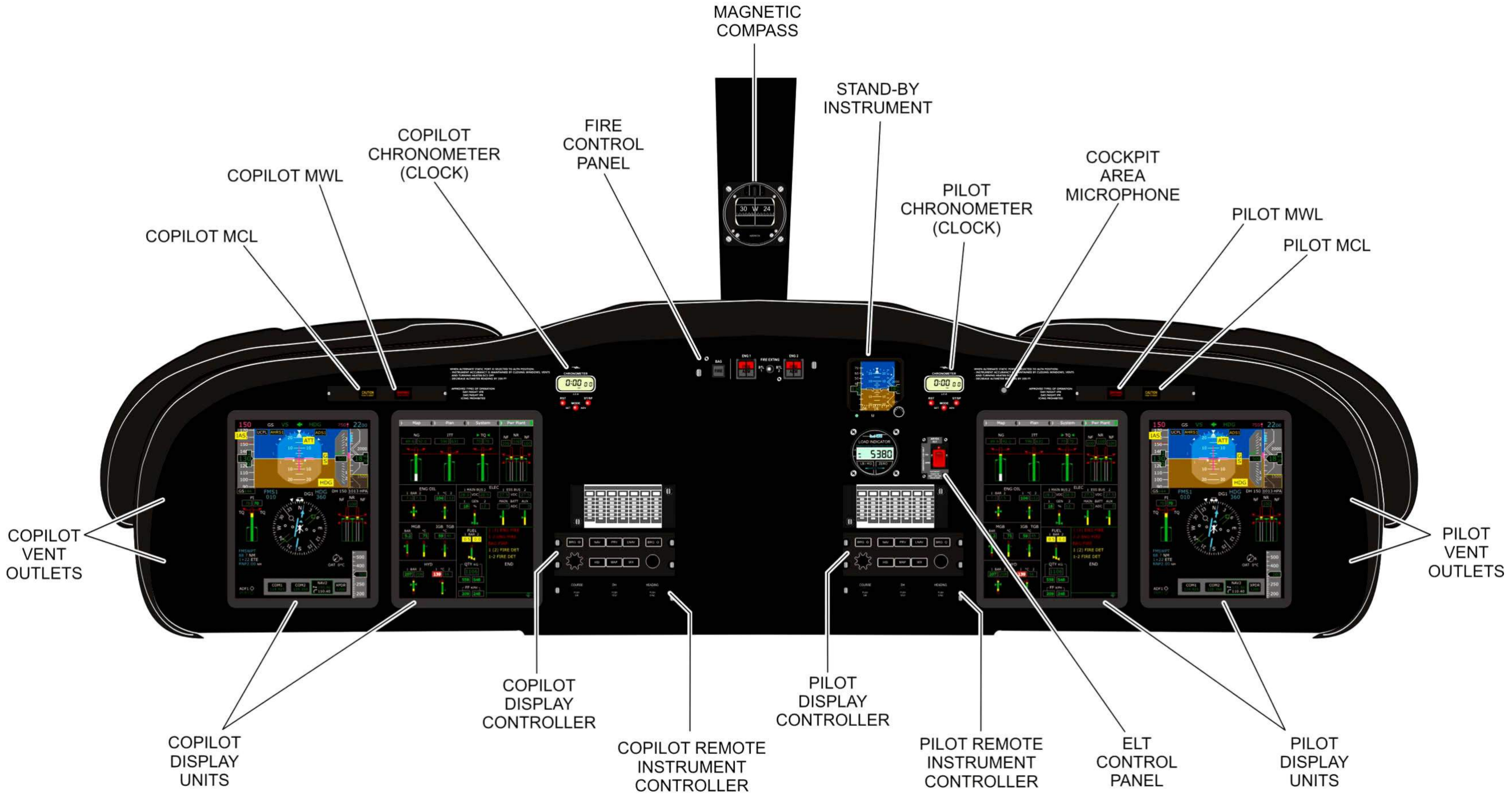
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ICN-39-A-000300-G-A0126-00430-A-01-1

AW139 HELICOPTER - COCKPIT (TYPICAL)

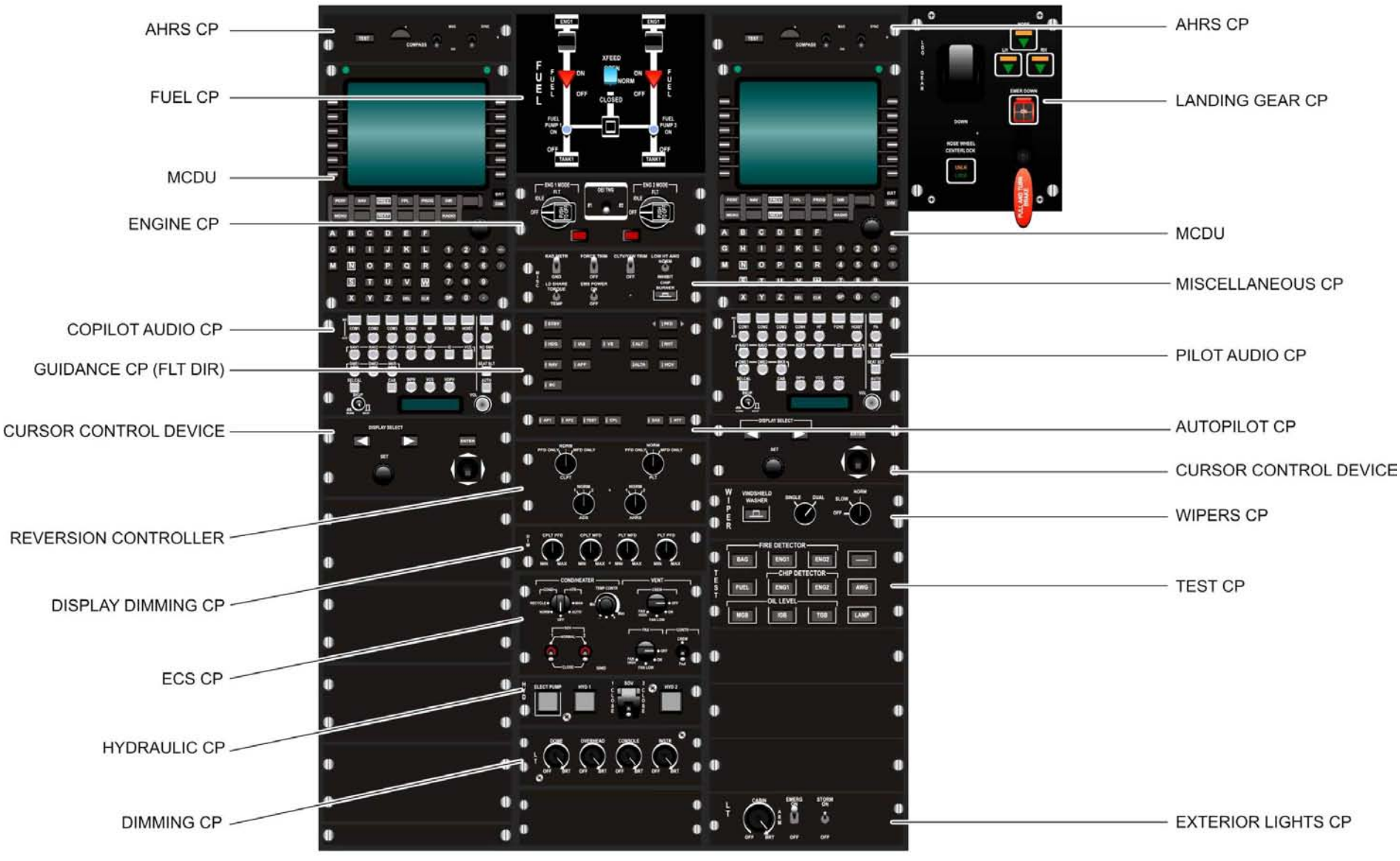
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ICN-39-A-000300-G-A0126-00431-A-01-1

AW139 HELICOPTER - INSTRUMENT PANEL (TYPICAL)

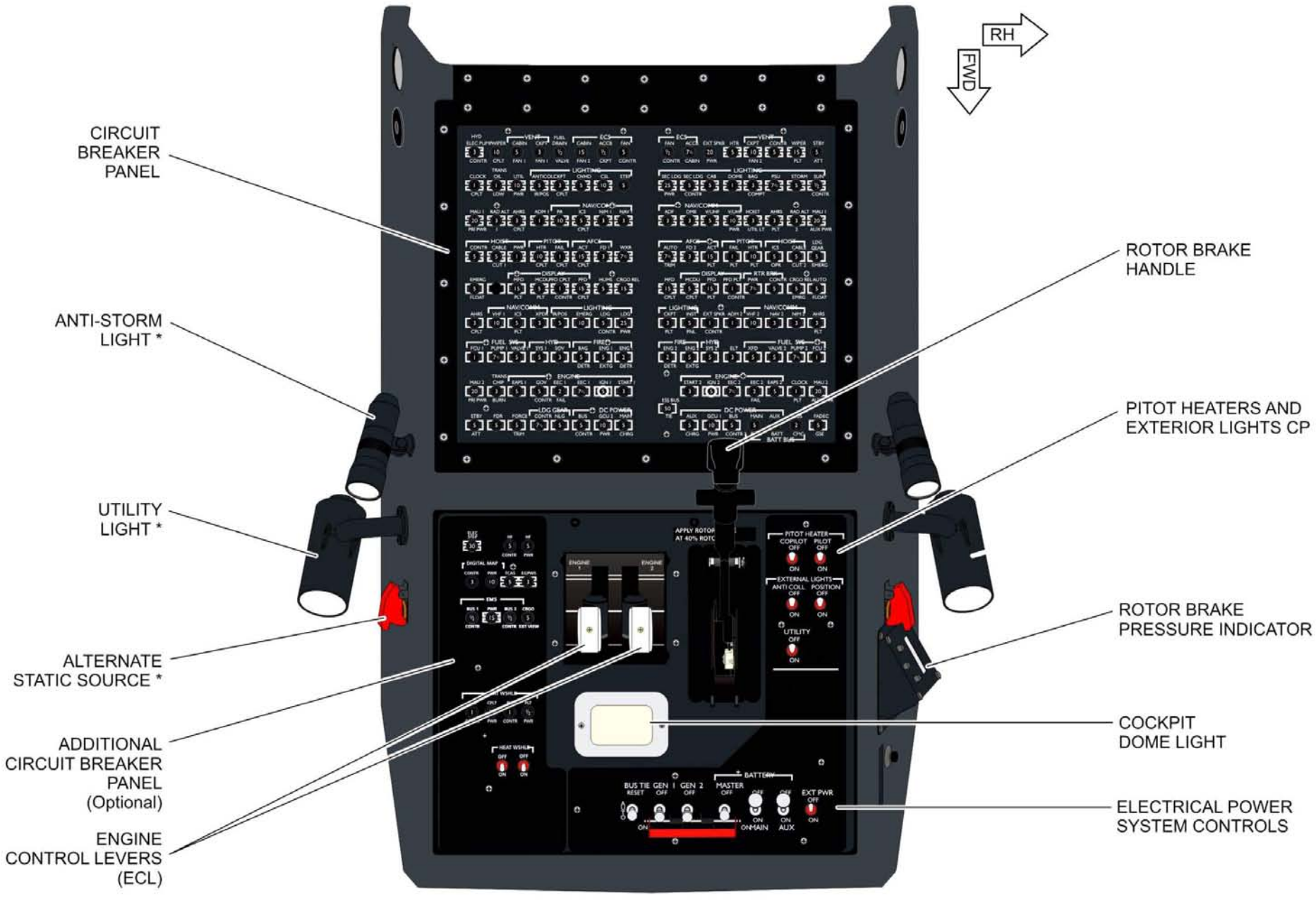
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ICN-39-A-000300-G-A0126-00432-A-01-1

AW139 HELICOPTER - CENTRAL CONSOLE (TYPICAL)

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ICN-39-A-000300-G-A0126-00433-A-01-1

AW139 HELICOPTER - OVERHEAD CONSOLE (TYPICAL)

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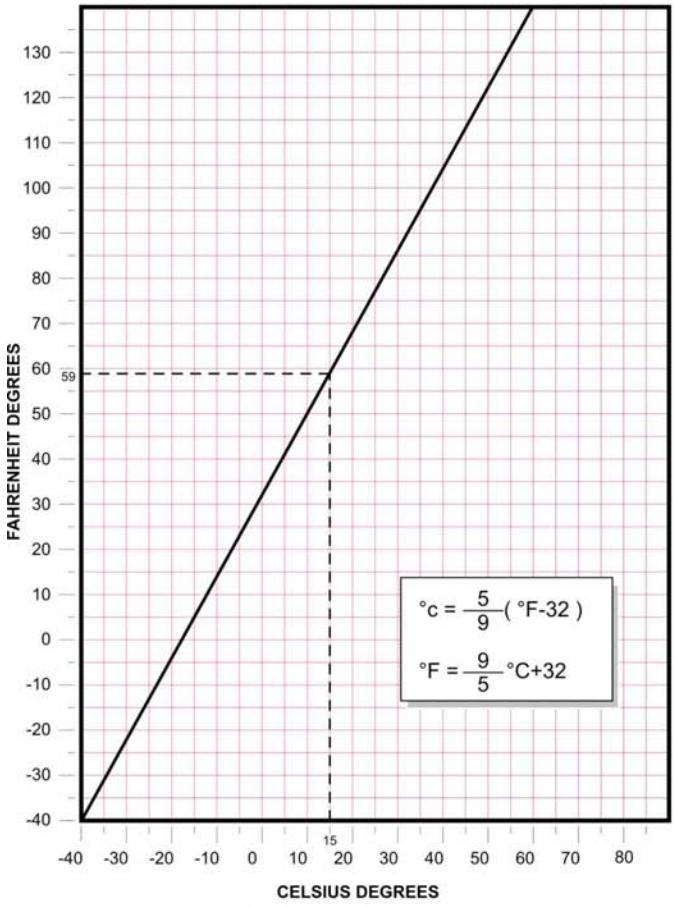
SECTION 12 - CONVERSION TABLES

GENERAL

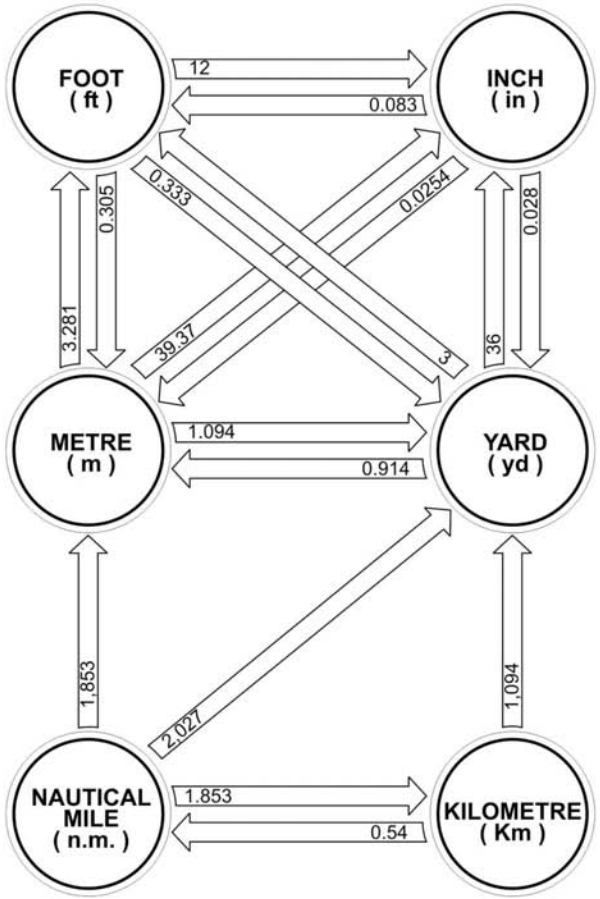
The following conversion tables are included:

- Temperature
- Length
- Pressure
- Torque
- Volume (US units)
- Volume (Imperial units)
- Power
- Weight / Force
- Jet B / JP4 Fuel
- Jet A1 / JP5 / JP8 Fuel

To obtain the unit conversion always multiply by the number following in the arrow.

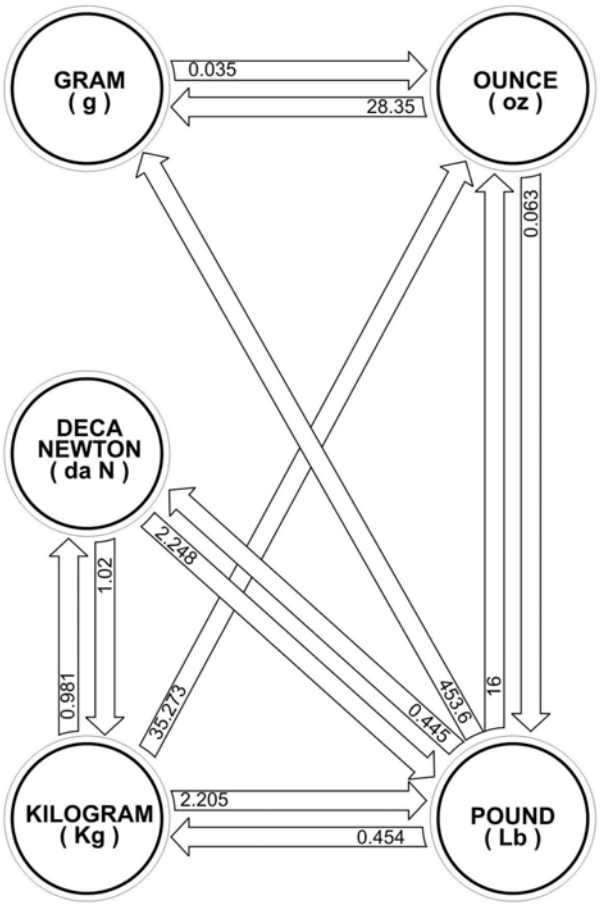


TEMPERATURE

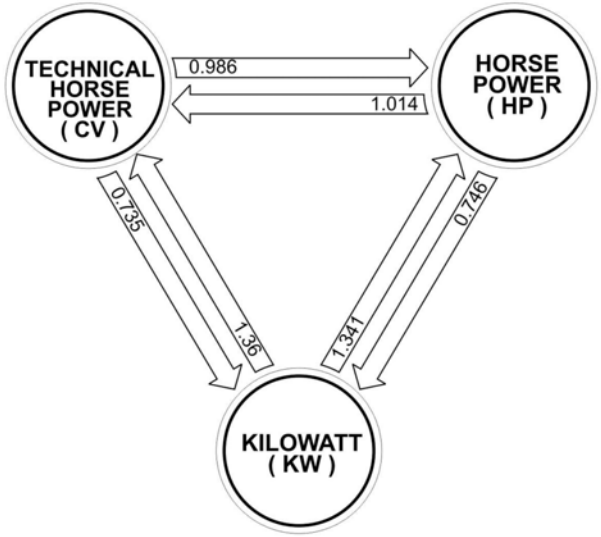


LENGTH

ICN-39-A-001200-G-A0126-00462-A-01-1

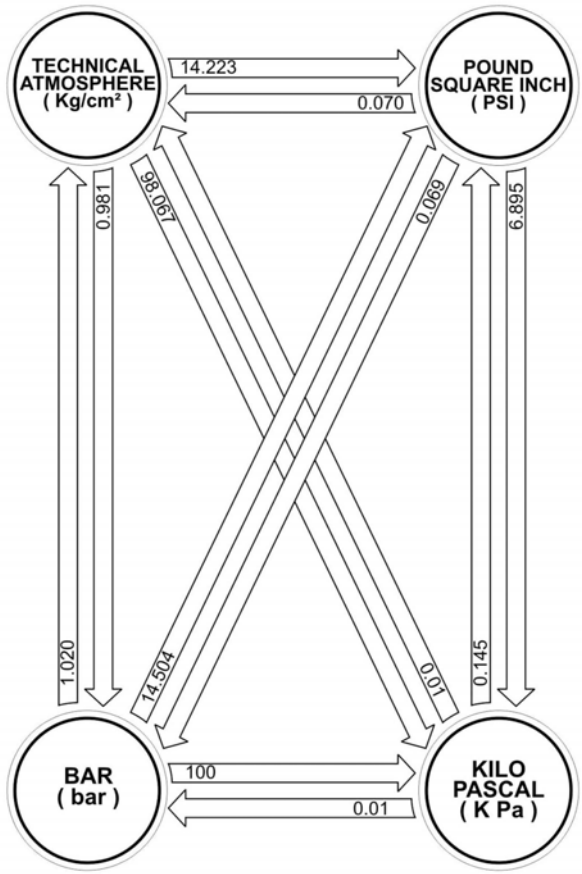


WEIGHT/FORCE

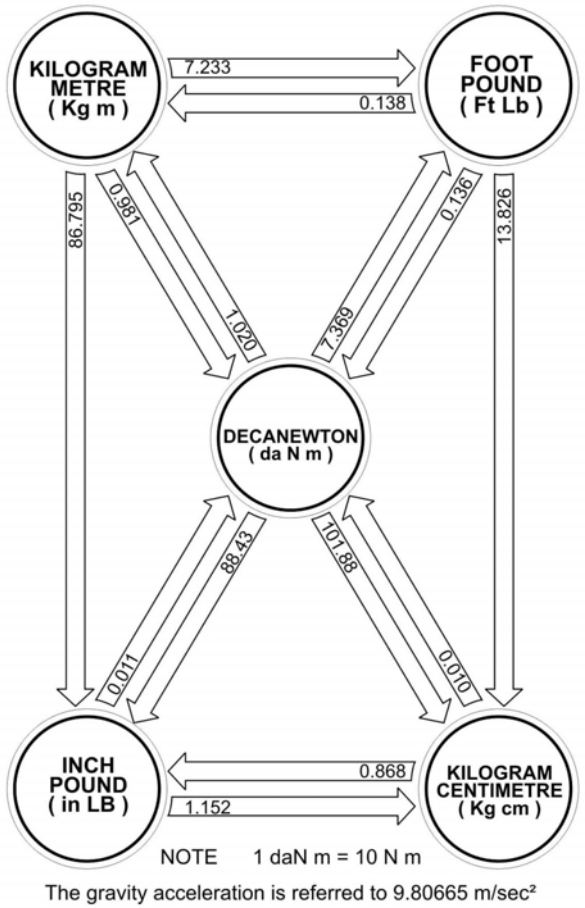


POWER

ICN-39-A-001200-G-A0126-00463-A-01-1



PRESSURE

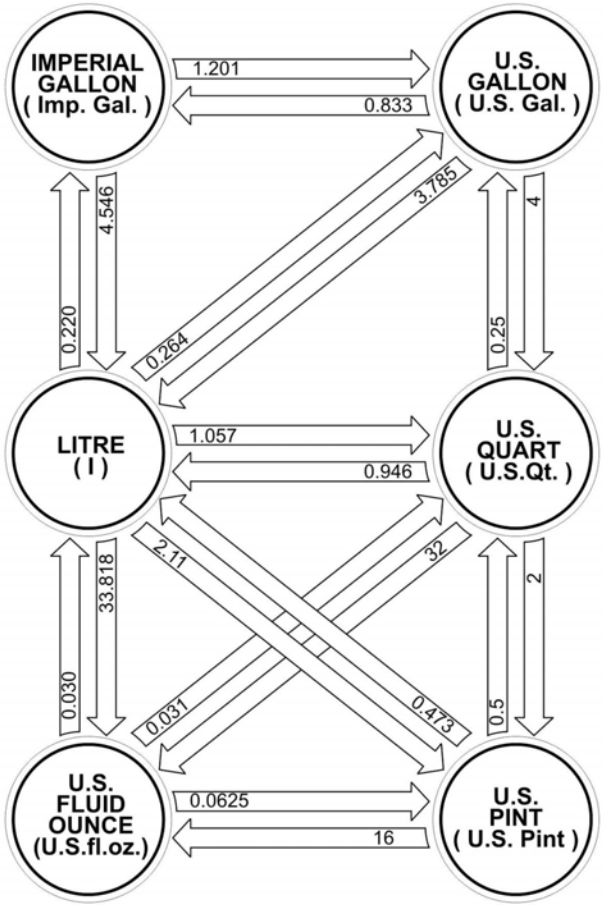


NOTE 1 daN m = 10 N m

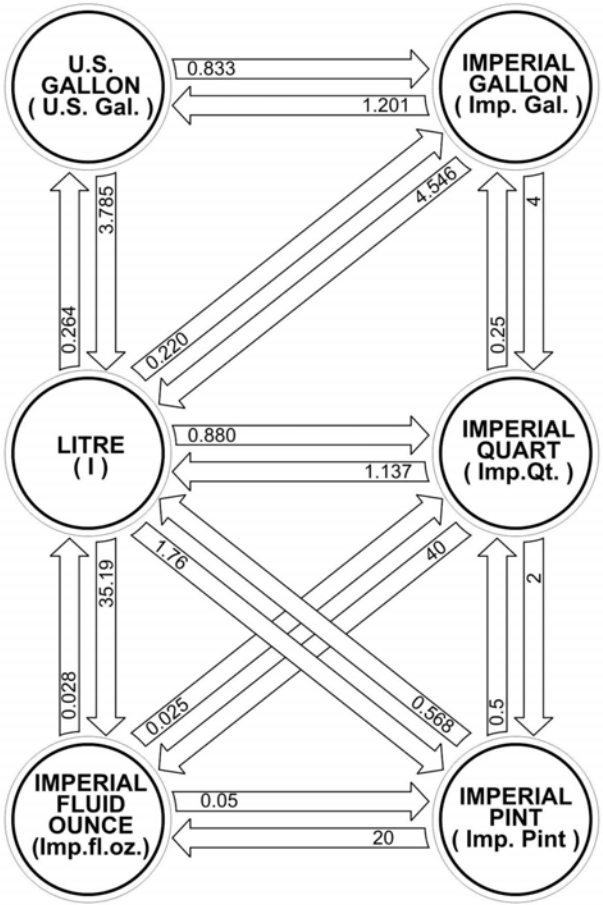
The gravity acceleration is referred to 9.80665 m/sec²

TORQUE

ICN-39-A-001200-G-A0126-00464-A-01-1

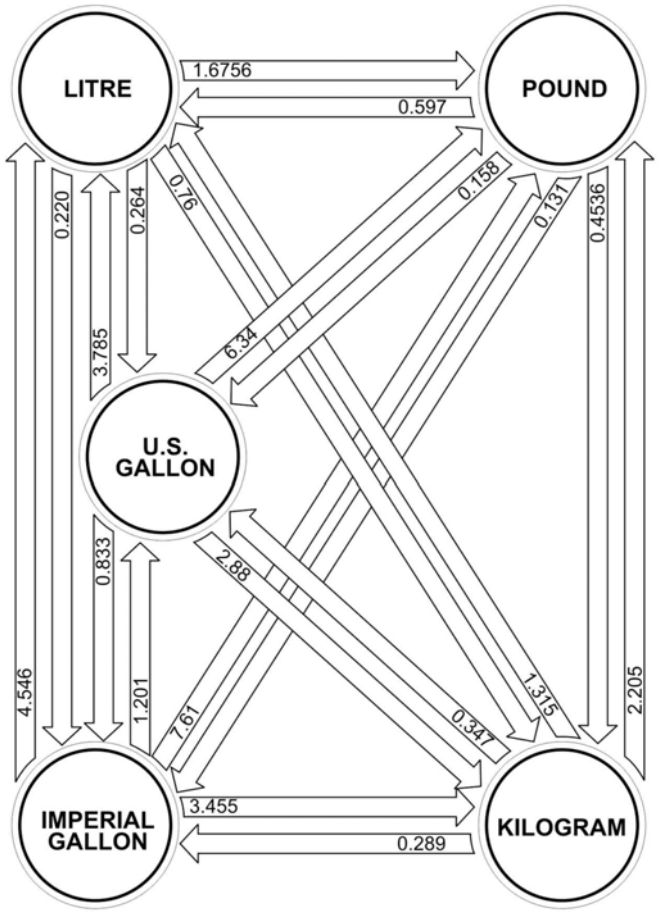


VOLUME (U.S.)

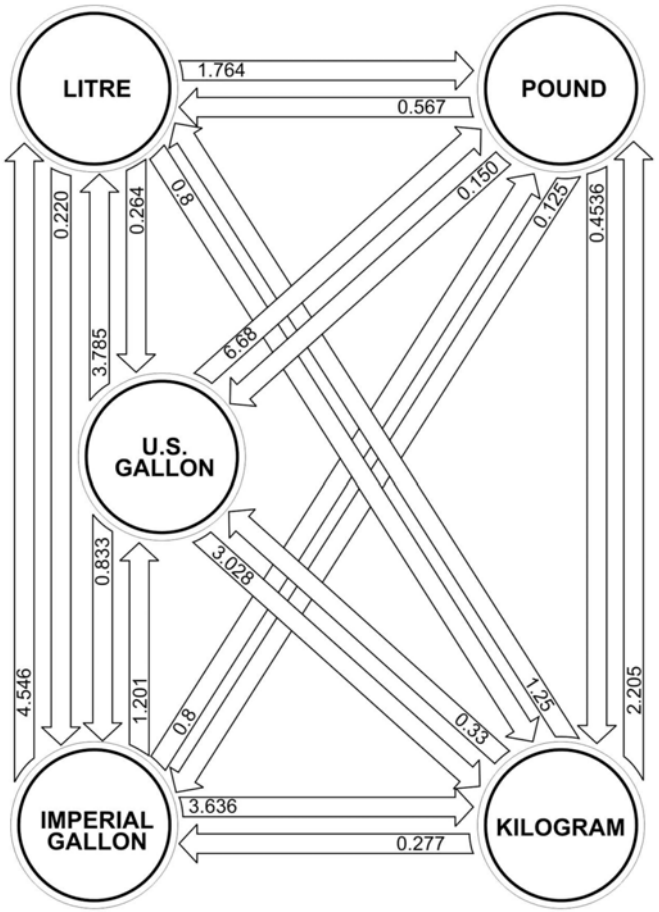


VOLUME (IMP.)

ICN-39-A-001200-G-A0126-00465-A-01-1

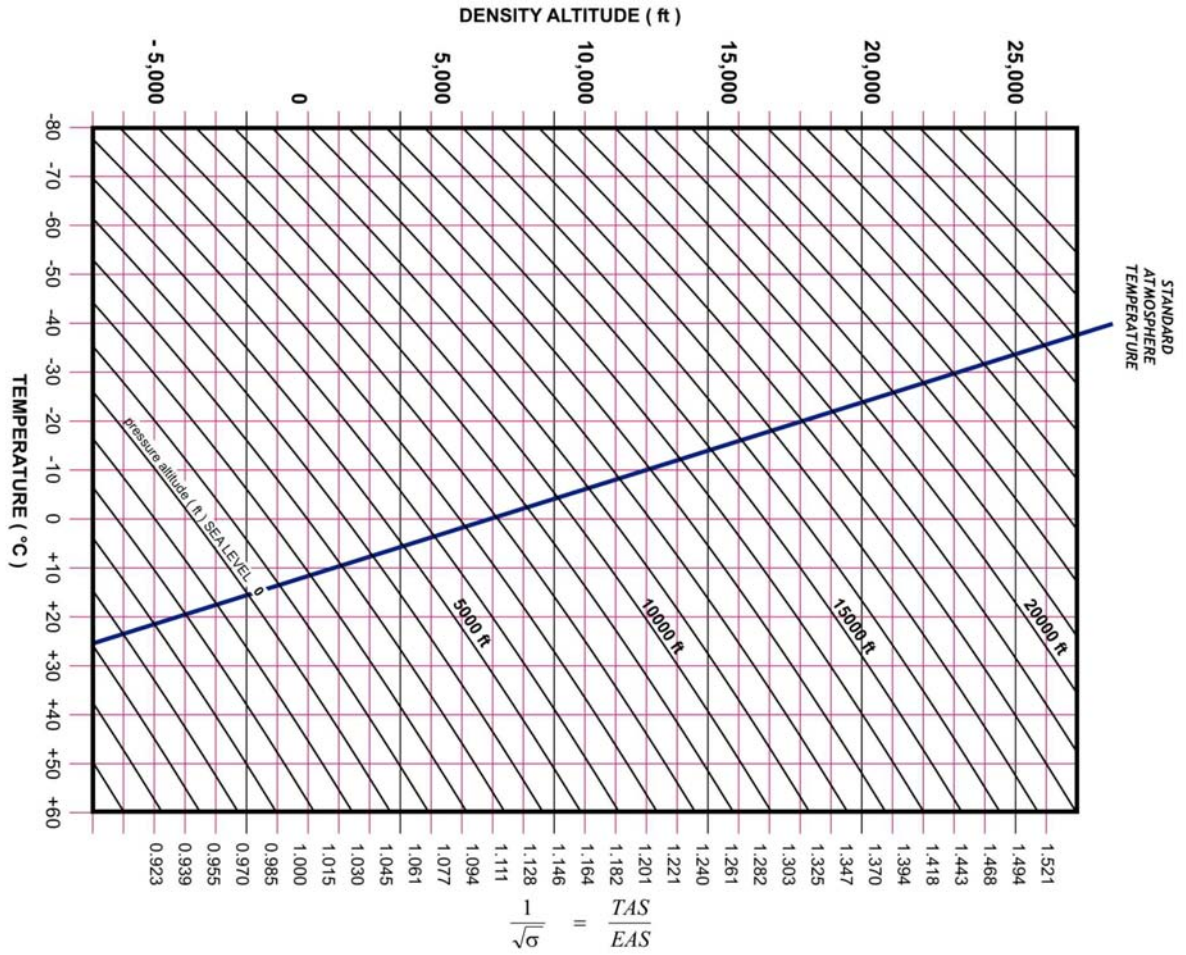


JET " B " AND JP 4 FUEL



JET " A1 ", JP5 AND JP8 FUEL

ICN-39-A-001200-G-A0126-00466-A-01-1



ICN-39-A-001200-G-A0126-00467-A-01-1

DENSITY/ALTITUDE CONVERSION TABLE

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SECTION 20 - HELICOPTER SAFETY FOR MAINTENANCE

MAINTENANCE INFORMATION

NOTE A

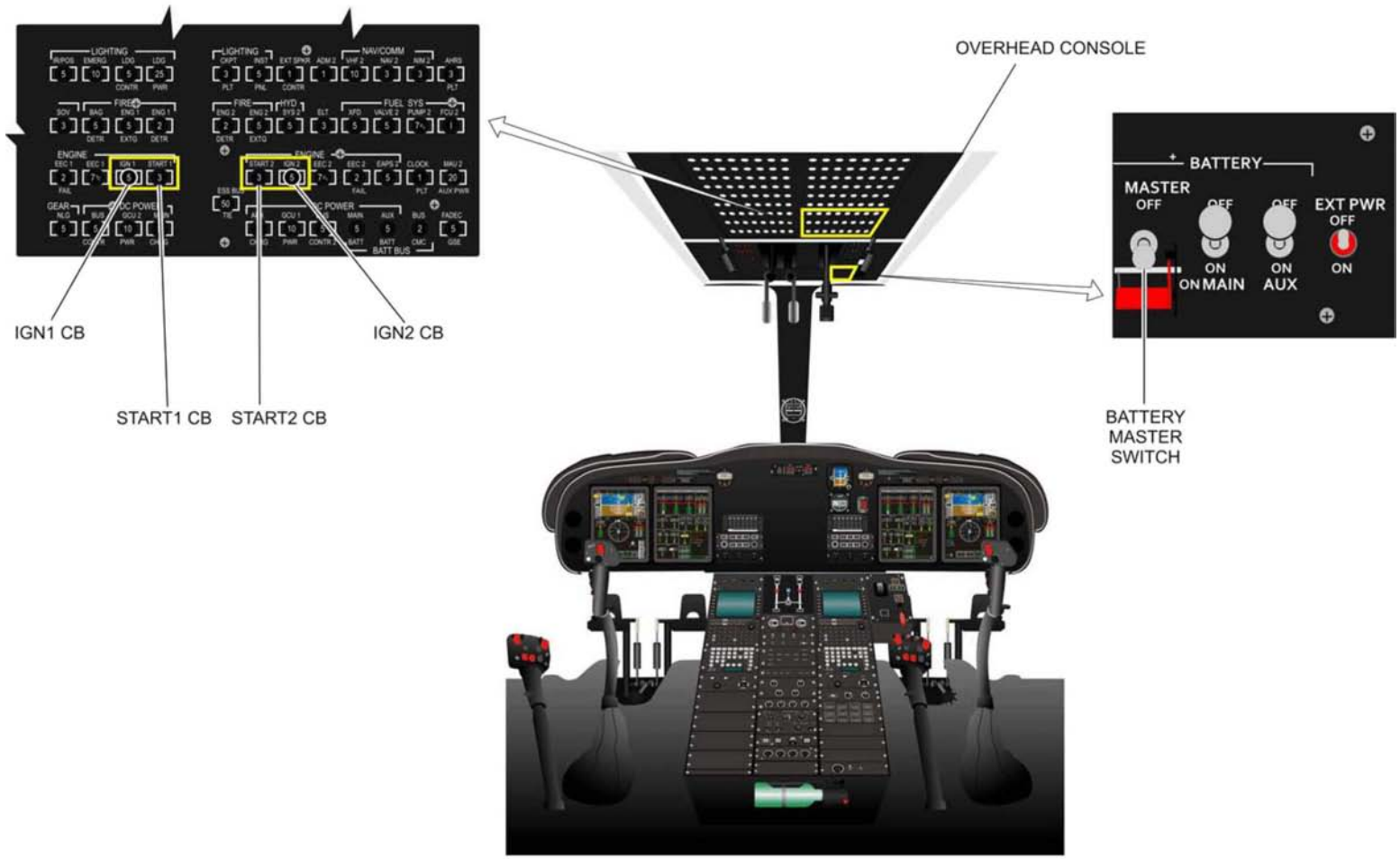
Refer to the following data modules:
DMC 39-A-00-20-00-00A-120A-A
Helicopter safety – Make the helicopter safe for maintenance
DMC 39-A-10-11-00-00A-000A-A
Helicopter – Parking in normal conditions – General

In order to prepare the helicopter for a safe maintenance, the following steps must be achieved:

- Connect an earth point to the helicopter
- Make sure the BATTERY MASTER switch is set to OFF
- Open the following circuit breakers under ENGINE system on the overhead console:
  - IGN 1
  - IGN 2
  - START 1
  - START 2
- Disconnect the quick-disconnect plug from the main and auxiliary batteries.

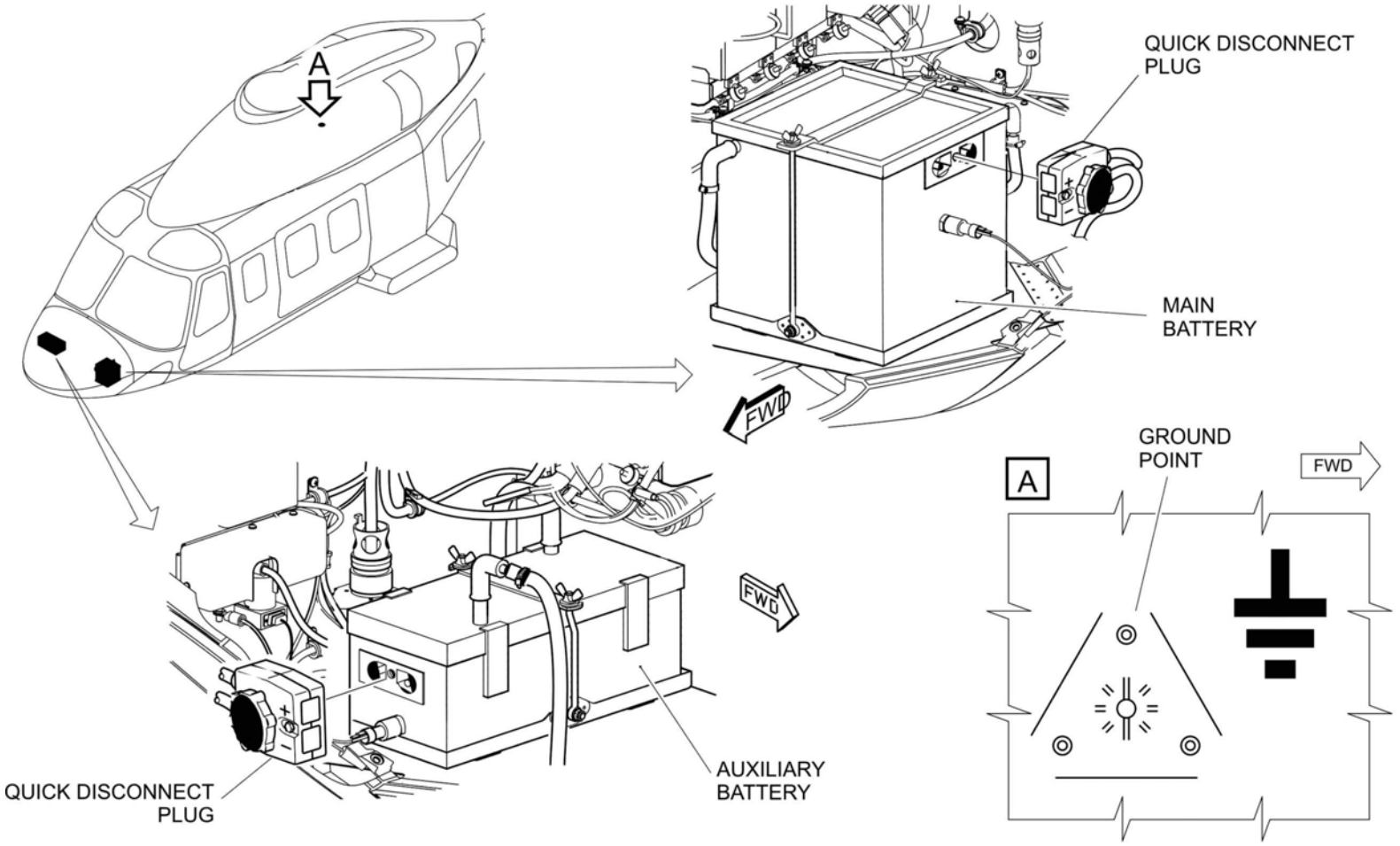
SAFETY PRECAUTIONS

- You must connect the bonding cable to the earth (ground) point before you connect it to the helicopter. If you do not obey this instruction you can cause an injury to persons and/or damage to the equipment.



ICN-39-A-002000-G-A0126-02879-A-01-1

INSTRUMENT AND INTERSEAT



ICN-39-A-002000-G-A0126-00469-A-01-1

BATTERIES DISCONNECTION

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SECTION 40 - TECHNICAL PUBLICATIONS

GENERAL

The flight and maintenance operations must be carried out according to the officially issued documents which include:

- Interactive Electronic Technical Publication (IETP)
  - Fault Isolation Publication
  - Material Data Information
  - Maintenance Publication
  - Maintenance Planning Information
  - Structural Repair Publication
  - Wiring Diagram Manual
  - Component Repair and Overhaul Manual
  - Illustrated Parts Data Publication
  - Illustrated Tools and Equipment Manual
  - Master Minimum Equipment List
  - NVG Compatibility Reference Handbook
- Rotorcraft Flight Manual (RFM)

In addition a Quick Reference Handbook (QRH) is also available as a checklist that is mandatory for single-pilot operations.
The helicopter publications are prepared in compliance with AECMA Specification S1000D.

INTERACTIVE ELECTRONIC TECHNICAL PUBLICATION (IETP)

The IETP is distributed on CD-ROM and includes all the technical publications used to properly perform all maintenance tasks to permit the Release To Service of the AW139 helicopter, including the Master Minimum Equipment List (MMEL).
A list of the latest Bollettini Tecnici (Technical Bulletins) and Information Letters is also included as a reference.

NOTE A

The Bollettini Tecnici (Technical Bulletins) and the Air vehicle Maintenance Planning Information set (AMPI) are given within the IETP for reference only and are updated at the date of issue of the IETP.
The primary media for the airworthiness aspects are either the paper collection or the Agusta website http://customersupport.agusta.com.

FAULT ISOLATION PUBLICATION
It provides all the information and procedures required by the
MATERIAL DATA INFORMATION
It includes all the data that are necessary for the identification, requisition, storage and utilization of the consumable materials recommended for the helicopter.



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IETP - INTERACTIVE ELECTRONIC TECHNICAL PUBLICATION

MAINTENANCE PUBLICATION
It provides all the information required to perform all the procedures used to preserve the airworthiness and flight characteristic of the helicopter.
It contains the following information:

- inspection requirements
- maintenance procedures
- removal and installation procedures
- test and inspection.

MAINTENANCE PLANNING INFORMATION
This publication gives you all the data necessary for the maintenance planning of the Agusta Westland AW139 helicopter.
The maintenance planning information includes the chapters that follow applicable to the helicopter maintenance information:

- Chapter 04 - Airworthiness limitations
- Chapter 05 - Schedules/unscheduled maintenance requirements

STRUCTURAL REPAIR MANUAL
It provides all the information required for the identification of the structure damages and the repair associated.

WIRING DIAGRAM PUBLICATION
It provides all the electrical/electronic wiring diagrams required for maintenance tasks.

COMPONENT REPAIR AND OVERHAUL MANUAL
It provides all the information required to the disassembly, inspection, repair and reassembly of the major helicopter components when applicable.

ILLUSTRATED PARTS DATA PUBLICATION
It provides the illustration and identification data about the replaceable parts of the air vehicle for which the maintenance procedures has been provided.

ILLUSTRATED TOOLS AND EQUIPMENT MANUAL
It provides all the characteristics and the illustration of all the special tools and equipment, including test equipment recommended for the maintenance of the air vehicle.

MASTER MINIMUM EQUIPMENT LIST
It provides the list of all the airborne equipment which is mandatory to achieve a safe flight condition.

NVG COMPATIBILITY REFERENCE HANDBOOK
It provides the means to determine if a given AW139 configuration is compatible with NVG (Night Vision Goggles) operations and how to manage deviations and non compliances.

ROTORCRAFT FLIGHT MANUAL

It provides all the information required to operate the helicopter in normal and emergency conditions.
It contains the following sections:
PART I EASA APPROVED

- Section 1 - Limitations
- Section 2 - Normal Procedures
- Section 3 - Emergency and Malfunctions Procedures

- Section 4 - Performances Data
- Section 5 - Optional Equipment Supplement

PART II - MANUFACTURER'S DATA

- Section 6 - Weight and Balance
- Section 7 - Systems Description
- Section 8 - Handling and Servicing
- Section 9 - Supplemental Performance Information

SECTION 1 - LIMITATIONS

The Limitations section contains limitations required by regulation or to safely operate rotorcraft, powerplant, systems, and equipment. It includes operating limitations, instrument markings, colour coding, and basic placards.

SECTION 2 - NORMAL PROCEDURES

The Normal Procedures section contains the checklist for the normal procedures ordered by phase of flight.
Normal procedures are the result of extensive flight tests and experience with the AW139 aircraft. They are intended to ensure that the level of safety required by the design and certification process is achieved.

SECTION 3 - EMERGENCY AND MALFUNCTIONS PROCEDURES

The Emergency and Malfunctions Procedures section contains the procedures that must be performed in the event of an emergency or malfunction. These procedures are based on experience acquired in the operation of helicopters, in general, and on flight tests conducted on AW139 helicopter.
The Emergency and Malfunction procedures are presented in the form of logic trees (flow charts). These flow charts have been formulated based on analysis and test of the cockpit indications that would be available to the flight crew following the failures/malfunctions that are included in this section.

The Emergency and Malfunctions Procedures section includes three sets of procedures:

- Emergency procedures for CAS messages
- Malfunction procedures for CAS messages
- Emergency and malfunction procedures for PFD indications

Emergency procedures are related to warning (red) messages/indications.
Malfunction procedures are related to caution (amber) messages/indications.

SECTION 4 - PERFORMANCE DATA

The Performance Data section includes charts with standard performance data based on flight test results and engineering analysis.

SECTION 5 - OPTIONAL EQUIPMENT SUPPLEMENT

The Optional Equipment Supplement section contains all the information necessary to operate optional equipment. Each supplement is arranged in 4 sections:

- Section 1 – Limitations
- Section 2 – Normal procedures
- Section 3 – Emergency and malfunction Procedures
- Section 4 – Performance Data

SECTION 6 - WEIGHT AND BALANCE

The Weight and Balance section contains the charts that permit to determine the aircraft weight and the position of the center of gravity.

SECTION 7 - SYSTEM DESCRIPTION

The System Description section provides a generaldescription of the AW139 helicopter and its systems.

SECTION 8 - HANDLING, SERVICING AND MAINTENANCE
The Handling, Servicing and Maintenance section includes the information to:

- Handle the helicopter on the ground, which consists of towing, parking, securing and mooring
- Service and lubricate the helicopter systems and components

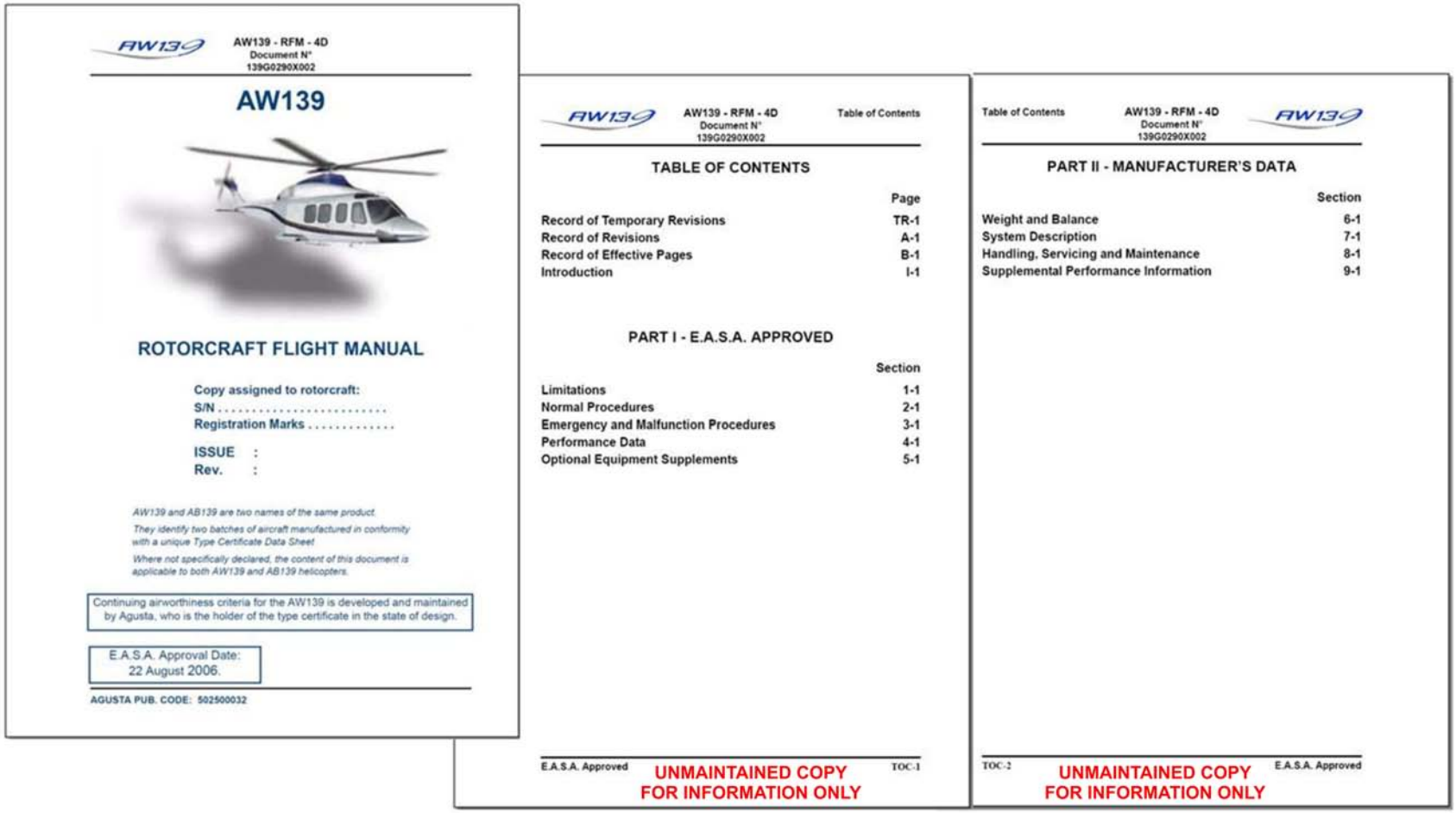
SECTION 9 - SUPPLEMENTAL PERFORMANCE INFORMATION
The Supplemental Performance Information section contains data to be used in conjunction with Section 4 and optional equipment Supplements, as applicable. These data are grouped in sections such as Cruise Charts, Climb Gradient Charts, Hover Ceiling with Headwind Benefit Charts.

QUICK REFERENCE HANDBOOK (QRH)

The QRH consists of 4 sections which have been grouped into two parts:

- The first part combines Limitations, Normal Procedures and Performance Data.
- The second part contains Emergency/Malfunction Procedures.

The two parts are mounted back-to-back to allow quick access to either. The various sections/systems are colour tabbed for ease and quickness of locating the page required. An Index of Content is included at the start of each of the two parts.



AW139 - RFM - 4D
Document N°
139G0290X002

AW139

ROTORCRAFT FLIGHT MANUAL

Copy assigned to rotorcraft:
S/N .....
Registration Marks .....

ISSUE :
Rev. :

AW139 and AB139 are two names of the same product.
They identify two batches of aircraft manufactured in conformity with a unique Type Certificate Data Sheet
Where not specifically declared, the content of this document is applicable to both AW139 and AB139 helicopters.

Continuing airworthiness criteria for the AW139 is developed and maintained by Agusta, who is the holder of the type certificate in the state of design.

E.A.S.A. Approval Date:
22 August 2006.

AGUSTA PUB. CODE: 502500032

AW139 - RFM - 4D
Document N°
139G0290X002
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Emergency and Malfunction Procedures	3-1
Performance Data	4-1
Optional Equipment Supplements	5-1

E.A.S.A. Approved
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139G0290X002

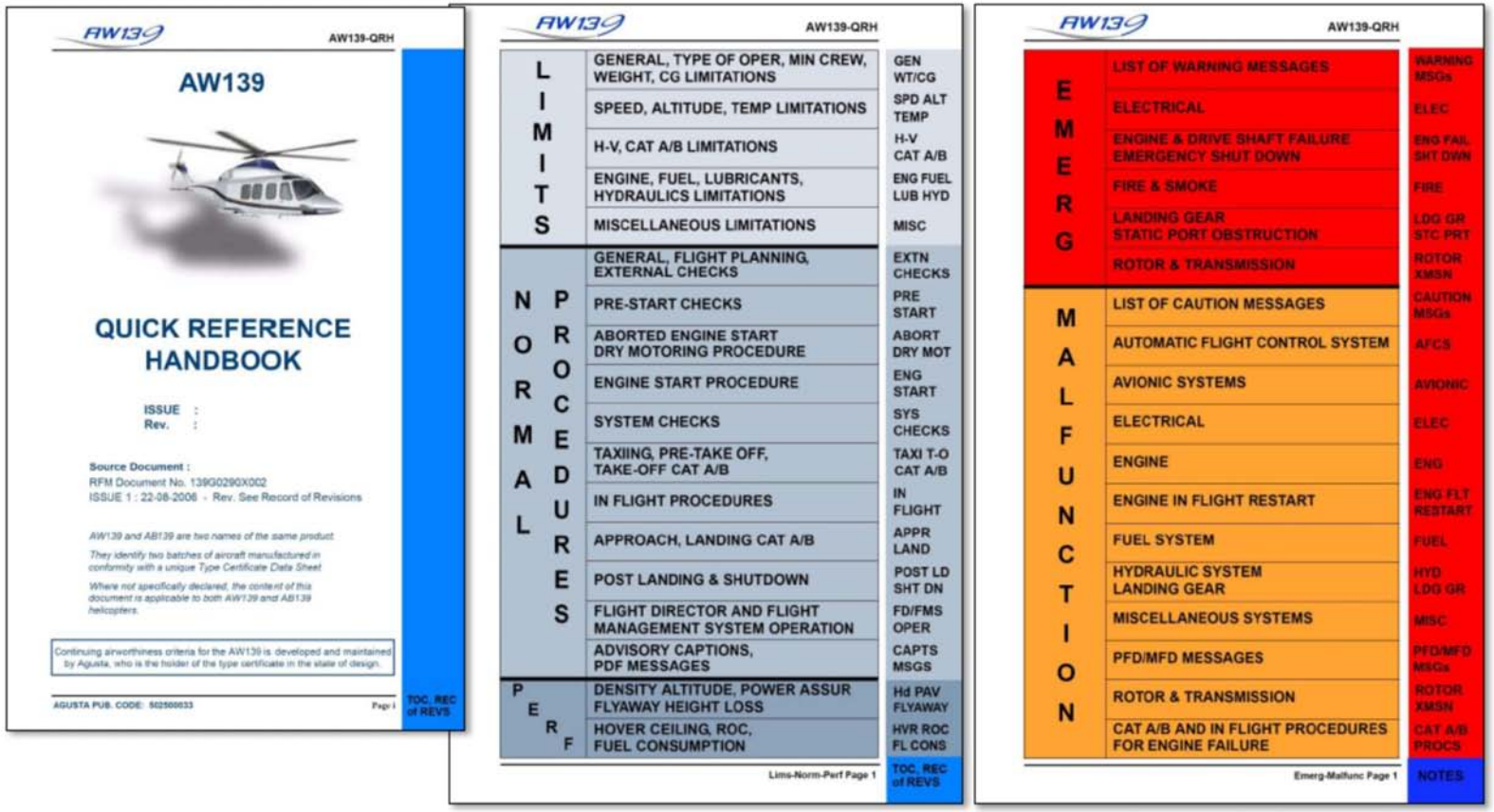
PART II - MANUFACTURER'S DATA

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System Description	7-1
Handling, Servicing and Maintenance	8-1
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UNMAINTAINED COPY
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ICN-39-A-004000-G-A0126-01643-A-01-1

RFM - ROTORCRAFT FLIGHT MANUAL



ICN-39-A-004000-G-A0126-01644-A-01-1

QRH - QUICK REFERENCE HANDBOOK

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SECTION 41 - MAINTENANCE PUBLICATION

GENERAL

The maintenance manual is divided in data modules which provide the required information to perform maintenance on the helicopter.

The data modules can be descriptive or procedural.

The data modules are identified by the following code:

DATA MODULE CODE (DMC)						
	1	**2**	**3**	**4**	**5**	**6**
DMC	39	A	XX-XX-XX	YXA	XXXA	A
Eg: 39-A-29-11-01-00A-720A-A						
	39	A	29-11-01	00	720	A

1 - Model identification code

The model identification code for the AW139 is 39.

2 - System difference code

The system difference code is a letter (letters I and O are not used).
This Letter identifies two or more sub-system that can be installed as alternative items because they do the same function (eg: VHF systems by different suppliers).

3 - Standard Numbering System (SNS) code

The SNS code includes three pairs of digits which show:

- The system (eg: 29, hydraulic power).
- The sub-system and sub-subsystem (eg: 11, Number 1 main-hydraulic system).

NOTE A

When the sub-subsystem code is zero, the two digits show all of the sub-system.

- The component (eg: 01, Number 1 power control module).

4 - Disassembly code and disassembly code variant

Disassembly code
The disassembly code is a two-digit code or three-digit code if variant.
The disassembly code shows the breakdown of a component, as follow:

- 00 – Data module for all of the helicopter, system, sub-system or component.
- 01 – Data module for the first assembly that you remove from the component.
- 02 – Data module for the second assembly that you remove from the component.

Code 03 and the subsequent codes refer to the third, fourth, etc assembly that you remove from the component.
The disassembly code also identifies the data modules in sequence.
Disassembly code variant

The disassembly code variant is a letter (letter I and O are not used) which identifies alternative items.
These items are different, but the difference is too small to cause a change in the system difference code.

5 - Information code and information code variant

Information code
The information code is a three-digit code.
The information code identifies the types of information you can find in the data module.
000 is used for the systems, for which the SNS identifies the type of information in the data module (eg: 05-10 is “time limits”, you must write 000).
The primary codes used in the publication set for the helicopter are written below:

Code — **Definition**

- 040 — Description of how it is made and its function
- 050 — Diagrams and lists
- 060 — Aerospace ground equipment, tools and software
- 070 — Consumables, material and expendables
- 080 — Mixtures and solutions
- 090 — Software documentation
- 200 — Servicing
- 500 — Disconnect, remove and disassemble procedures
- 510 — Disconnect procedures
- 520 — Remove procedures
- 530 — Disassemble procedures
- 550 — Unload software procedures
- 600 — Repairs and locally make procedures and data
- 610 — Add materials
- 620 — Attach materials
- 630 — Change the mechanical strength/ structure of materials
- 640 — Change the surface finish of materials
- 650 — Remove materials
- 660 — Structure repair procedures and data
- 670 — Locally make procedures and data
- 700 — Assemble, install and connect procedures
- 710 — Assemble procedures
- 720 — Install procedures
- 800 — Storage procedures and data
- 810 — Preservation procedures
- 820 — Procedures to remove preservation material
- 830 — Procedures to put items in containers
- 840 — Procedures to remove items from containers

- 850..... Procedures to keep items serviceable when in storage
- 860..... Procedures to move items when in storage

Information code variant
The information code variant is a letter (letters I and O are not used).
The information code variant is used for different data modules that are applicable to the same component and type of operation.

6 - Item location code

The item location code is a letter which shows where you must do the maintenance task.
In this publication, you can find these item location codes:

- A on the helicopter.
- B on the removed assembly/component.
- D applicable to all the locations.

The letter K is used as the item location code for the identification of the data modules applicable to:

- Optional installations.
- Optional systems.
- Optional equipment.

References
Below this heading you find a list of the data modules and technical publications which are included as references in the data module.
Preliminary requirement
Below this heading you find all the data you must know before you do the procedure.
The preliminary requirements are written below these headings:

- Required conditions
- Support equipment
- Supplies
- Spares
- Safety conditions.

Required conditions
Below this heading you find a list of the tasks to be performed before starting the procedure.
Support equipment
Below this heading you find a list of the special tools and support equipment that are necessary to do the procedure.
The identification number is the same number as used in the Illustrated Tool and Equipment publication.
Supplies
Below this heading you find a list of all of the consumables, materials and expendables necessary to do the procedure.
Spares
Below this heading you find a list of the spares that are necessary to do the procedure.
Safety conditions
Below this heading you find the general warnings, cautions and notes that are applicable to the data module.
Procedure
Below this heading you find the sequence of the steps and illustrations which give the instructions necessary to do the procedure.
Requirements after job completion
Below this heading you find all the instructions necessary to put the helicopter back to operation.

DC main generation system – Description of how it is made

Table of contents

References
1 Primary components
2 Number 1 starter generator
3 Number 1 starter generator mount flange
4 Number 1 starter generator air inlet duct
5 Number 1 starter generator air outlet duct
6 Number 1 generator control unit
7 Number 1 shunt
8 Number 2 starter generator
9 Number 2 starter generator mount flange
10 Number 2 starter generator air inlet duct
11 Number 2 starter generator air outlet duct
12 Number 2 generator control unit
13 Number 2 shunt
14 Number 3 and Number 4 DC current transformers
15 Number 1 and Number 2 DC current transformers
16 GEN 1 and GEN 2 switches

List of tables

1 References

List of figures

1 DC main generation system - Primary components (Sheet 1 of 4)
2 DC main generation system - Primary components (Sheet 2 of 4)
3 DC main generation system - Primary components (Sheet 3 of 4)
4 DC main generation system - Primary components (Sheet 4 of 4)

References

Table 1 References

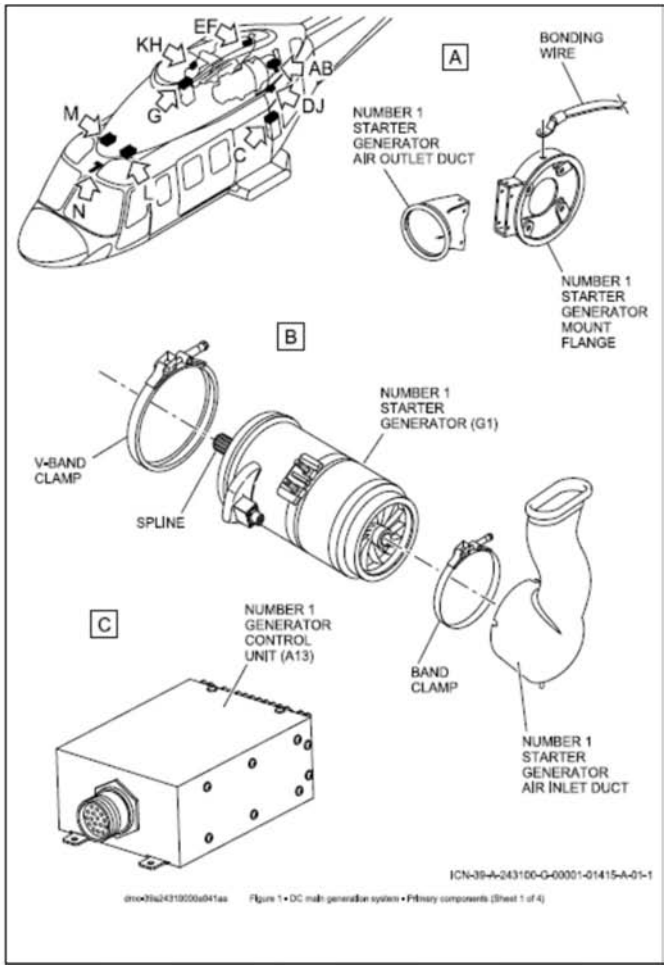
Data Module	Title
No references	

Description

1 Primary components

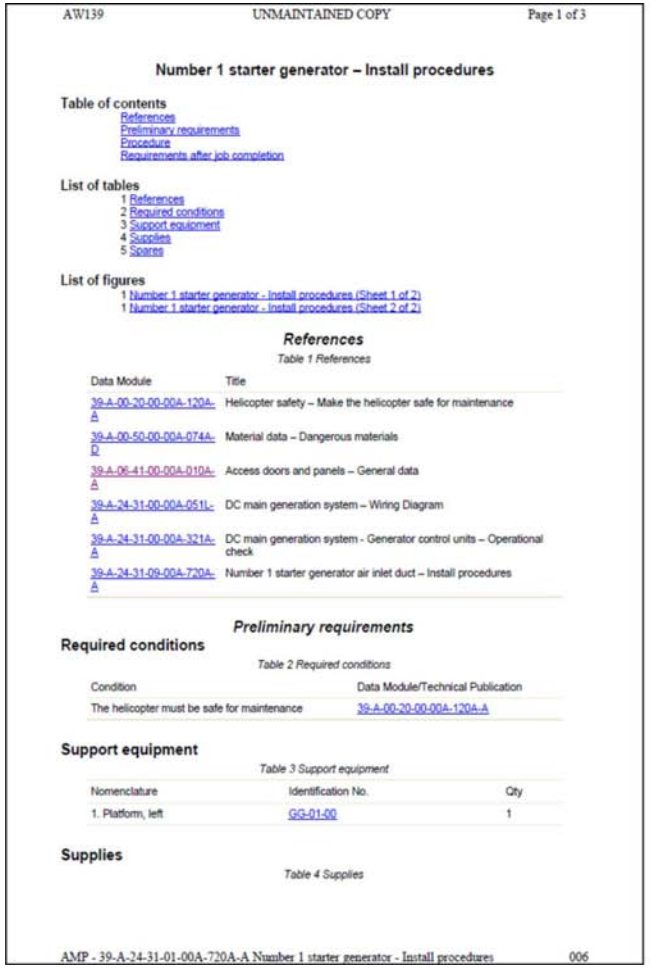
The DC main generation system includes these primary components:

– The Number 1 starter generator
– The Number 1 starter-generator mount-flange
– The Number 1 starter-generator air-inlet duct
– The Number 1 starter-generator air-outlet duct
– The Number 1 generator-control unit
– The Number 1 shunt
– The Number 2 starter generator
– The Number 2 starter-generator mount-flange
– The Number 2 starter-generator air-inlet duct
– The Number 2 starter-generator air-outlet duct



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DESCRIPTIVE DATA MODULE



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Number 1 starter generator – Install procedures

Table of contents

References
Preliminary requirements
Procedure
Requirements after job completion

List of tables

1 References
2 Required conditions
3 Support equipment
4 Supplies
5 Spares

List of figures

1 Number 1 starter generator - Install procedures (Sheet 1 of 2)
1 Number 1 starter generator - Install procedures (Sheet 2 of 2)

References

Table 1 References

Data Module	Title
39-A-00-20-00-00A-120A-A	Helicopter safety – Make the helicopter safe for maintenance
39-A-00-50-00-00A-074A-D	Material data – Dangerous materials
39-A-06-41-00-00A-010A-A	Access doors and panels – General data
39-A-24-31-00-00A-051L-A	DC main generation system – Wiring Diagram
39-A-24-31-00-00A-321A-A	DC main generation system - Generator control units – Operational check
39-A-24-31-09-00A-720A-A	Number 1 starter generator air inlet duct – Install procedures

Preliminary requirements

Required conditions

Table 2 Required conditions

Condition	Data Module/Technical Publication
The helicopter must be safe for maintenance	39-A-00-20-00-00A-120A-A

Support equipment

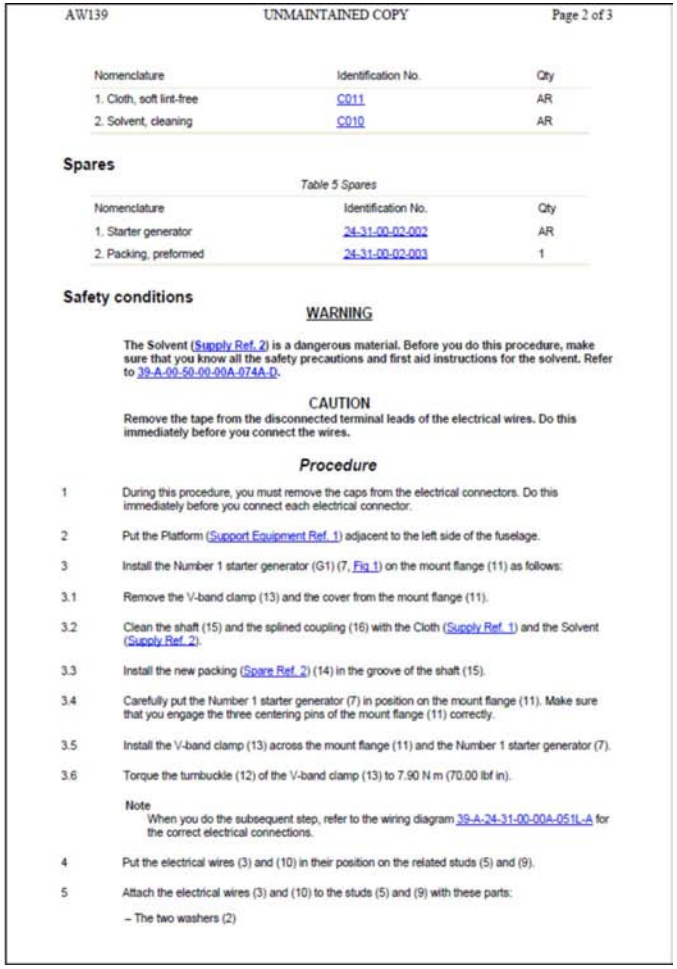
Table 3 Support equipment

Nomenclature	Identification No.	Qty
1. Platform, left	GG-01-00	1

Supplies

Table 4 Supplies

AMP - 39-A-24-31-01-00A-720A-A Number 1 starter generator - Install procedures 006



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Nomenclature	Identification No.	Qty
1. Cloth, soft lint-free	C011	AR
2. Solvent, cleaning	C010	AR

Spares

Table 5 Spares

Nomenclature	Identification No.	Qty
1. Starter generator	24-31-00-02-002	AR
2. Packing, preformed	24-31-00-02-003	1

Safety conditions

WARNING

The Solvent (Supply Ref. 2) is a dangerous material. Before you do this procedure, make sure that you know all the safety precautions and first aid instructions for the solvent. Refer to 39-A-00-50-00-00A-074A-D.

CAUTION

Remove the tape from the disconnected terminal leads of the electrical wires. Do this immediately before you connect the wires.

Procedure

1 During this procedure, you must remove the caps from the electrical connectors. Do this immediately before you connect each electrical connector.

2 Put the Platform (Support Equipment Ref. 1) adjacent to the left side of the fuselage.

3 Install the Number 1 starter generator (G1) (7, Fig 1) on the mount flange (11) as follows:

3.1 Remove the V-band clamp (13) and the cover from the mount flange (11).

3.2 Clean the shaft (15) and the splined coupling (16) with the Cloth (Supply Ref. 1) and the Solvent (Supply Ref. 2).

3.3 Install the new packing (Spare Ref. 2) (14) in the groove of the shaft (15).

3.4 Carefully put the Number 1 starter generator (7) in position on the mount flange (11). Make sure that you engage the three centering pins of the mount flange (11) correctly.

3.5 Install the V-band clamp (13) across the mount flange (11) and the Number 1 starter generator (7).

3.6 Torque the turnbuckle (12) of the V-band clamp (13) to 7.90 N·m (70.00 lbf·in).

Note
When you do the subsequent step, refer to the wiring diagram 39-A-24-31-00-00A-051L-A for the correct electrical connections.

4 Put the electrical wires (3) and (10) in their position on the related studs (5) and (9).

5 Attach the electrical wires (3) and (10) to the studs (5) and (9) with these parts:

– The two washers (2)

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PROCEDURAL DATA MODULE (SHEET 1 OF 2)

– The two self-locking nuts (1).

6 Torque the two self-locking nuts (1) to 24.9 thru 26.6 N m (220 thru 235 lbf in).

7 Connect the rubber cover (4) to the terminal board (6) with your hand.

8 Connect the connector G1P1 (8) to the Number 1 starter generator (G1) (7).

Requirements after job completion

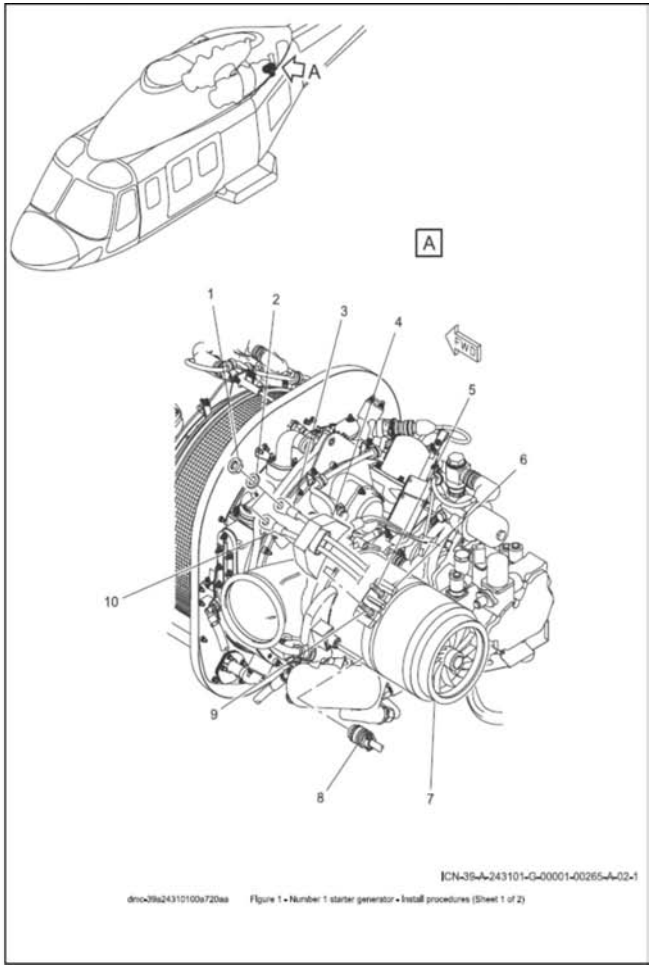
1 Remove all the tools and the other items from the work area. Make sure that the work area is clean.

2 Close the access door 461AL. Refer to 39-A-06-41-00-00A-010A-A.

3 Remove the platform from the left side of the fuselage.

4 Install the Number 1 starter generator air inlet duct. Refer to 39-A-24-31-09-00A-720A-A.

5 Do the operational check of the DC main generation system. Refer to 39-A-24-31-00-00A-321A-A.



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PROCEDURAL DATA MODULE (SHEET 2 OF 2)

Maintenance publication - Table of contents

Chapter	Title	Issue	Date	Change	Date
00	Helicopter general	Third	2006-07-15	9	2009-11-15
06	Dimensions and areas	Third	2006-07-15	3	2009-11-15
07	Lifting, shoring, recovering and transporting	Third	2006-07-15	4	2009-11-15
08	Leveling and weighing	Third	2006-07-15	--	--
09	Handling and taxiing	Third	2006-07-15	2	2009-11-15
10	Parking, mooring and storing	Fourth	2008-07-31	2	2009-11-15
11	Placards and markings	Third	2006-07-15	3	2009-11-15
12	Servicing	Third	2006-07-15	6	2009-11-15
18	Vibration and noise analysis and attenuation	Fourth	2007-10-15	5	2009-11-15
20	Standard practices - Airframe systems	Third	2006-07-15	6	2009-11-15
21	Environmental control	Third	2006-07-15	6	2009-11-15
22	Auto flight	Third	2006-07-15	7	2009-11-15
23	Communications	Third	2006-07-15	8	2009-11-15
24	Electrical power	Third	2006-07-15	7	2009-11-15
25	Equipment/furnishings	Fourth	2006-07-15	7	2009-02-28
26	Fire protection	Third	2006-07-15	7	2009-11-15
28	Fuel	Third	2006-07-15	7	2009-11-15
29	Hydraulic power	Third	2006-07-15	7	2009-11-15
30	Ice and rain protection	Fourth	2007-07-15	4	2009-11-15
31	Indicating/recording systems	Third	2006-07-15	8	2009-11-15
32	Landing gear	Third	2006-07-15	8	2009-11-15
33	Lights	Fourth	2007-07-15	5	2009-11-15
34	Navigation	Third	2006-07-15	8	2009-11-15

45	Central maintenance system (CMS)	Third	2006-07-15	4	2009-02-28
46	System integration and display	Second	2008-07-31	2	2009-11-15
52	Doors	Third	2006-07-15	6	2009-11-15
53	Fuselage	Third	2006-07-15	8	2009-11-15
55	Stabilizers	Third	2006-07-15	5	2009-11-15
56	Windows	Third	2006-07-15	7	2009-11-15
60	Standard practices - rotors	Third	2006-07-15	3	2009-02-28
62	Main rotor	Third	2006-07-15	8	2009-11-15
63	Main rotor drive	Third	2006-07-15	8	2009-11-15
64	Tail rotor	Third	2006-07-15	8	2009-11-15
65	Tail rotor drive	Third	2006-07-15	7	2009-11-15
67	Rotors flight control	Fourth	2008-02-29	3	2009-11-15
71	Power plant	Third	2006-07-15	7	2009-11-15
73	Engine fuel and control	Third	2006-07-15	3	2009-02-28
74	Ignition	Third	2006-07-15	3	2009-02-28
75	Air	Third	2006-07-15	4	2009-02-28
76	Engine controls	Third	2006-07-15	8	2009-11-15
78	Exhaust	Third	2006-07-15	4	2009-11-15
80	Starting	Third	2006-07-15	--	--
91	Helicopter wiring	Third	2006-07-15	2	2008-02-29
93	Surveillance	First	2007-10-15	4	2009-11-15
95	Crew escape and safety	Second	2006-07-15	8	2009-11-15
97	Image recording	First	2008-07-31	1	2009-02-28

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Record of temporary revision

TR No	Issue date	Insertion date	by (signature)	Removal date	by (signature)
001	2006-02-15	2006-02-18	AGU	2006-07-15	AGU
002	2007-11-06	2007-11-06	AGU	2008-02-29	AGU
003	2008-07-21	2008-07-21	AGU	2008-09-15	AGU
004	2008-08-29	2008-08-29	AGU	2008-09-15	AGU
005					
006					
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s - Keep this page in the publication immediately after the cover page - All the data modules included in a Temporary Revision (TR) can be found in the List of Effective Data Modules (LOEDM) page of the related TR

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RECORD OF TEMPORARY REVISION PAGE

List of effective data modules

Dates of issue and changes are:
Third issue 2006-07-15
Change 9 2009-11-15

The Third Issue of this chapter, after the insertion of Change 9, includes the modules that follow:

Document identifier	Issue Type	Issue Date	Effectivity
39-A-00-00-00-00A-011A-D	R	2009-09-30 *	All
39-A-00-00-00-00A-015A-D	C	2008-06-08	All
39-A-00-00-00-00A-017A-D	C	2009-02-03	All
39-A-00-00-00-00A-018A-D	C	2007-09-26	All
39-A-00-00-00-00A-032A-A	R	2009-09-30 *	All
39-A-00-00-00-00A-044A-A	N	2008-07-28	All
39-A-00-00-00-00A-044B-A	C	2009-02-19 *	All
39-A-00-00-00-00A-345A-A	N	2006-05-02	All
39-A-00-00-00-00A-750A-A	N	2009-04-16 *	All
39-A-00-10-00-00A-915A-A	C	2009-07-21 *	All
39-A-00-10-00-00A-916A-A	C	2009-07-21 *	All
39-A-00-10-00-00A-917A-A	N	2006-12-20	All
39-A-00-10-00-00A-918A-A	N	2006-12-21	All
39-A-00-20-00-00A-120A-A	C	2009-01-16	All
39-A-00-50-00-00A-071A-D	N	2004-03-15	All
39-A-00-50-00-00A-074A-D	N	2004-03-15	All
39-A-00-60-00-00A-010A-A	R	2009-09-07 *	All
39-A-00-70-00-00A-288A-A	N	2006-05-02	All
39-A-00-70-00-01A-288A-A	N	2005-03-11	All
39-A-00-70-00-02A-288A-A	N	2005-03-11	All
39-A-00-70-00-03A-288A-A	R	2007-06-11	All
39-A-00-70-00-04A-288A-A	C	2006-05-02	All
39-A-00-70-00-05A-288A-A	R	2007-05-30	All
39-A-00-70-00-06A-288A-A	R	2007-05-30	All
39-A-00-70-00-06A-288B-A	N	2007-06-11	All
39-A-00-70-00-06A-288C-A	N	2007-05-30	All
39-A-00-70-00-07A-288A-A	R	2009-06-09 *	All
39-A-00-70-00-08A-288A-A	C	2006-05-02	All
39-A-00-70-00-09A-288A-A	N	2005-03-11	All
39-A-00-70-00-10A-288A-A	C	2006-05-02	All
39-A-00-70-00-11A-288A-A	N	2005-03-11	All
39-A-00-70-00-12A-288A-A	C	2007-05-25	All

39-A-00-70-00-13A-288A-A	N	2005-03-11	All
39-A-00-70-00-14A-288A-A	C	2006-05-02	All
39-A-00-80-00-06A-369A-A	R	2009-10-09 *	All

C = Changed module N = New module
D = Deleted module R = Revised module
(*) The asterisks identify the modules changed, revised, added or deleted with this change

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LIST OF EFFECTIVE DATA MODULES PAGES

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CHAPTER 04

WORTHINESS (FIT FOR PURPOSE) LIMITATIONS

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SECTION 00 - WORTHINESS (FIT FOR PURPOSE) LIMITATIONS

GENERAL

This chapter provides the airworthiness limitations applicable to the AW139 helicopter.
The airworthiness limitation are approved by EASA and cannot be changed without approval.
The airworthiness limitations includes the following contents:

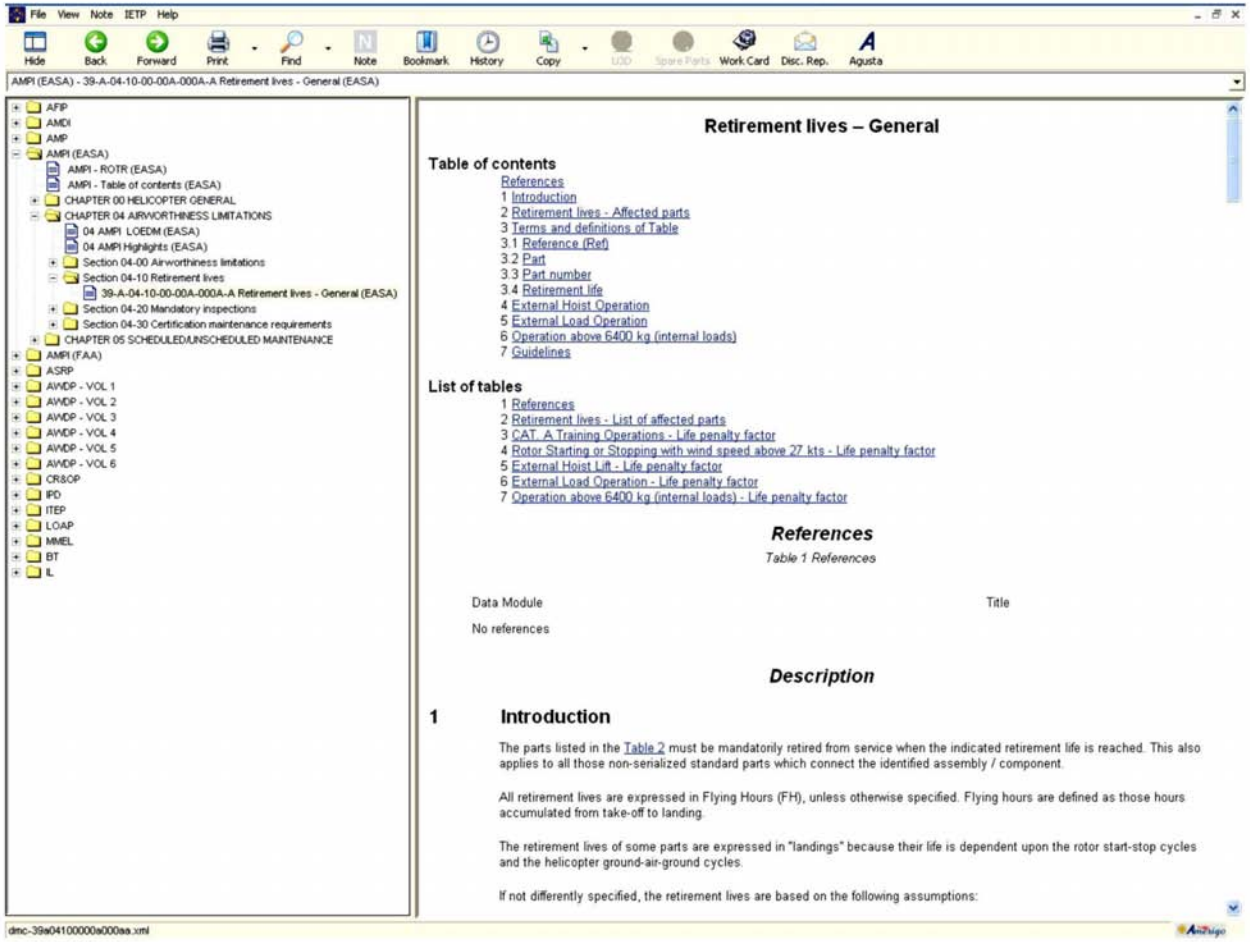
- The retirement lives
- The mandatory inspections
- The certifications requirements

RETIREMENT LIVES

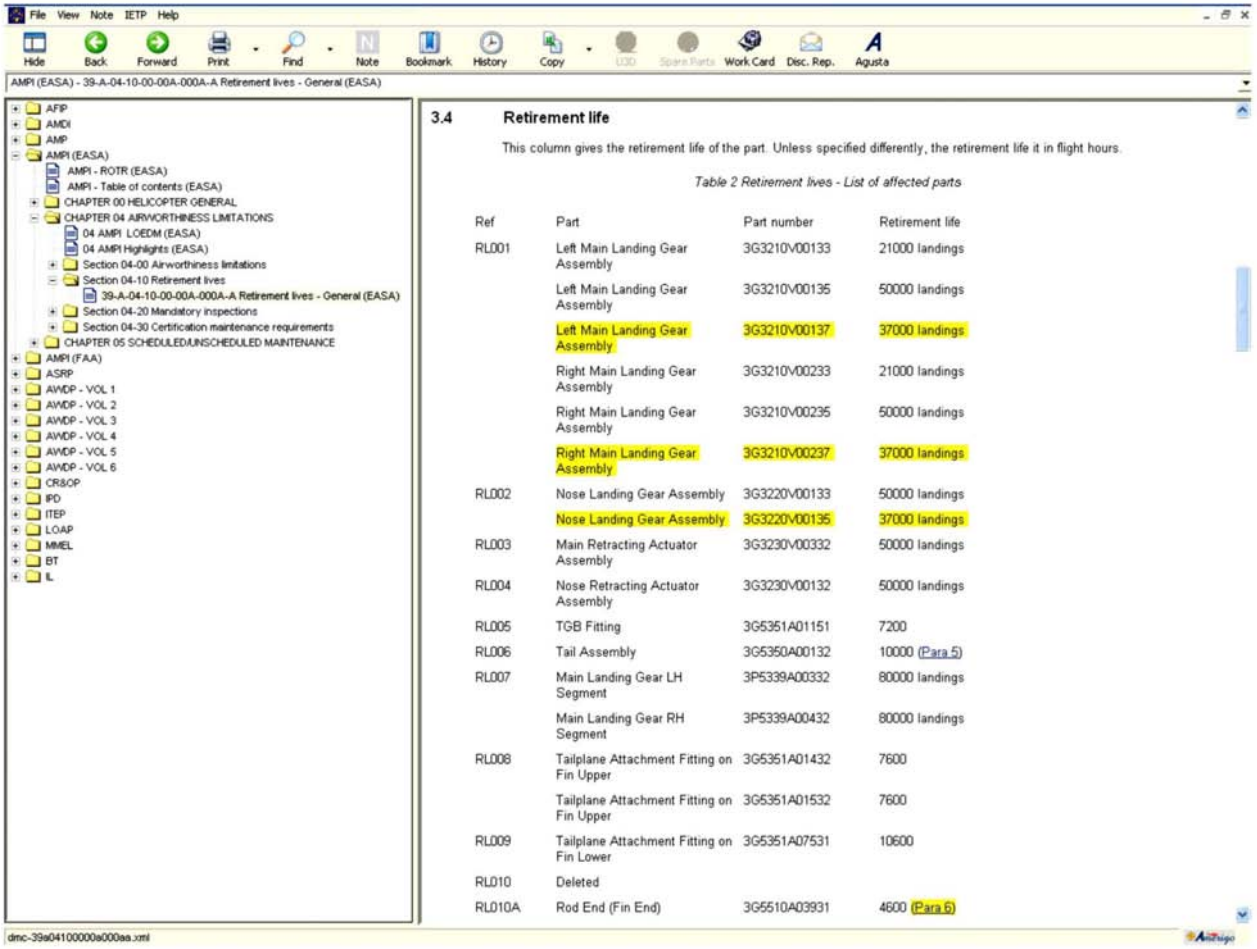
These parts must be mandatorily retired from service when the indicated retirement life is reached. This also applies to all those non-serialized standard parts with connect the indentified assembly component.
The retirement life specified for any given part number contained in this section applies to indicated and all successive dash numbers for that item, if not differently specified.
All retirement lives are expressed in Flying Hours (FH), if not differently specified. Flying hours are defined as those hours accumulated from take-off landing.
The retirement lives of some parts are expressed in “landings” because their life is dependent upon the rotor start-stop cycles and the helicopter ground-air-ground cycles.
If not differently specified, the retirement lives are based on the following assumptions:

- 800 landings in 100 flying hours, including 400 rotor start-stop cycles.
- 20 percent of usage in CAT. A Training Operations, on yearly basis.
- For External Load Operations: 6 External Load Cycles and 3 Landings per Flying Hour, calculated on at least 100 FH basis;
- For External Hoist Operations: 1.65 Lift per Flying Hour, calculated on at least 100 FH basis.

Any mission profile using more cycles per hour than those listed above requires the retirement lives to be recalculated and approved by EASA.
If parts with the same part number can be interchanged between different AW139 variants, the retirement life of the part must be restricted to the lowest value between those variants.



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3.4 Retirement life

This column gives the retirement life of the part. Unless specified differently, the retirement life it in flight hours.

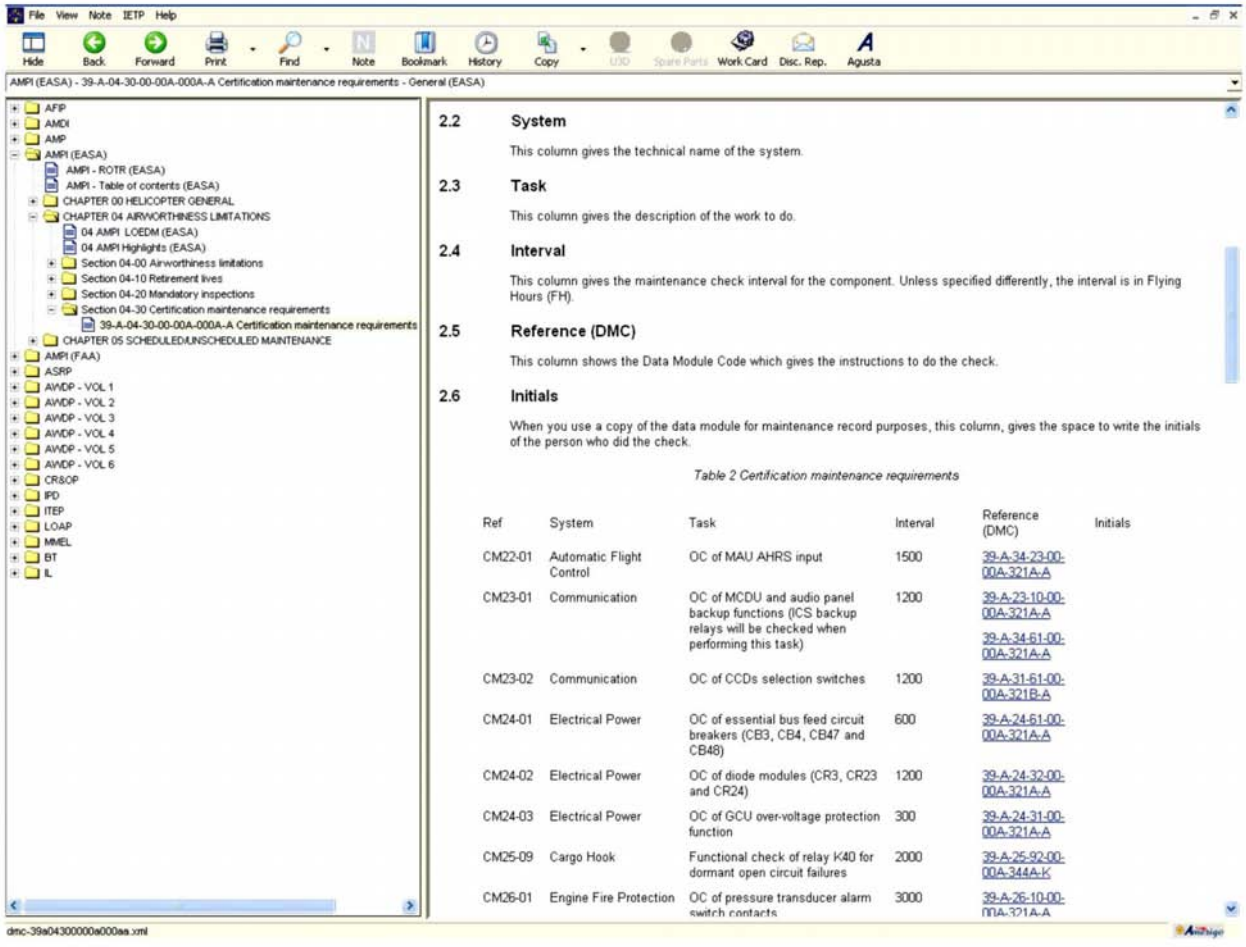
Table 2 Retirement lives - List of affected parts

Ref	Part	Part number	Retirement life
RL001	Left Main Landing Gear Assembly	3G3210V00133	21000 landings
	Left Main Landing Gear Assembly	3G3210V00135	50000 landings
	Left Main Landing Gear Assembly	3G3210V00137	37000 landings
	Right Main Landing Gear Assembly	3G3210V00233	21000 landings
	Right Main Landing Gear Assembly	3G3210V00235	50000 landings
	Right Main Landing Gear Assembly	3G3210V00237	37000 landings
RL002	Nose Landing Gear Assembly	3G3220V00133	50000 landings
	Nose Landing Gear Assembly	3G3220V00135	37000 landings
RL003	Main Retracting Actuator Assembly	3G3230V00332	50000 landings
RL004	Nose Retracting Actuator Assembly	3G3230V00132	50000 landings
RL005	TGB Fitting	3G5351A01151	7200
RL006	Tail Assembly	3G5350A00132	10000 (Para 5)
RL007	Main Landing Gear LH Segment	3P5339A00332	80000 landings
	Main Landing Gear RH Segment	3P5339A00432	80000 landings
RL008	Tailplane Attachment Fitting on Fin Upper	3G5351A01432	7600
	Tailplane Attachment Fitting on Fin Upper	3G5351A01532	7600
RL009	Tailplane Attachment Fitting on Fin Lower	3G5351A07531	10600
RL010	Deleted		
RL010A	Rod End (Fin End)	3G5510A03931	4600 (Para 6)

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MANDATORY INSPECTIONS

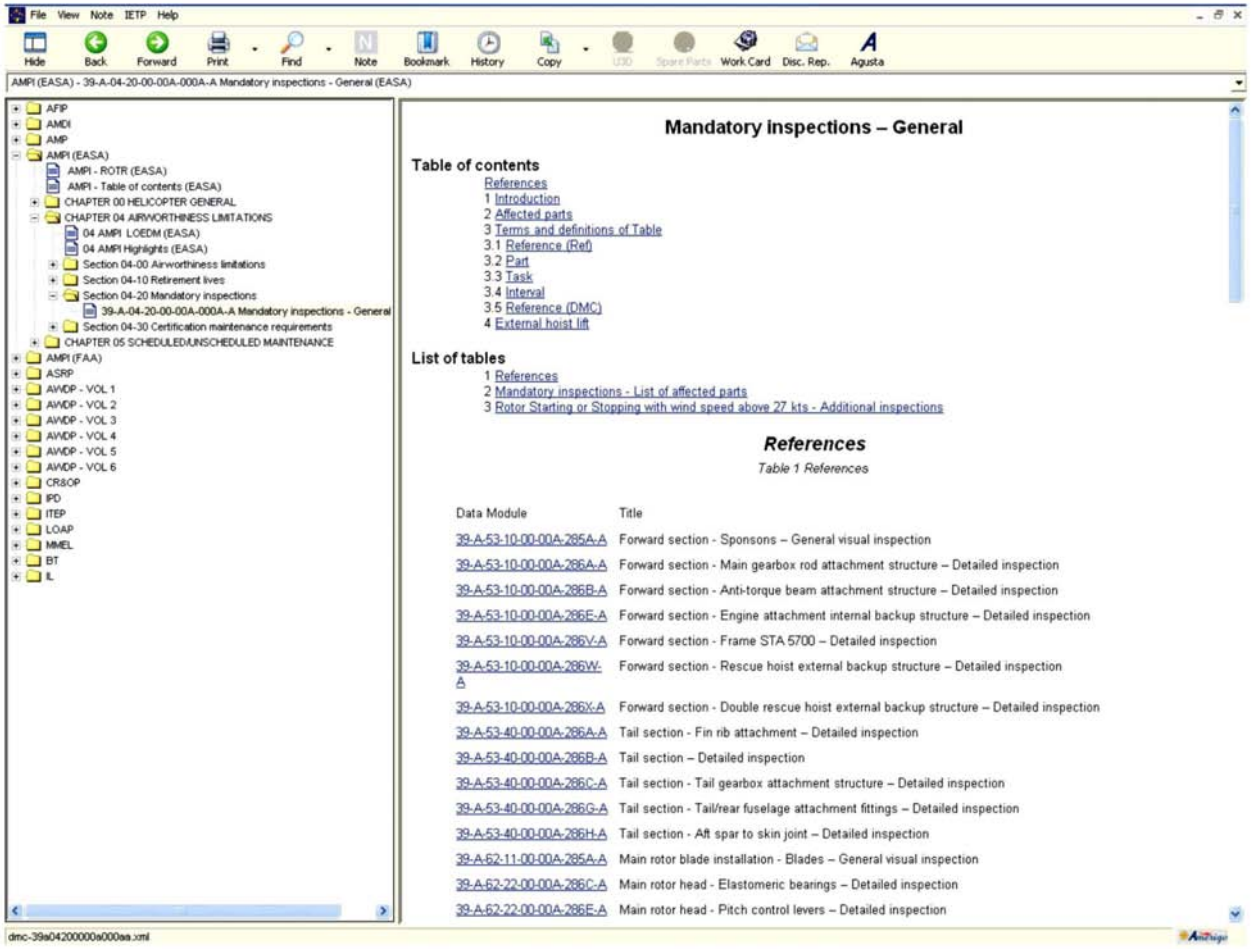
These parts must be mandatorily inspect when the interval is reached.



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CERTIFICATION MAINTENANCE REQUIREMENTS

This section gives you the data about the mandatory maintenance checks identified during the certification process. The parts listed in the schedule that follows must be mandatorily inspected when the indicated interval is reached. No tolerance is permitted on Certification Maintenance Requirements.



Mandatory inspections – General

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References
1 Introduction
2 Affected parts
3 Terms and definitions of Table
3.1 Reference (Ref)
3.2 Part
3.3 Task
3.4 Interval
3.5 Reference (DMC)
4 External hoist lift

List of tables

1 References
2 Mandatory inspections - List of affected parts
3 Rotor Starting or Stopping with wind speed above 27 kts - Additional inspections

References

Table 1 References

Data Module	Title
39-A-53-10-00-00A-285A-A	Forward section - Sponsons – General visual inspection
39-A-53-10-00-00A-286A-A	Forward section - Main gearbox rod attachment structure – Detailed inspection
39-A-53-10-00-00A-286B-A	Forward section - Anti-torque beam attachment structure – Detailed inspection
39-A-53-10-00-00A-286E-A	Forward section - Engine attachment internal backup structure – Detailed inspection
39-A-53-10-00-00A-286V-A	Forward section - Frame STA 5700 – Detailed inspection
39-A-53-10-00-00A-286W-A	Forward section - Rescue hoist external backup structure – Detailed inspection
39-A-53-10-00-00A-286X-A	Forward section - Double rescue hoist external backup structure – Detailed inspection
39-A-53-40-00-00A-286A-A	Tail section - Fin rib attachment – Detailed inspection
39-A-53-40-00-00A-286B-A	Tail section – Detailed inspection
39-A-53-40-00-00A-286C-A	Tail section - Tail gearbox attachment structure – Detailed inspection
39-A-53-40-00-00A-286G-A	Tail section - Tail/rear fuselage attachment fittings – Detailed inspection
39-A-53-40-00-00A-286H-A	Tail section - Aft spar to skin joint – Detailed inspection
39-A-62-11-00-00A-285A-A	Main rotor blade installation - Blades – General visual inspection
39-A-62-22-00-00A-286C-A	Main rotor head - Elastomeric bearings – Detailed inspection
39-A-62-22-00-00A-286E-A	Main rotor head - Pitch control levers – Detailed inspection

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CHAPTER
05
SCHEDULED/UNSCHEDULED MAINTENANCE

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SECTION 00 - SCHEDULED/UNSCHEDULED MAINTENANCE

GENERAL

This chapter contains the inspection requirements concerning the AW139 helicopters equipped with PW PT6-67 C engines, but does not contain inspection procedures. Such procedures will be found in the pertinent Chapters of the Maintenance Manual.
The inspections are presented in typographic form suitable for the local reproduction in such a way as to be used by personnel to perform helicopter inspections and to constitute, if desired, a data collection.
The inspections must be accomplished by qualified personnel to ascertain the airworthiness of the helicopter. Eventual discrepancies must be eliminated before flight.
Prior to inspection, open or remove as necessary, fairings, cowlings panels and inspection doors.
Flight airworthiness must be determined by inspection of all parts to check the general condition, security of attachment, cleanliness, freedom of movement and security, in accordance with the operational inspection rules and instructions contained in the pertinent Chapters of the Maintenance Manual.
The airworthiness of the helicopter is determined by acceptance of all parts with quality standard aircraft practice and specified instructions contained in the pertinent Chapters of Maintenance Manual.
Each inspected component must meet pertinent and applicable Airworthiness Directives requirements.
For acceptance requirements and limits refer to the pertinent Chapters of the Maintenance Manual and/or Overhaul Manual.
The inspection and maintenance of the engine, electronic equipment, instruments, etc. must be performed in accordance with the instructions and procedures required by relevant manufacturers and in accordance with approved maintenance schedule.
The inspection and overhaul schedule requirements contained in the Maintenance Manual must not be exceeded without explicit approval.

NOTE A

When operating under particular environmental conditions (contaminated ambient, near the sea or special missions) it is prerogative and responsibility of the operator to increase or intensify the prescribed inspections as necessary to assure safe operation.

Permissible inspection interval tolerances

In order to facilitate the inspection planning in accordance with the helicopter inspection schedule, some tolerances are permissible.

NOTE B

Tolerances are not cumulative and do not change the date at which the next inspection was scheduled.

The AMPI gives you all the data necessary for the maintenance planning of the AgustaWestland AW139 helicopter.
The maintenance requirements identified in AMPI chapter 05, together with those in the following list, constitute the instructions for Continued Airworthiness for the helicopter:

- Airworthiness Limitations as in Chapter 04
- Helicopter pre-flight checks identified in Section 2 of the Rotorcraft Flight Manual.
- PT6C-67C engine scheduled maintenance requirements.
- Technical bulletins, where applicable to the specific helicopter serial number configuration.

The maintenance requirements identified in AMPI chapter 05 includes:

- 05-10 Time limits
  - 05-11 Permitted inspection interval tolerance

- 05-12 Components overhaul schedule
- 05-13 Discard time schedule

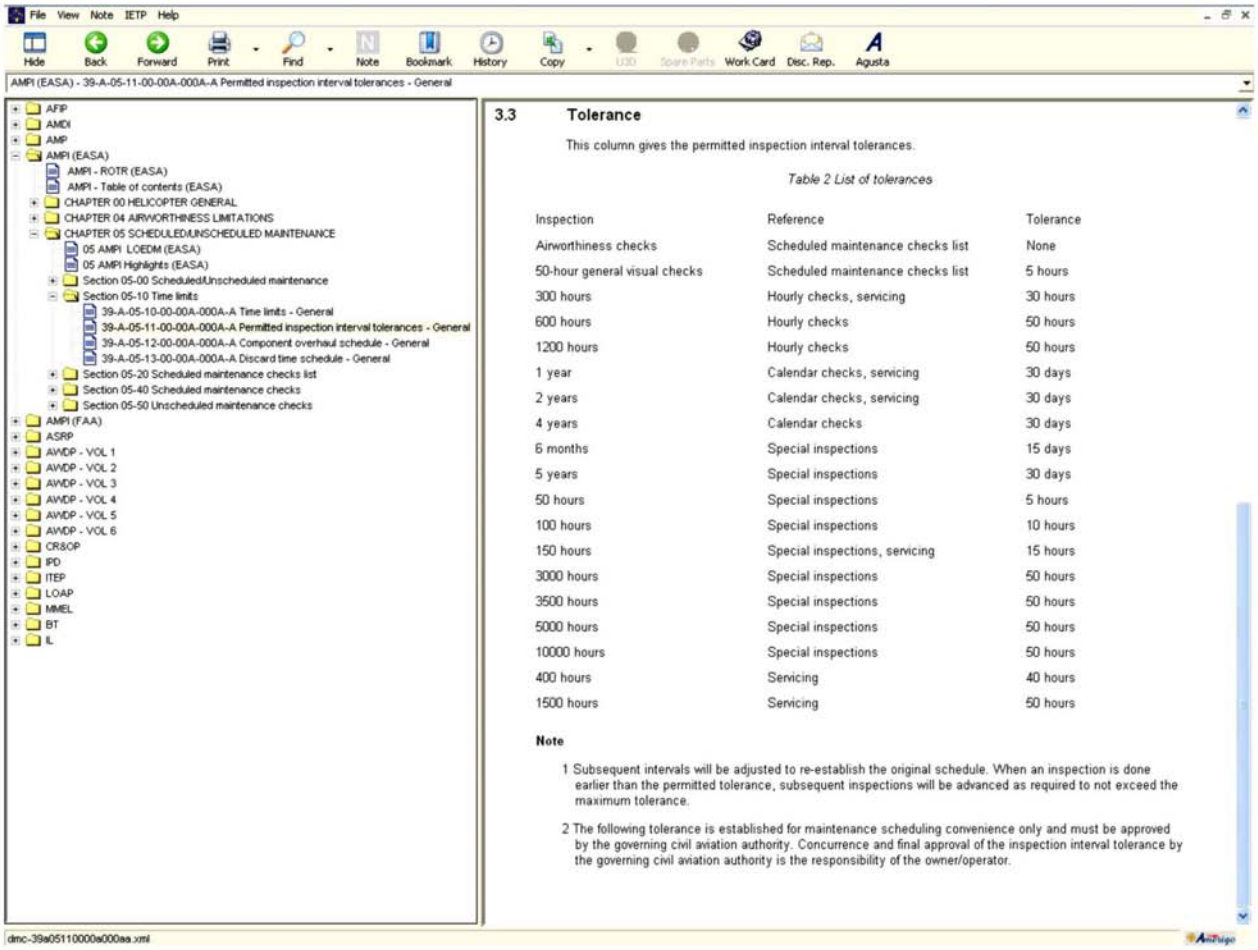
- 05-20 Scheduled maintenance checks lists
  - 05-21 Overview
  - 05-22 Airworthiness checks
  - 05-23 50 Hours general visual inspection
- 05-40 Scheduled maintenance checks
  - 05-41 Hourly checks
  - 05-42 Calendar checks
  - 05-43 Special inspections
  - 05-44 Servicing
- 05-50 Unscheduled maintenance checks
  - 05-51 Out of phase maintenance
  - 05-52 Conditional inspection

05-10 TIME LIMITS

This section gives the recommended time limits requirements for the components of the AW139.
The time limits are divided into:

- Permitted inspection interval tolerances;
- Components overhaul schedule;
- Discard time schedule

The tolerance is established for maintenance scheduling convenience only and must be approved by the governing civil aviation authority. Concurrence and final approval of the inspection interval tolerance by the governing civil aviation authority is the responsibility of the owner/operator. Permitted inspection interval tolerances introduce a policy of 10% of tolerance to the schedule interval with a maximum value of 50 flight hours and 30 days. Subsequent intervals should be adjusted to re-establish the original schedule. When an inspection is done earlier than the permitted tolerance, subsequent inspections should be advanced as required to not exceed maximum tolerance.
The components overhaul schedule give a list of overhaul components installed on AW139 at which no tolerance is applicable. In case of an Operator desire an overhaul interval extension he should submit a formal request to AgustaWestland. The overhaul intervals, specified for the AgustaWestland part numbers (e.g.:3G6310A00531), apply to all successive part numbers with the same first ten digits and different last two digits, unless different specified.
The Discard time schedule gives the indication of the numbers of hours/ months/years at which point the component must be discarded. The discard time of some parts are expressed in “landings” because their usage is dependent upon the rotor start-stop cycles and the helicopter ground air–ground cycles. No tolerance is permitted on the discard time. The discard times specified for the AgustaWestland part numbers (e.g.: 3G3350A01811) apply also to all successive part numbers with the same first ten digits and different last two digits, unless differently specified.



3.3 **Tolerance**

This column gives the permitted inspection interval tolerances.

Table 2 List of tolerances

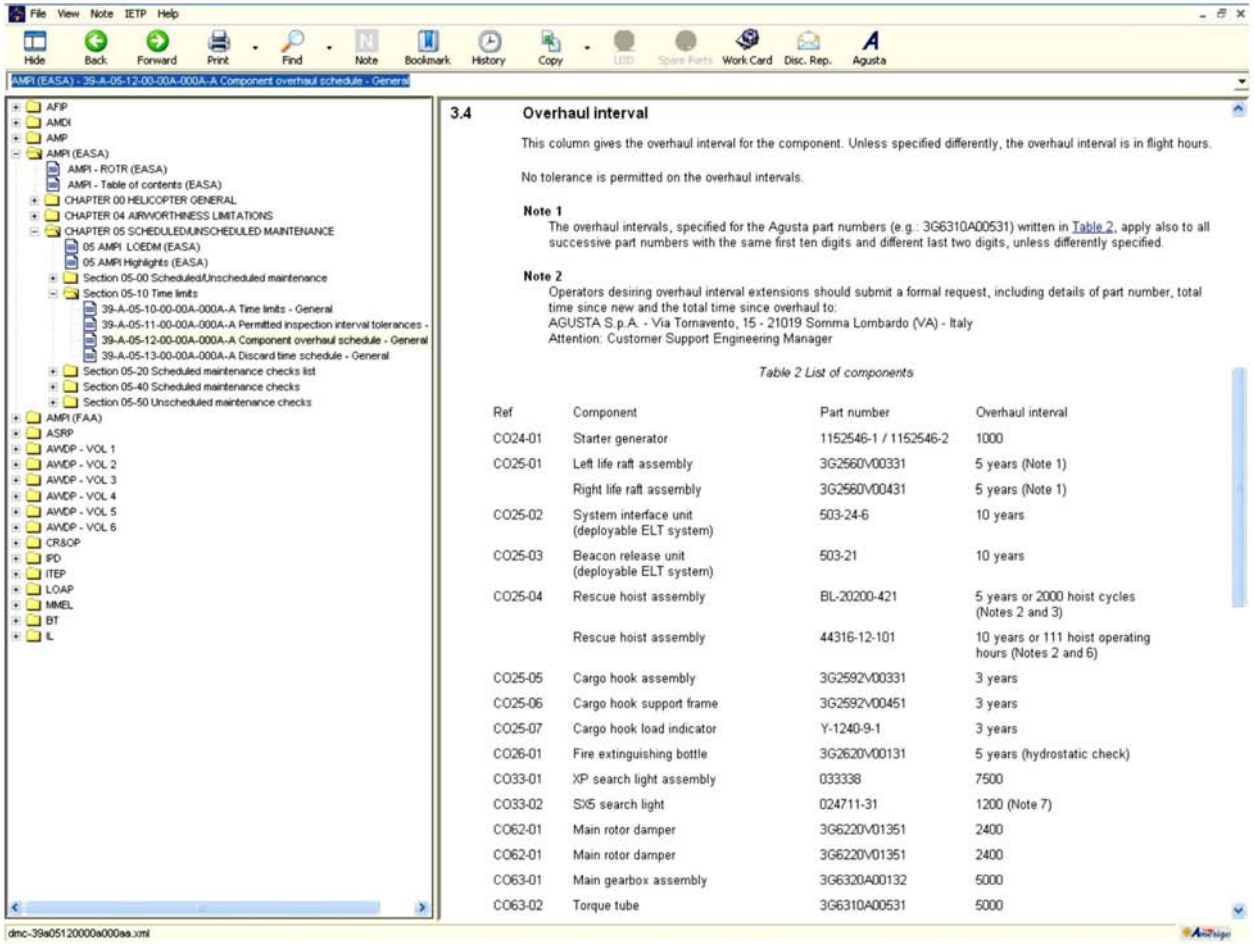
Inspection	Reference	Tolerance
Airworthiness checks	Scheduled maintenance checks list	None
50-hour general visual checks	Scheduled maintenance checks list	5 hours
300 hours	Hourly checks, servicing	30 hours
600 hours	Hourly checks	50 hours
1200 hours	Hourly checks	50 hours
1 year	Calendar checks, servicing	30 days
2 years	Calendar checks, servicing	30 days
4 years	Calendar checks	30 days
6 months	Special inspections	15 days
5 years	Special inspections	30 days
50 hours	Special inspections	5 hours
100 hours	Special inspections	10 hours
150 hours	Special inspections, servicing	15 hours
3000 hours	Special inspections	50 hours
3500 hours	Special inspections	50 hours
5000 hours	Special inspections	50 hours
10000 hours	Special inspections	50 hours
400 hours	Servicing	40 hours
1500 hours	Servicing	50 hours

Note

1 Subsequent intervals will be adjusted to re-establish the original schedule. When an inspection is done earlier than the permitted tolerance, subsequent inspections will be advanced as required to not exceed the maximum tolerance.

2 The following tolerance is established for maintenance scheduling convenience only and must be approved by the governing civil aviation authority. Concurrence and final approval of the inspection interval tolerance by the governing civil aviation authority is the responsibility of the owner/operator.

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3.4 Overhaul interval

This column gives the overhaul interval for the component. Unless specified differently, the overhaul interval is in flight hours.

No tolerance is permitted on the overhaul intervals.

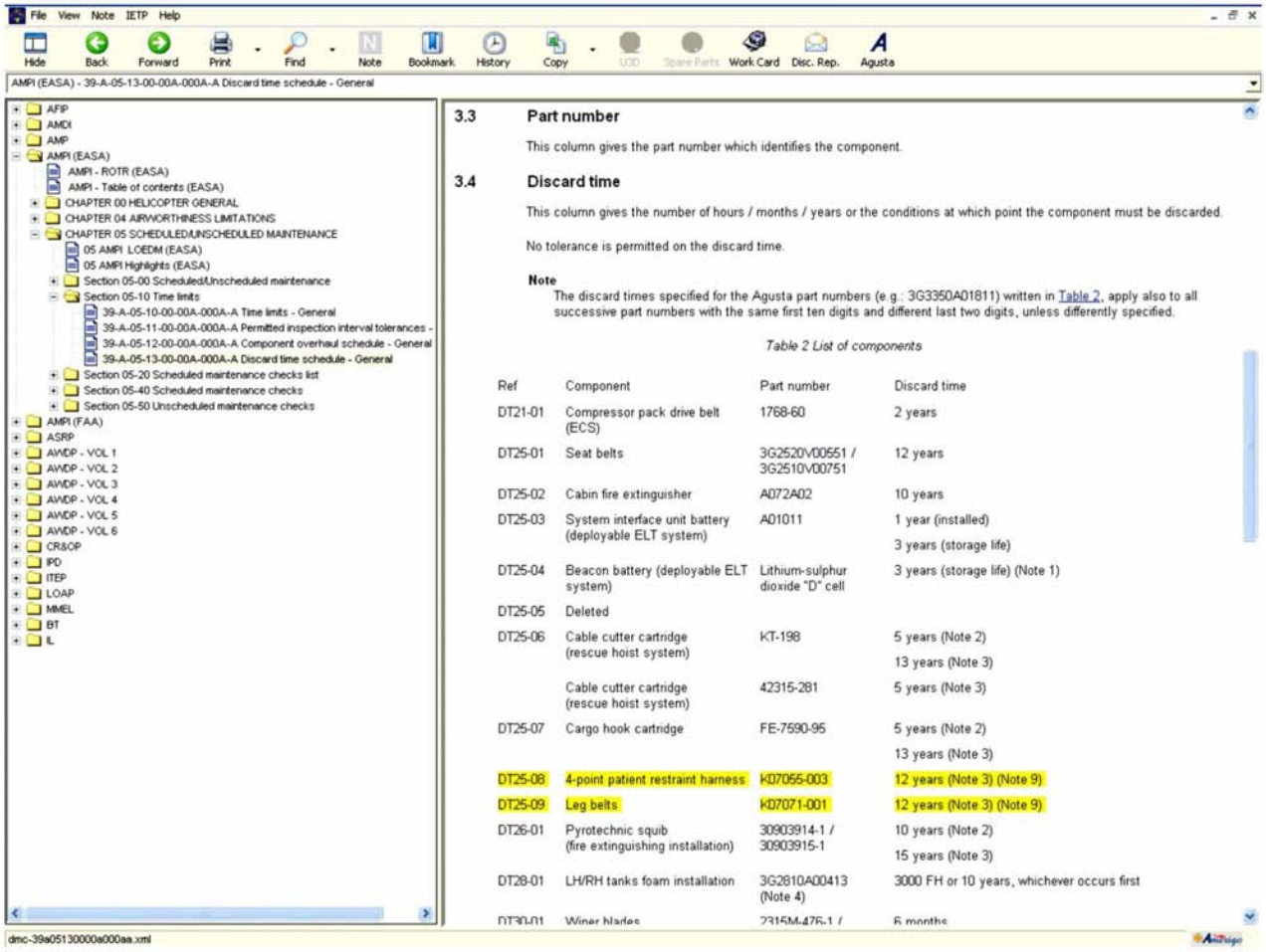
Note 1
The overhaul intervals, specified for the Agusta part numbers (e.g.: 3G6310A00531) written in Table 2, apply also to all successive part numbers with the same first ten digits and different last two digits, unless differently specified.

Note 2
Operators desiring overhaul interval extensions should submit a formal request, including details of part number, total time since new and the total time since overhaul to:
AGUSTA S.p.A. - Via Tornavento, 15 - 21019 Somma Lombardo (VA) - Italy
Attention: Customer Support Engineering Manager

Table 2 List of components

Ref	Component	Part number	Overhaul interval
CO24-01	Starter generator	1152546-1 / 1152546-2	1000
CO25-01	Left life raft assembly	3G2560V00331	5 years (Note 1)
	Right life raft assembly	3G2560V00431	5 years (Note 1)
CO25-02	System interface unit (deployable ELT system)	503-24-6	10 years
CO25-03	Beacon release unit (deployable ELT system)	503-21	10 years
CO25-04	Rescue hoist assembly	BL-20200-421	5 years or 2000 hoist cycles (Notes 2 and 3)
	Rescue hoist assembly	44316-12-101	10 years or 111 hoist operating hours (Notes 2 and 6)
CO25-05	Cargo hook assembly	3G2592V00331	3 years
CO25-06	Cargo hook support frame	3G2592V00451	3 years
CO25-07	Cargo hook load indicator	Y-1240-9-1	3 years
CO26-01	Fire extinguishing bottle	3G2620V00131	5 years (hydrostatic check)
CO33-01	XP search light assembly	033338	7500
CO33-02	SX5 search light	024711-31	1200 (Note 7)
CO62-01	Main rotor damper	3G6220V01351	2400
CO62-01	Main rotor damper	3G6220V01351	2400
CO63-01	Main gearbox assembly	3G6320A00132	5000
CO63-02	Torque tube	3G6310A00531	5000

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3.3 Part number

This column gives the part number which identifies the component.

3.4 Discard time

This column gives the number of hours / months / years or the conditions at which point the component must be discarded.

No tolerance is permitted on the discard time.

Note

The discard times specified for the Agusta part numbers (e.g.: 3G3350A01811) written in Table 2, apply also to all successive part numbers with the same first ten digits and different last two digits, unless differently specified.

Table 2 List of components

Ref	Component	Part number	Discard time
DT21-01	Compressor pack drive belt (ECS)	1768-60	2 years
DT25-01	Seat belts	3G2520V00551 / 3G2510V00751	12 years
DT25-02	Cabin fire extinguisher	A072A02	10 years
DT25-03	System interface unit battery (deployable ELT system)	A01011	1 year (installed) 3 years (storage life)
DT25-04	Beacon battery (deployable ELT system)	Lithium-sulphur dioxide "D" cell	3 years (storage life) (Note 1)
DT25-05	Deleted		
DT25-06	Cable cutter cartridge (rescue hoist system)	KT-198	5 years (Note 2) 13 years (Note 3)
	Cable cutter cartridge (rescue hoist system)	42315-281	5 years (Note 3)
DT25-07	Cargo hook cartridge	FE-7590-95	5 years (Note 2) 13 years (Note 3)
DT25-08	4-point patient restraint harness	KD7055-003	12 years (Note 3) (Note 9)
DT25-09	Leg belts	KD7071-001	12 years (Note 3) (Note 9)
DT26-01	Pyrotechnic squib (fire extinguishing installation)	30903914-1 / 30903915-1	10 years (Note 2) 15 years (Note 3)
DT28-01	LH/RH tanks foam installation	3G2810A00413 (Note 4)	3000 FH or 10 years, whichever occurs first
DT30-01	Wiper blades	2315M-476-1 /	6 months

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05-20 SCHEDULED MAINTENANCE CHECKS LISTS

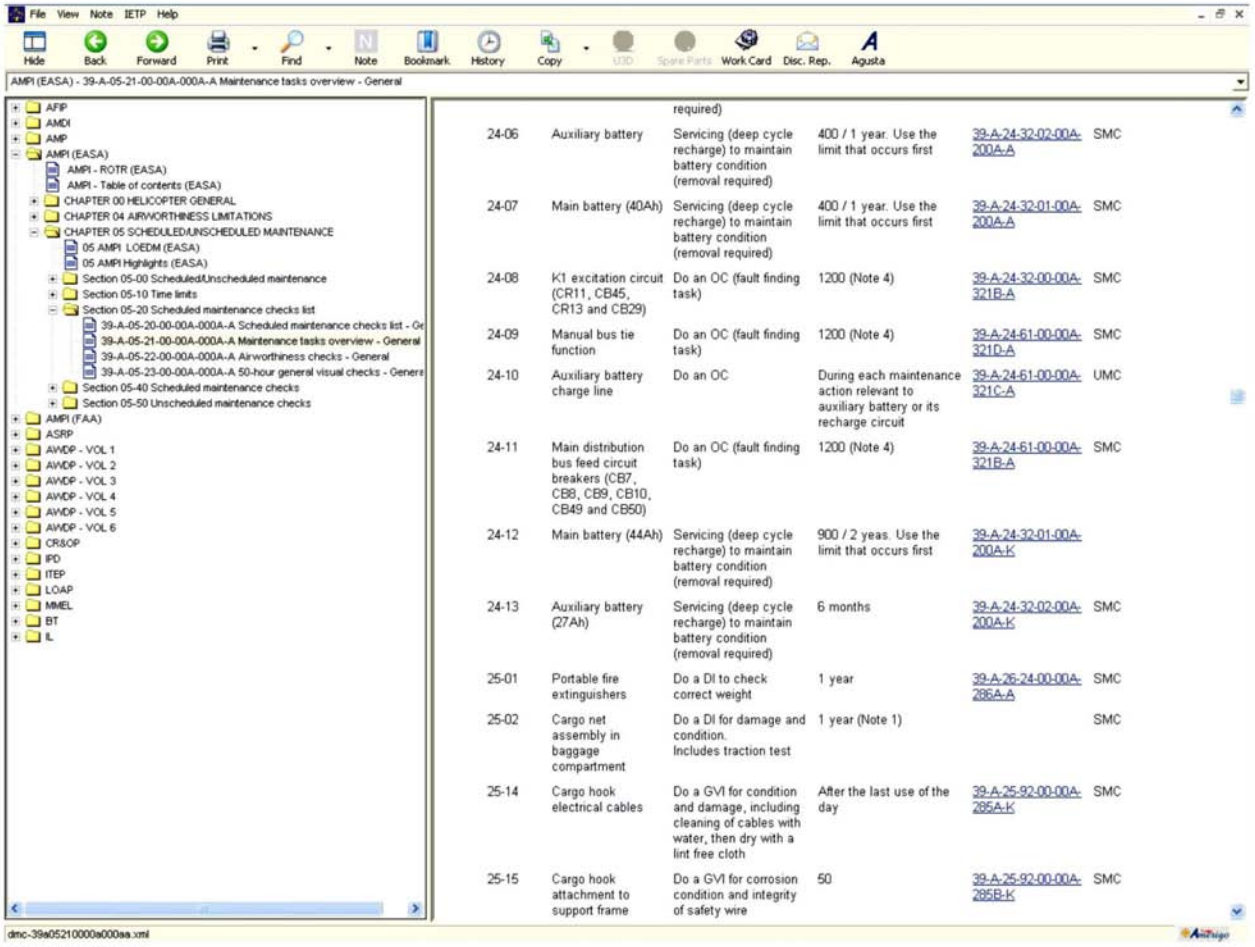
05-21 Maintenance Tasks Overview

This section contains the checks list of the complete scheduled/unscheduled maintenance tasks, to provide a comprehensive view of the preventive maintenance. Inspection can follow hourly of calendar periodicity, or even either of them.
The chapter includes:

- Maintenance tasks overview;
- Airworthiness checks;
- 50 hour general visual checks.

The Maintenance Tasks Overview gives the list of Agusta recommended scheduled and unscheduled maintenance tasks, ordered in sequence of chapter numbers. The task list includes the content of sections:

- Scheduled maintenance checks
- Unscheduled maintenance checks



		required)			
24-06	Auxiliary battery	Servicing (deep cycle recharge) to maintain battery condition (removal required)	400 / 1 year. Use the limit that occurs first	39-A-24-32-02-00A-200A-A	SMC
24-07	Main battery (40Ah)	Servicing (deep cycle recharge) to maintain battery condition (removal required)	400 / 1 year. Use the limit that occurs first	39-A-24-32-01-00A-200A-A	SMC
24-08	K1 excitation circuit (CR11, CB45, CR13 and CB29)	Do an OC (fault finding task)	1200 (Note 4)	39-A-24-32-00-00A-321B-A	SMC
24-09	Manual bus tie function	Do an OC (fault finding task)	1200 (Note 4)	39-A-24-61-00-00A-321D-A	SMC
24-10	Auxiliary battery charge line	Do an OC	During each maintenance action relevant to auxiliary battery or its recharge circuit	39-A-24-61-00-00A-321C-A	UMC
24-11	Main distribution bus feed circuit breakers (CB7, CB8, CB9, CB10, CB49 and CB50)	Do an OC (fault finding task)	1200 (Note 4)	39-A-24-61-00-00A-321B-A	SMC
24-12	Main battery (44Ah)	Servicing (deep cycle recharge) to maintain battery condition (removal required)	900 / 2 yeas. Use the limit that occurs first	39-A-24-32-01-00A-200A-K	
24-13	Auxiliary battery (27Ah)	Servicing (deep cycle recharge) to maintain battery condition (removal required)	6 months	39-A-24-32-02-00A-200A-K	SMC
25-01	Portable fire extinguishers	Do a DI to check correct weight	1 year	39-A-26-24-00-00A-286A-A	SMC
25-02	Cargo net assembly in baggage compartment	Do a DI for damage and condition. Includes traction test	1 year (Note 1)		SMC
25-14	Cargo hook electrical cables	Do a GVI for condition and damage, including cleaning of cables with water, then dry with a lint free cloth	After the last use of the day	39-A-25-92-00-00A-285A-K	SMC
25-15	Cargo hook attachment to support frame	Do a GVI for corrosion condition and integrity of safety wire	50	39-A-25-92-00-00A-285B-K	SMC

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05-22 Airworthiness Checks

The AW139 maintenance program has been developed without a daily inspection.
However, there are special cases in which the Airworthiness Authorities require to the Maintenance Organization the accomplishment of a comprehensive turn-around inspection to be performed with daily periodicity, by qualified maintenance personnel.
The following paragraphs report the summary of requirements necessary to accomplish an Airworthiness Check, only in the cases in which it is required by the Authorities.

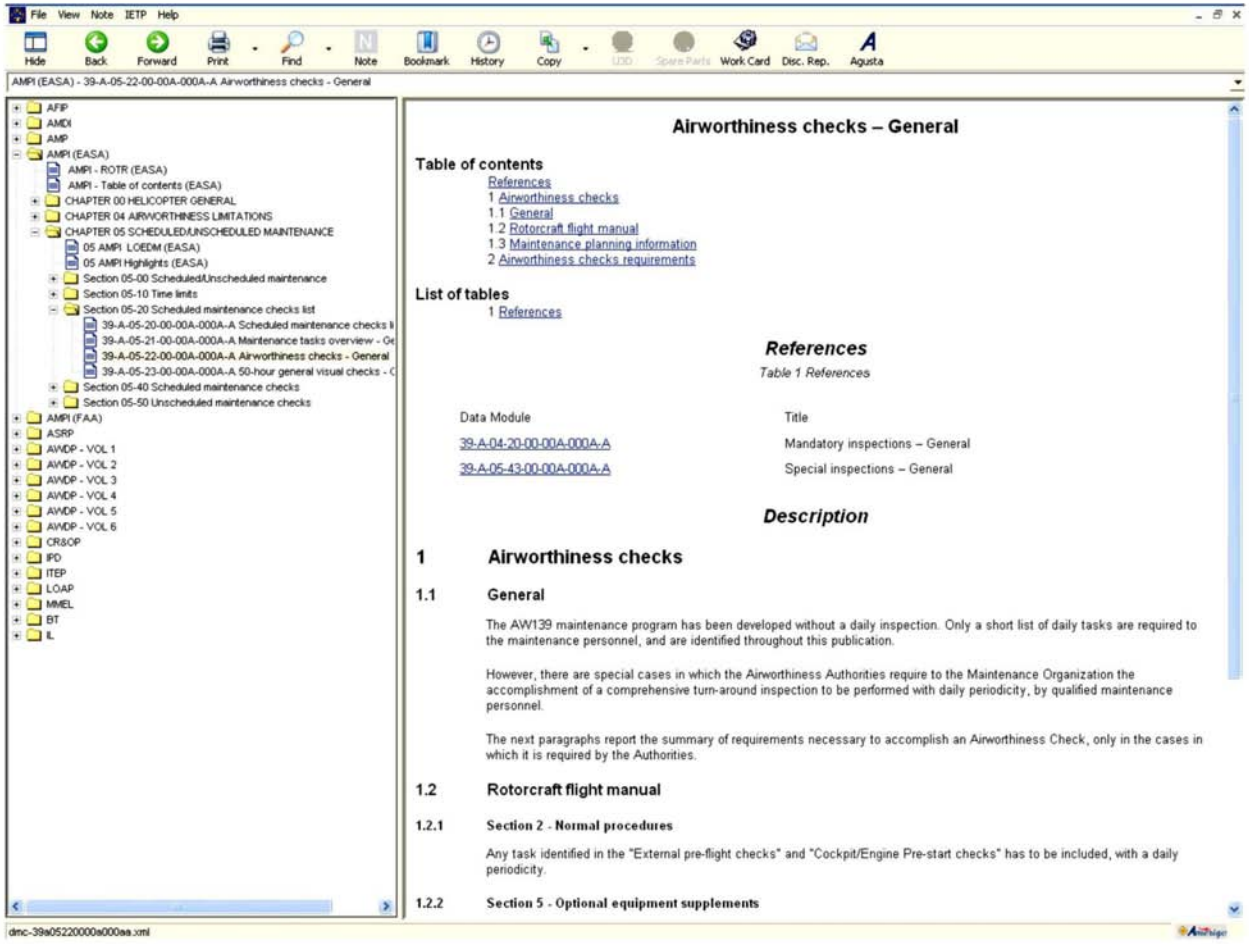
- RFM – Section 2 NORMAL PROCEDURE “External pre-flight checks” and “Cockpit/Engine Pre-start checks”;
-

NOTE C

Checks marked with a large black arrow are required only before the first flight of the day.

- RFM – Section 5 OPTIONAL EQUIPMENT: any section 2 NORMAL PROCEDURE with regard to “External pre-flight checks” and “Cockpit/ Engine Pre-start Checks, if the relevant optional equipment is installed;
- AMPI – Chap. 04 and 05: Any daily tasks identified in the mandatory inspection list and special inspection task list;
- Any daily tasks identified in the mandatory inspection of the Engine Technical Publication;
- Any daily tasks identified in the mandatory inspection of any optional equipment installed.
- Any daily tasks identified by any Airworthiness Directive and/or Bollettini Tecnici if applicable.

The Airworthiness Check is an inspection that has to be performed by qualified maintenance personnel.
It’s expire after 72 hours from the end of the inspection if the helicopter has not flown and does not replace the Rotorcraft Flight Manual requirements list which must be performed by a Pilot.



Airworthiness checks – General

Table of contents

References
1 Airworthiness checks
1.1 General
1.2 Rotorcraft flight manual
1.3 Maintenance planning information
2 Airworthiness checks requirements

List of tables

1 References

References

Table 1 References

Data Module	Title
39-A-04-20-00-00A-000A-A	Mandatory inspections – General
39-A-05-43-00-00A-000A-A	Special inspections – General

Description

1 Airworthiness checks

1.1 General

The AW139 maintenance program has been developed without a daily inspection. Only a short list of daily tasks are required to the maintenance personnel, and are identified throughout this publication.

However, there are special cases in which the Airworthiness Authorities require to the Maintenance Organization the accomplishment of a comprehensive turn-around inspection to be performed with daily periodicity, by qualified maintenance personnel.

The next paragraphs report the summary of requirements necessary to accomplish an Airworthiness Check, only in the cases in which it is required by the Authorities.

1.2 Rotorcraft flight manual

1.2.1 Section 2 - Normal procedures

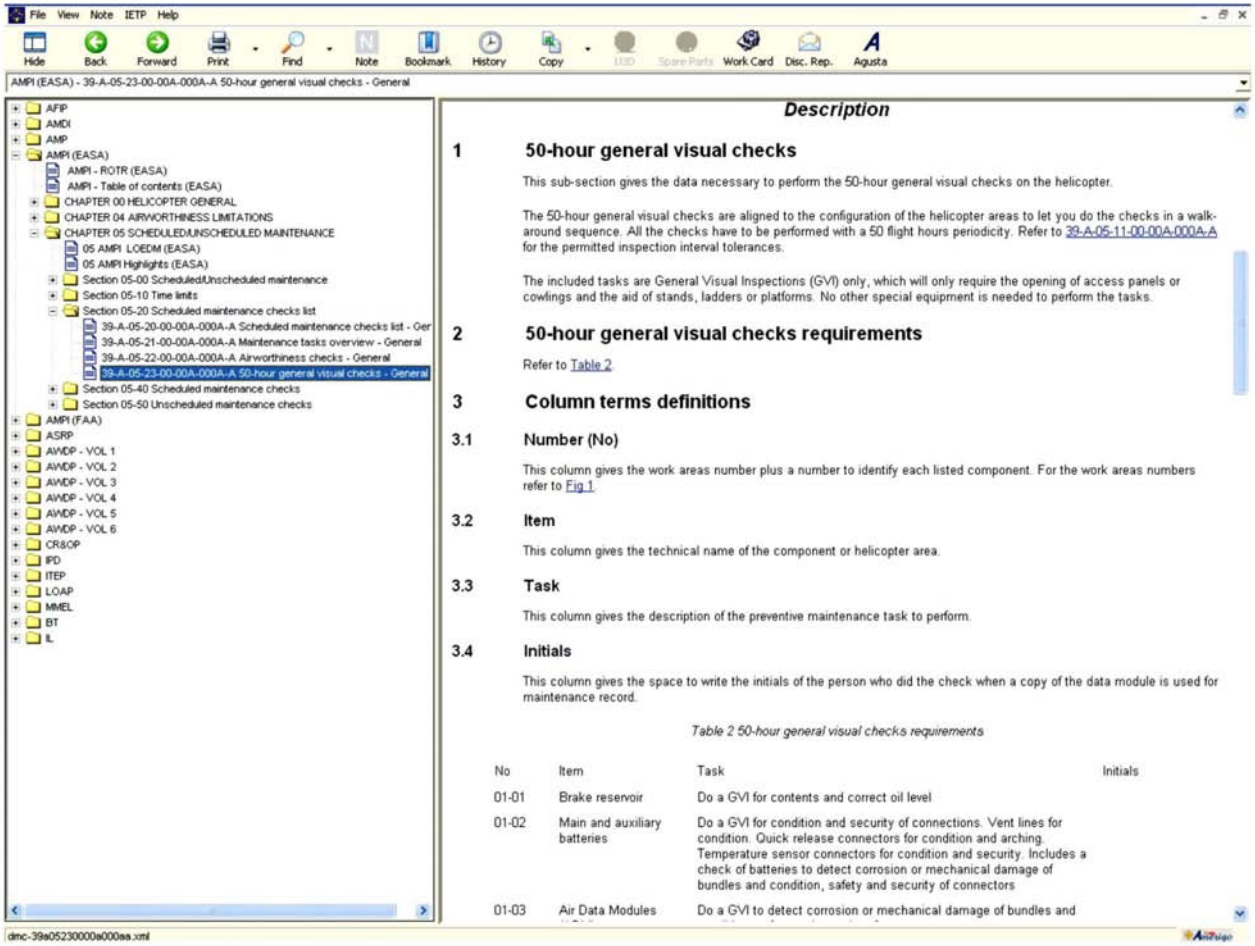
Any task identified in the "External pre-flight checks" and "Cockpit/Engine Pre-start checks" has to be included, with a daily periodicity.

1.2.2 Section 5 - Optional equipment supplements

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05-23 50 Hour General Visual Check

The 50 – hour general visual checks are aligned to the configuration of the helicopter areas to let the operator to do the checks in a walk-around sequence. All the checks have to be performed with a 50 flight hours periodicity. The tasks are General Visual Inspections (GVI) only, which will only require the opening of access panels or cowlings and the aid of stands, ladders or platforms. The permitted inspection interval tolerance is applicable.



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05-40 SCHEDULED MAINTENANCE CHECKS

This section gives the data necessary to perform the scheduled maintenance checks on the AW139 and is divided into:

- Hourly checks
- Calendar checks
- Special inspections
- Servicing

The hourly checks sub-section list all the maintenance tasks that must be performed at the same hour interval. They are divided into:

- 300 hours checks
- 600 hours checks
- 1200 hours checks

The calendar check sub section list all the maintenance tasks that must be performed at the same calendar interval. They are divided into:

- 1 year checks
- 2 year checks
- 4 year checks

The special inspection sub-section gives the data necessary to perform the maintenance checks which can not be fitted into the standard maintenance program, even if they have a scheduled periodicity.
The servicing sub-section contains the servicing intervals for items applicable to the AW139.

AW139



ROTORCRAFT FLIGHT MANUAL

Copy assigned to rotorcraft:
S/N . . . . . .
Registration Marks . . . . . .

ISSUE :
Rev. :

AW139 and AB139 are two names for the same product.
They identify two batches of aircraft manufactured in conformity with a unique Type Certificate Data Sheet
- AB139 up to SN 31054;
- AW139 from SN 31055 onward.
Where not specifically declared, the content of this document is applicable to both AW139 and AB139 helicopters.

Continuing airworthiness criteria for the AW139 is developed and maintained by Agusta, who is the holder of the type certificate in the state of design.

E.A.S.A. Approval Date:
-

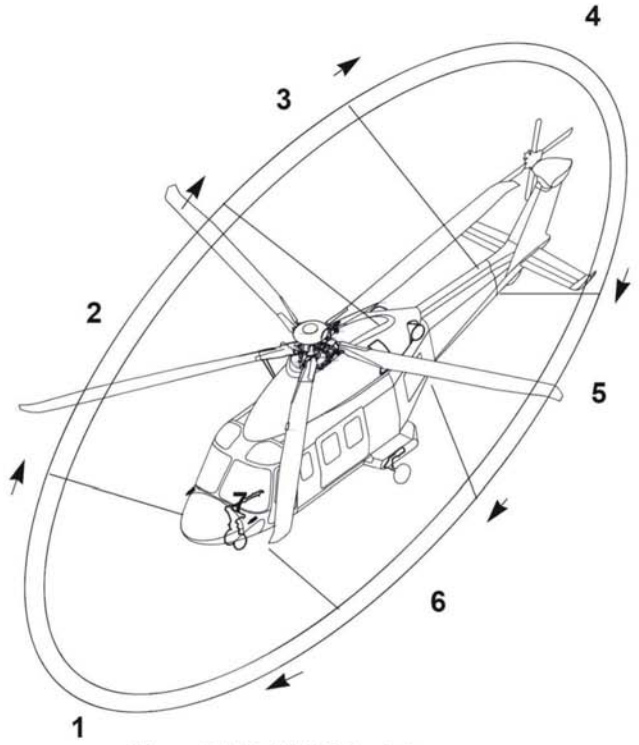
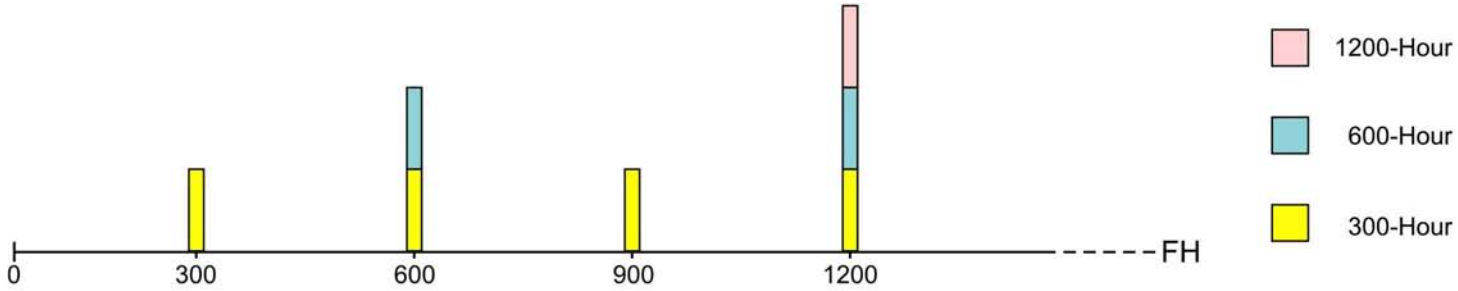


Figure 2-1 Preflight Check Sequence

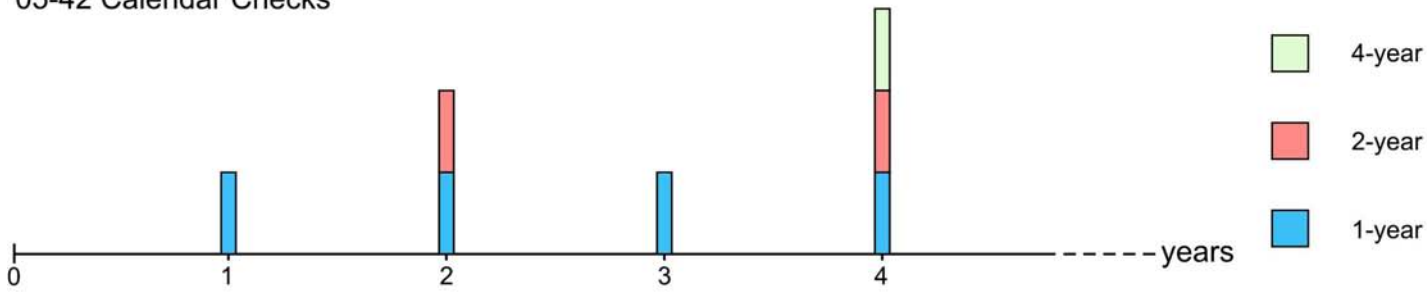
AREA N°1 : Helicopter nose
AREA N°2 : Fuselage - RH side
AREA N°3 : Tail boom - RH side
AREA N°4 : Fin, intermediate/tail gearbox, tail rotor
AREA N°5 : Tail boom LH side
AREA N°6 : Fuselage - LH side
AREA N°7 : Cabin and Cockpit interior

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05-41 Hourly Checks



05-42 Calendar Checks



05-43 Special Inspections
Maintenance checks which can not be fitted into the standard maintenance program even if they have a scheduled periodicity

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05-50 UNSCHEDULED INSPECTION

05-51 Out Of Phase Maintenance

Maintenance checks which are opportune at the occurrence of another maintenance task, or which cannot be included in an hourly/calendar schedule.

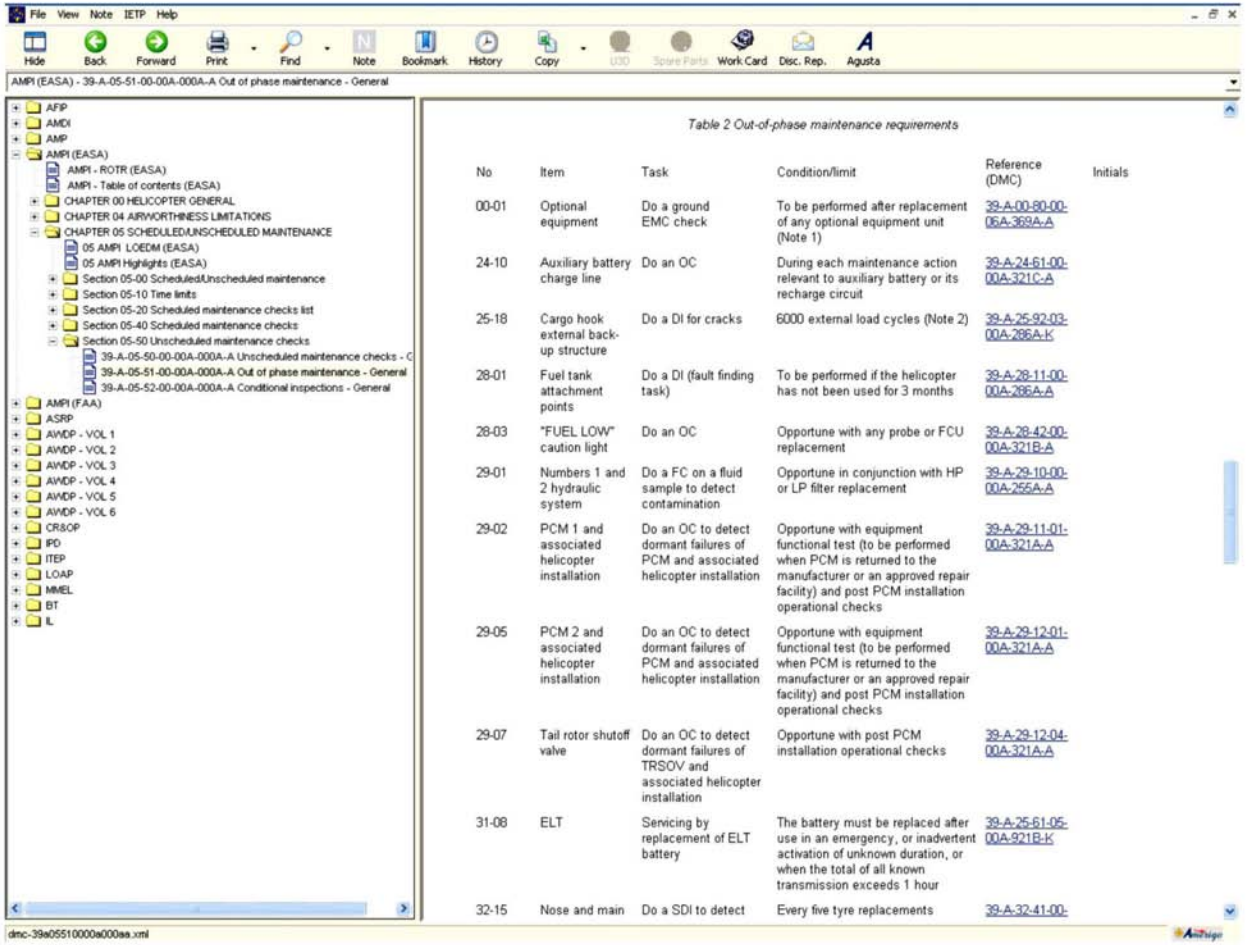


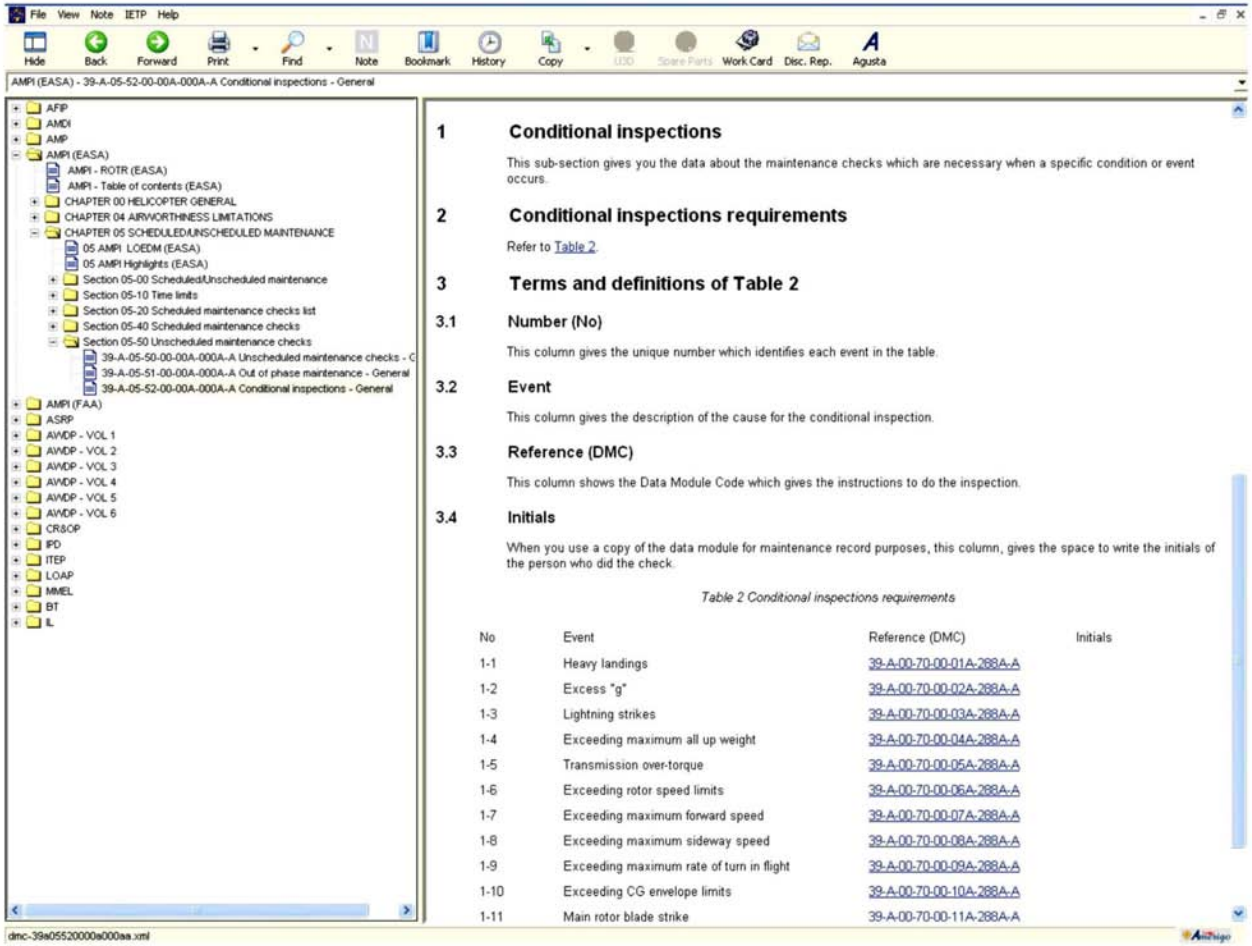
Table 2 Out-of-phase maintenance requirements

No	Item	Task	Condition/limit	Reference (DMC)	Initials
00-01	Optional equipment	Do a ground EMC check	To be performed after replacement of any optional equipment unit (Note 1)	39-A-00-80-00-00A-369A-A	
24-10	Auxiliary battery charge line	Do an OC	During each maintenance action relevant to auxiliary battery or its recharge circuit	39-A-24-61-00-00A-321C-A	
25-18	Cargo hook external back-up structure	Do a DI for cracks	6000 external load cycles (Note 2)	39-A-25-92-03-00A-286A-K	
28-01	Fuel tank attachment points	Do a DI (fault finding task)	To be performed if the helicopter has not been used for 3 months	39-A-28-11-00-00A-286A-A	
28-03	"FUEL LOW" caution light	Do an OC	Opportune with any probe or FCU replacement	39-A-28-42-00-00A-321B-A	
29-01	Numbers 1 and 2 hydraulic system	Do a FC on a fluid sample to detect contamination	Opportune in conjunction with HP or LP filter replacement	39-A-29-10-00-00A-265A-A	
29-02	PCM 1 and associated helicopter installation	Do an OC to detect dormant failures of PCM and associated helicopter installation	Opportune with equipment functional test (to be performed when PCM is returned to the manufacturer or an approved repair facility) and post PCM installation operational checks	39-A-29-11-01-00A-321A-A	
29-05	PCM 2 and associated helicopter installation	Do an OC to detect dormant failures of PCM and associated helicopter installation	Opportune with equipment functional test (to be performed when PCM is returned to the manufacturer or an approved repair facility) and post PCM installation operational checks	39-A-29-12-01-00A-321A-A	
29-07	Tail rotor shutoff valve	Do an OC to detect dormant failures of TRSOV and associated helicopter installation	Opportune with post PCM installation operational checks	39-A-29-12-04-00A-321A-A	
31-08	ELT	Servicing by replacement of ELT battery	The battery must be replaced after use in an emergency, or inadvertent activation of unknown duration, or when the total of all known transmission exceeds 1 hour	39-A-25-61-05-00A-921B-K	
32-15	Nose and main	Do a SDI to detect	Every five tyre replacements	39-A-32-41-00-	

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05-52 Conditional Inspections

Maintenance checks which are necessary when a specific condition or event occurs.

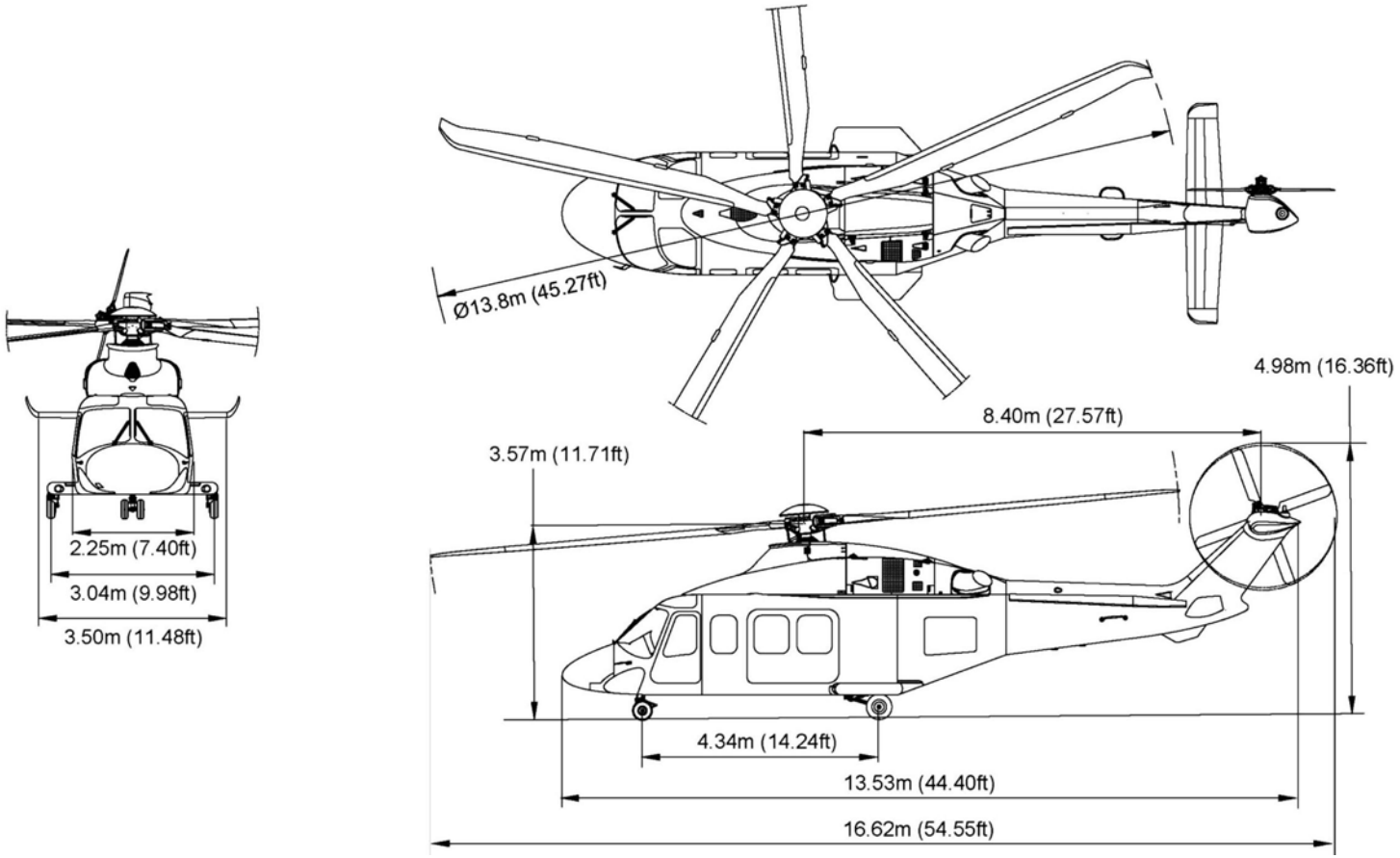


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CHAPTER
06
DIMENSIONS AND AREAS

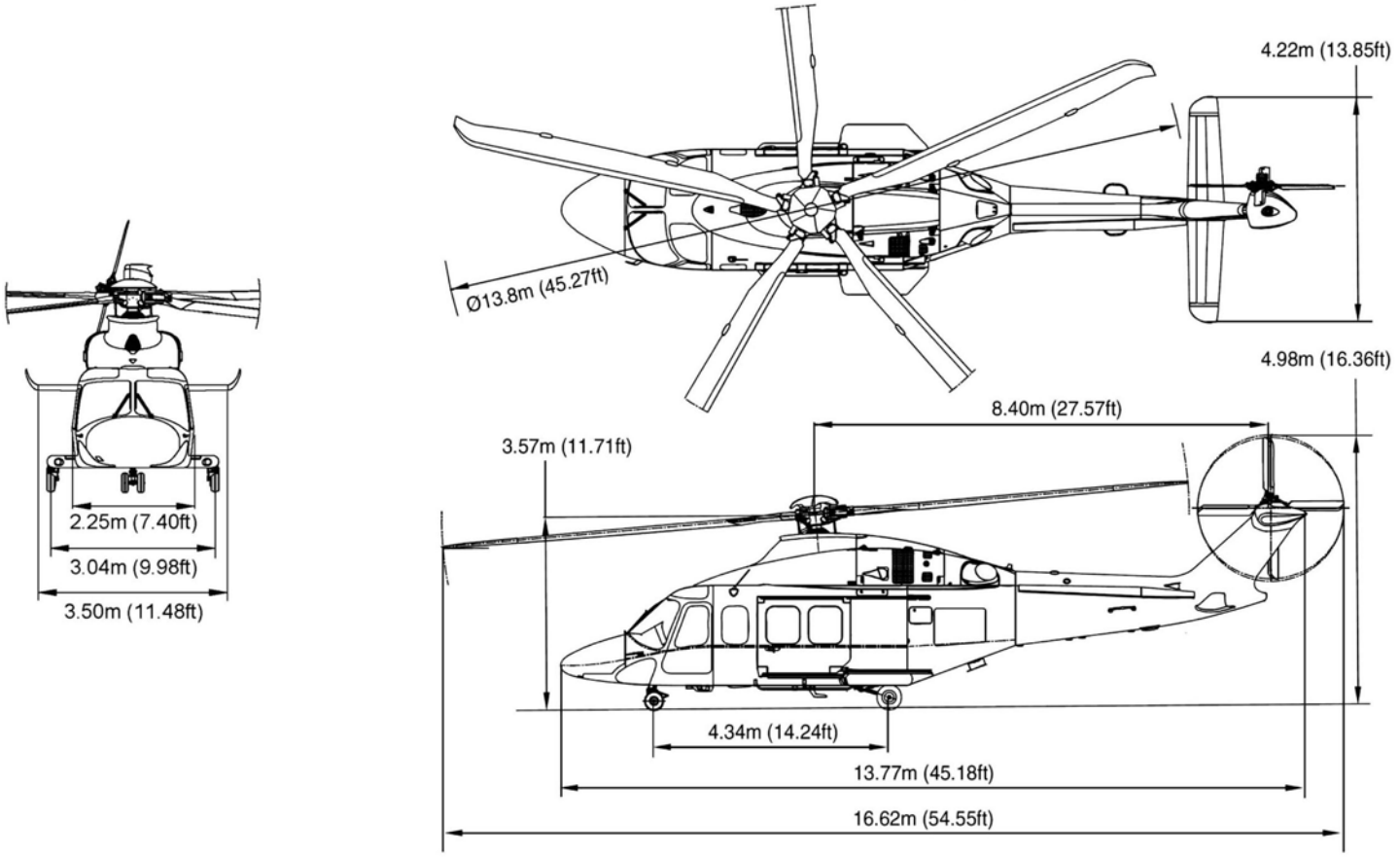
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SECTION 10 - PRINCIPAL DIMENSIONS



ICN-39-A-000000-G-A0126-01594-A-01-1

PRINCIPAL DIMENSIONS



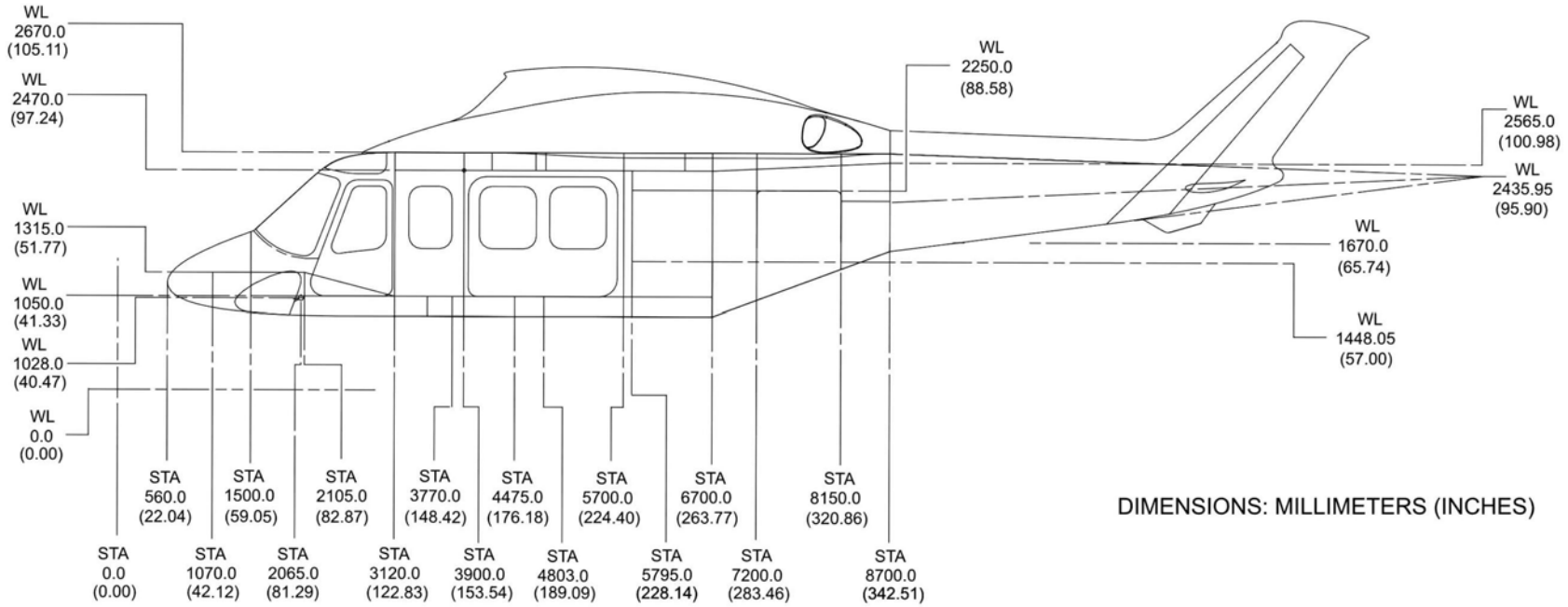
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AW139 (LONG NOSE)-PRINCIPAL DIMENSIONS

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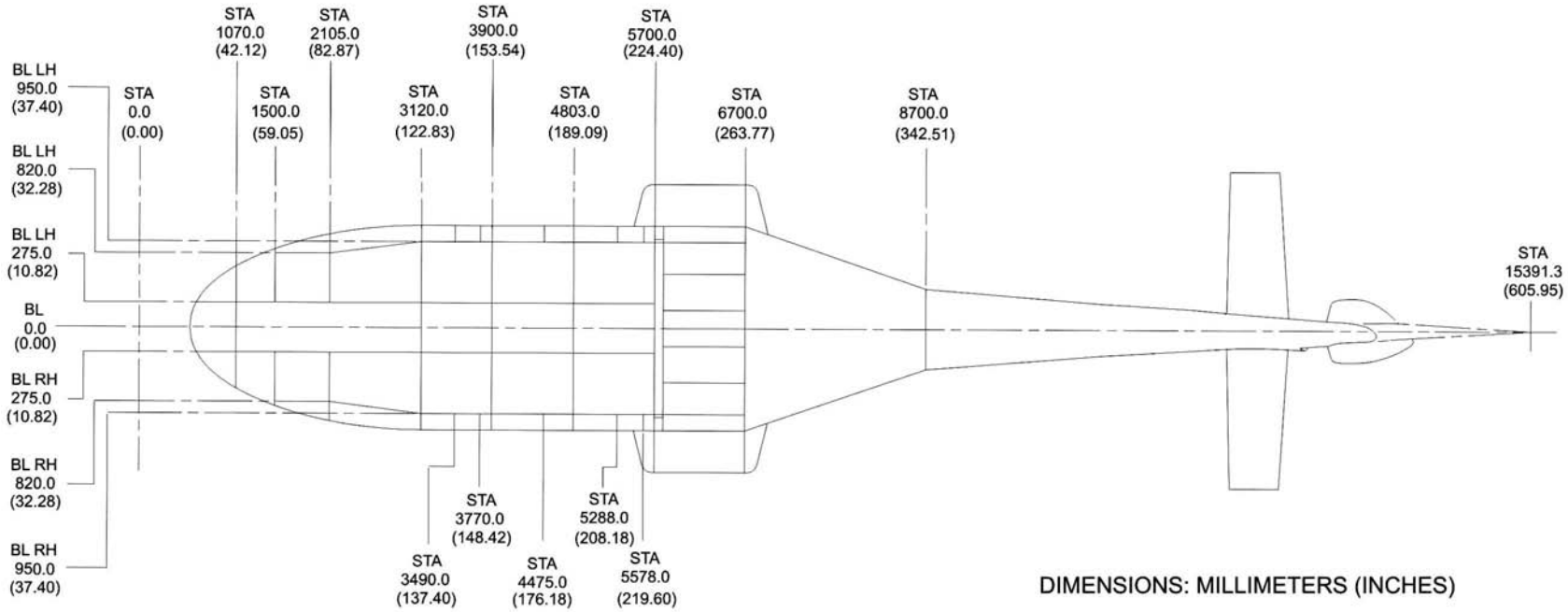
SECTION 20 - REFERENCE LINES

BL = Buttock Line (LH:Left - RH:Right); WL = Water Line;
STA = Station Line



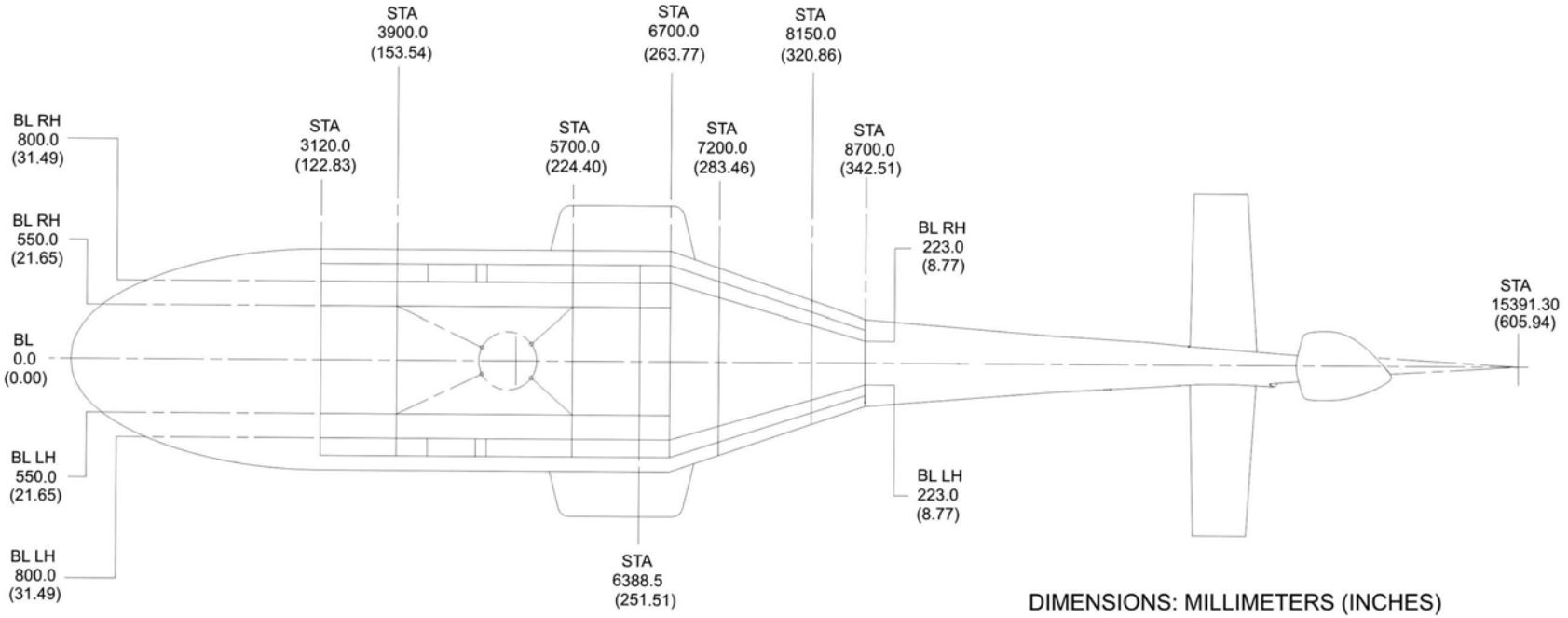
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REFERENCE LINES (LATERAL VIEW)



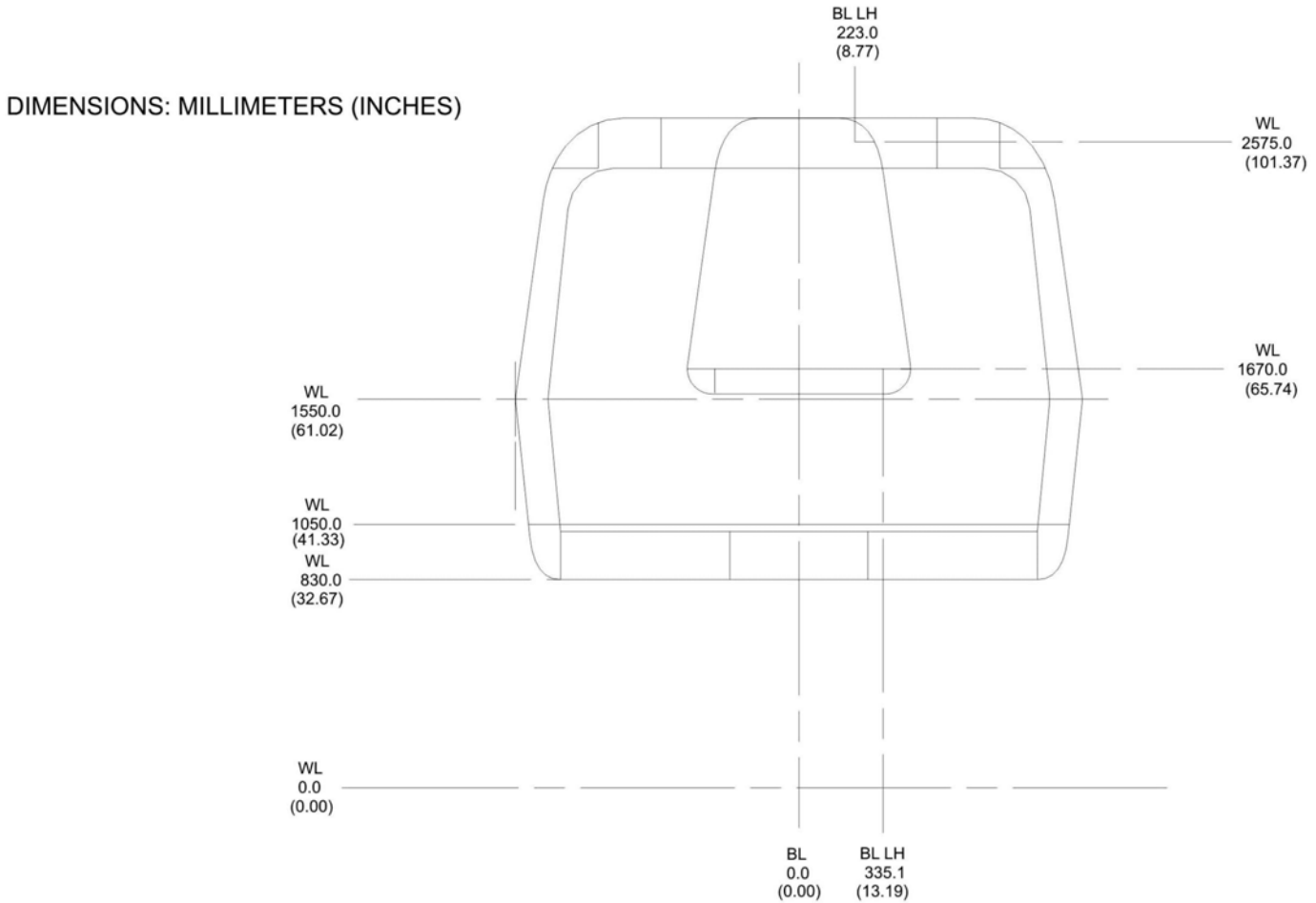
ICN-39-A-062000-G-A0126-00495-A-01-1

REFERENCE LINES (VIEW FROM THE TOP)



ICN-39-A-062000-G-A0126-00496-A-01-1

REFERENCE LINES (VIEW FROM THE BOTTOM)



ICN-39-A-062000-G-A0126-00497-A-01-1

CROSS SECTION FROM STA 3120.0 (122.83) TO STA 6700.0 (263.77)

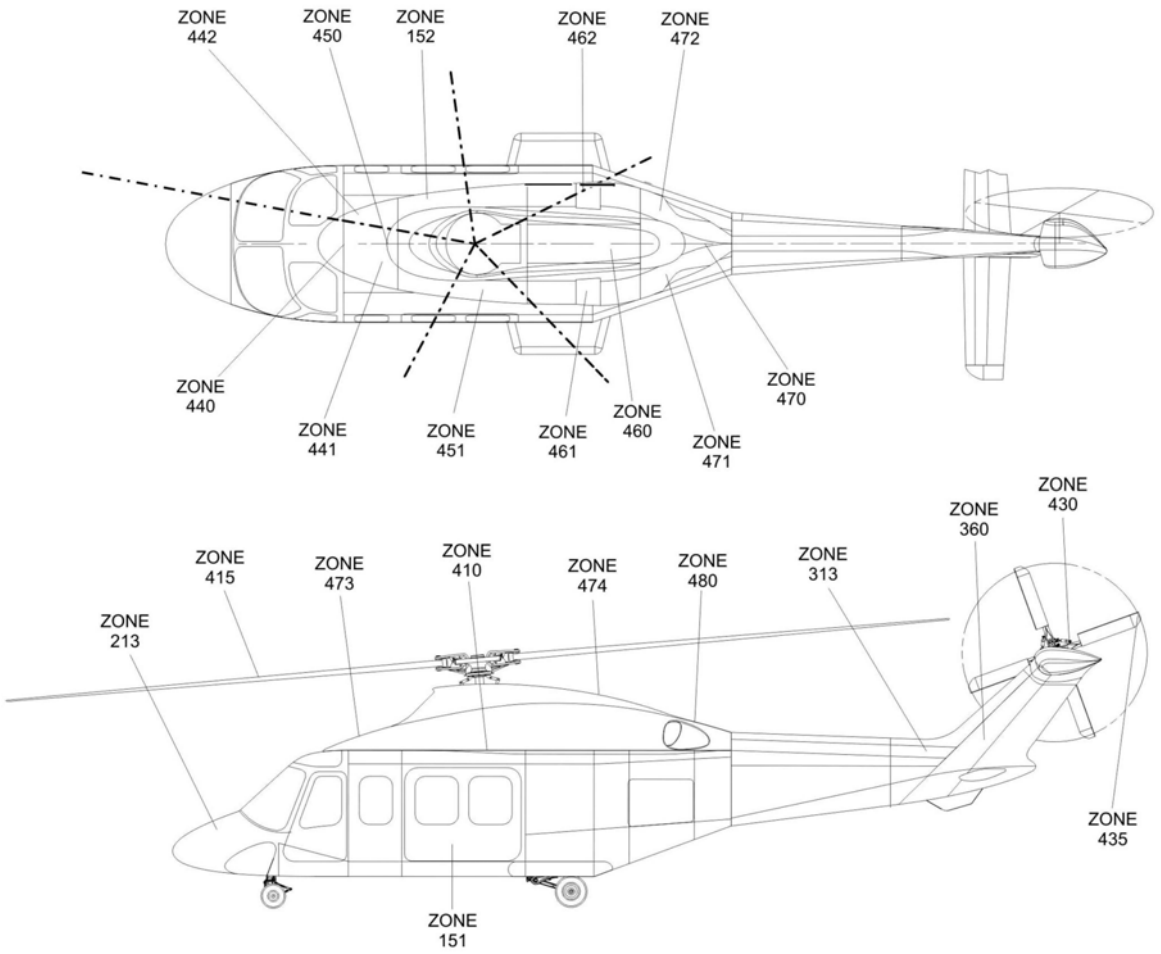
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SECTION 40 - ACCESS PROVISIONS

GENERAL DESCRIPTION

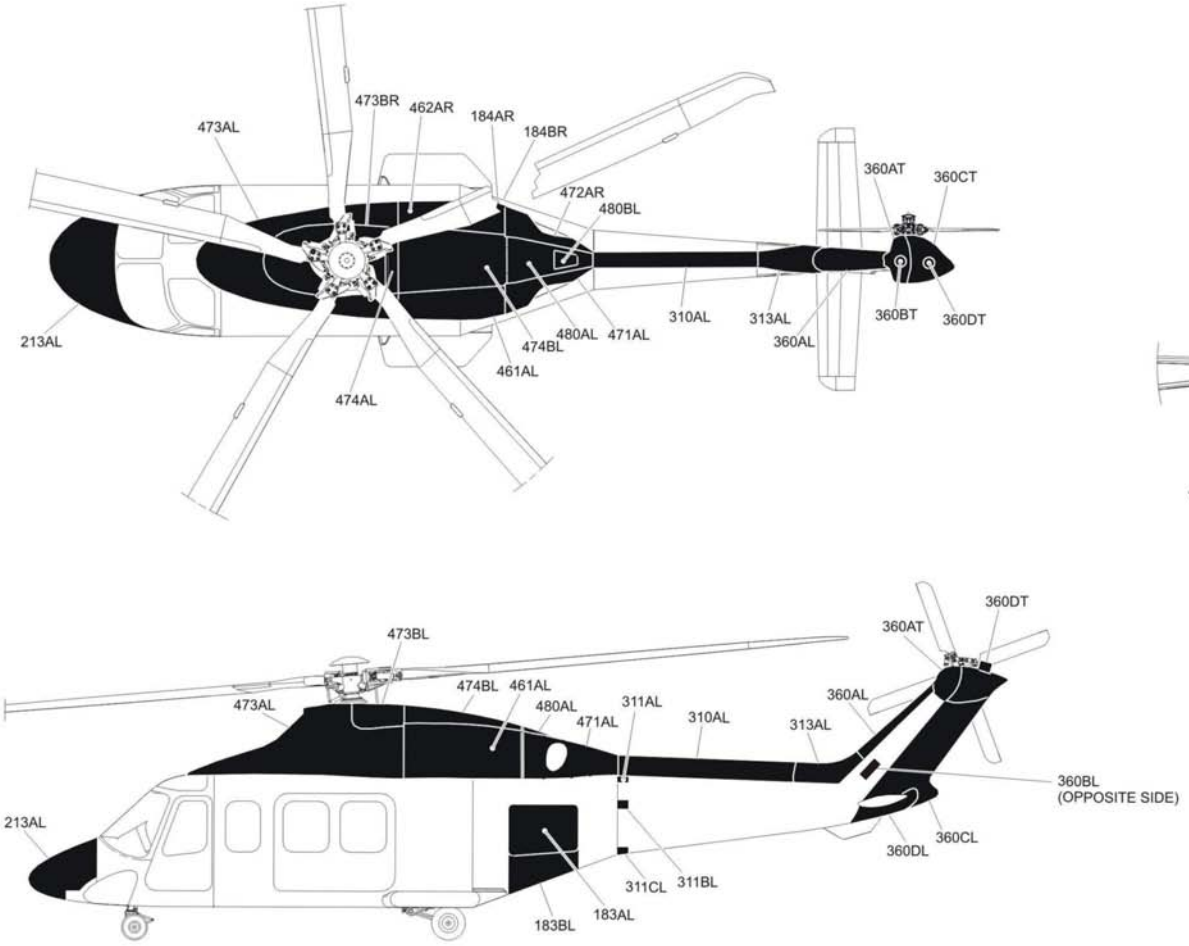
The panels and the doors are identified by the following code (111 AL):

- 111..... Number of the smallest zonal area that the panel is in
- A..... Primary identifier = identifies the panel in an alphabetical sequence (front to rear, inboard to outboard, top to bottom)
- L..... Locator = gives the location of the panel in relation to its position in the helicopter (Bottom, Left, Right, Top, Z = internal)



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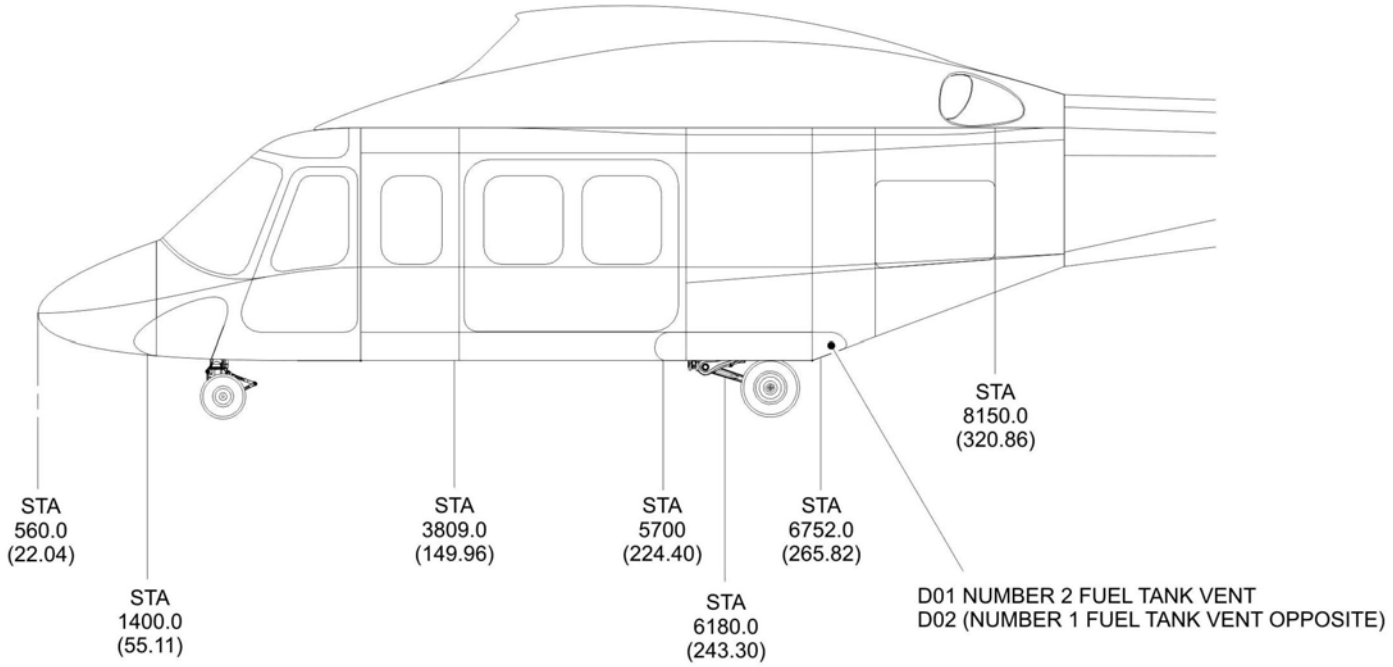
SMALLEST AREA





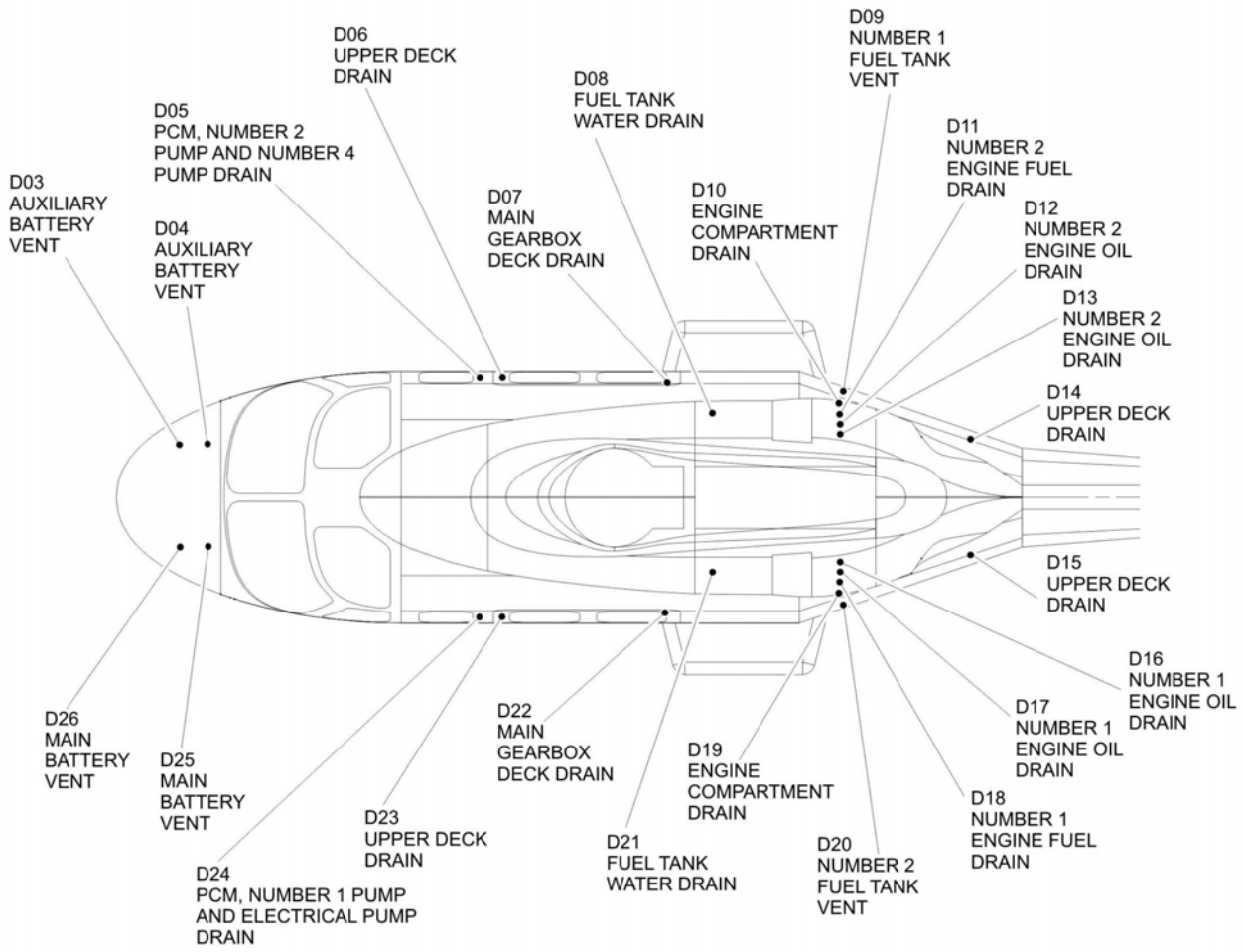
ICN-39-A-064000-G-A0126-00499-A-01-1

ACCESS PROVISIONS



ICN-39-A-064000-G-A0126-00500-A-01-1

DRAIN HOLES (SHEET 1 OF 2)



ICN-39-A-064000-G-A0126-00501-A-01-1

DRAIN HOLES (SHEET 2 OF 2)

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CHAPTER 07

LIFTING, SHORING, RECOVERING AND TRANSPORTING

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SECTION 10 - JACKING

MAINTENANCE INFORMATION

CAUTION

IN THIS PROCEDURE TO LIFT ON JACK/LOWER THE HELICOPTER ARE NECESSARY FOUR PERSONS. THIS IS TO OPERATE THE JACKS AT THE SAME TIME.

CAUTION

IF THE HELICOPTER HAS A WEATHER RADAR SYSTEM:

- OPEN THE WXR CIRCUIT BREAKER (WEATHER RADAR PRIMUS 660) OR THE WXR AND WXR CONTR CIRCUIT BREAKERS (WEATHER RADAR PRIMUS 701).
  SAFETY THE CIRCUIT BREAKER(S) WITH AN APPLICABLE TAG(S).

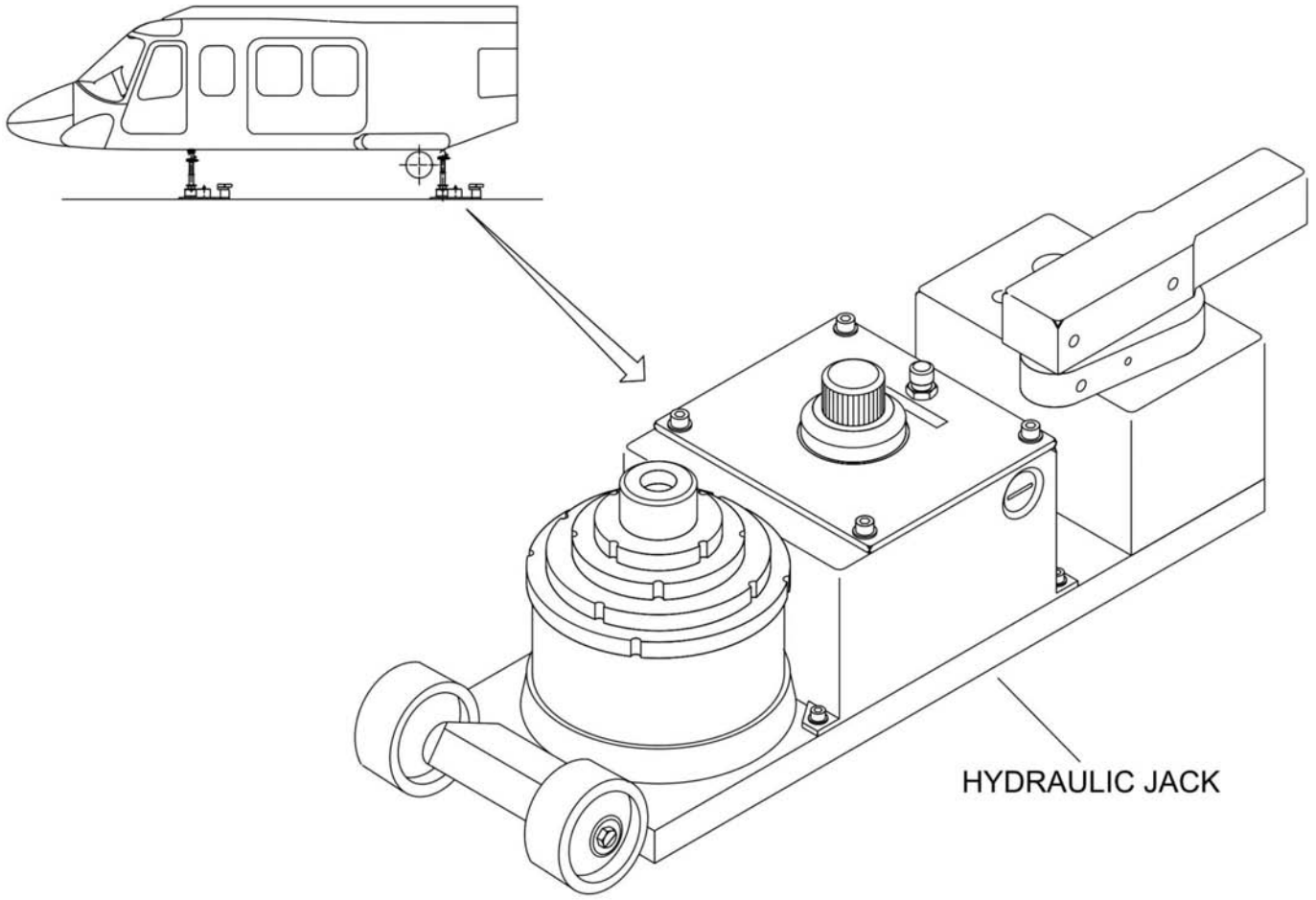
- Before jacking the helicopter must be on a level ground, if not put bases below jacks.
- Jacks must be operated at the same time.
- Before lowering the helicopter, by operating slowly the release valve knob, make sure that the landing gears are fully extended and locked

SAFETY PRECAUTIONS

- Before careful when lifting or lowering the helicopter. Make sure all the persons who are not necessary for the procedures are clear of the helicopter.
  Do not stay near or on the helicopter.
- Put ropes around the work area and put the jack warning signs at the four end sides of the helicopter.

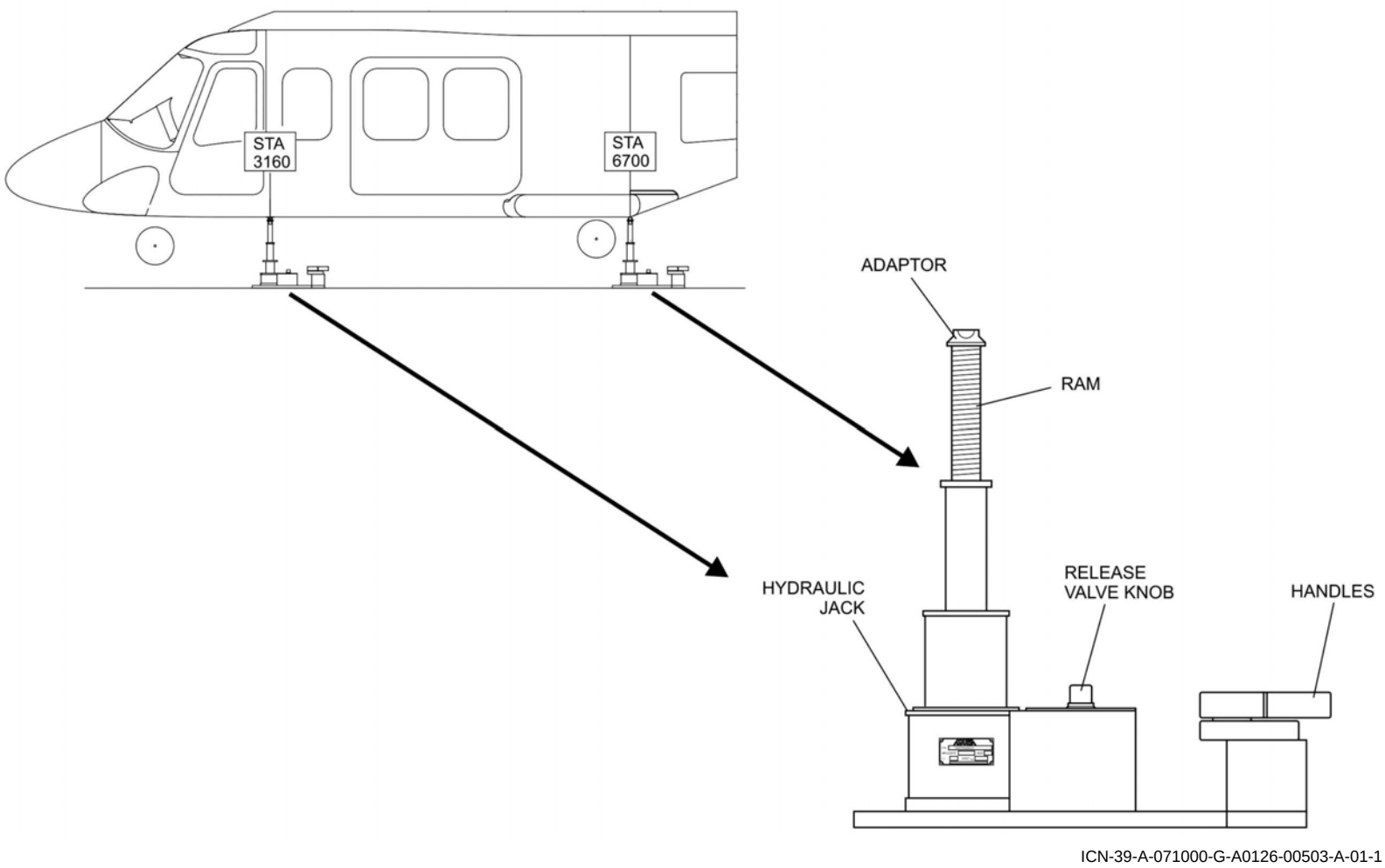
SPECIAL TOOLS/TEST EQUIPMENT

- Hydraulic jacks.



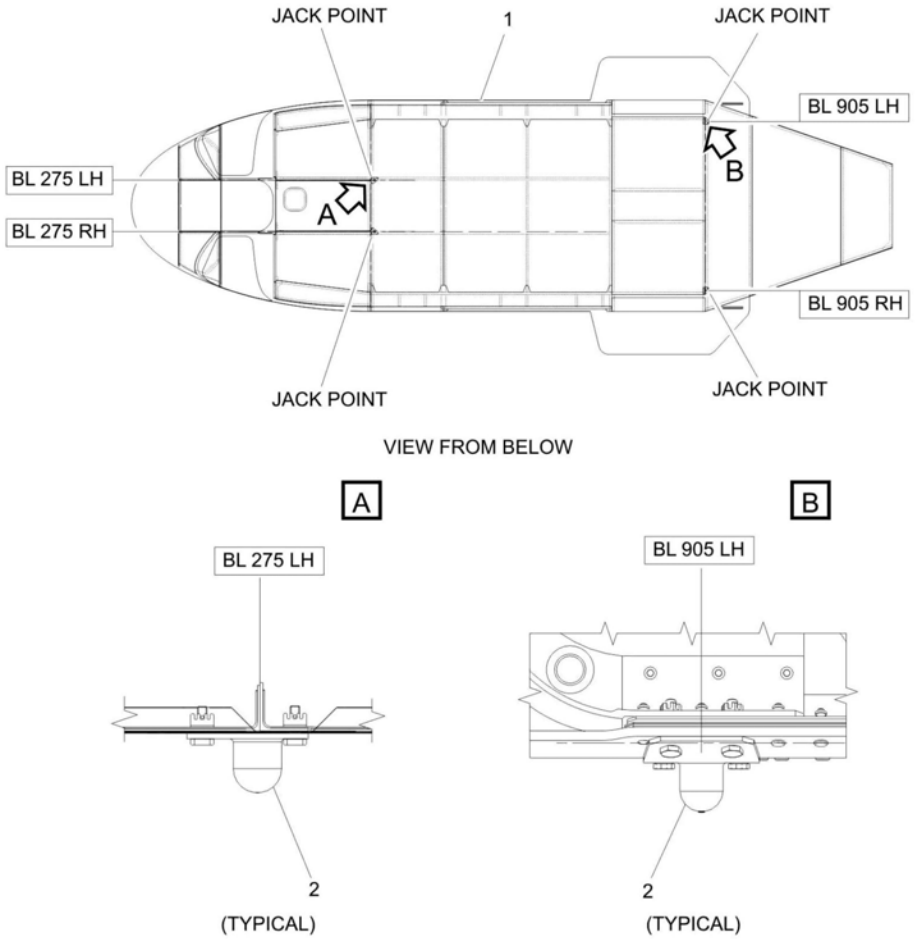
ICN-39-A-071000-G-A0126-00502-A-01-1

SPECIAL TOOLS/TEST EQUIPMENT



ICN-39-A-071000-G-A0126-00503-A-01-1

HELICOPTER JACKING (SHEET 1 OF 2)



ICN-39-A-071000-G-A0126-00504-A-01-1

HELICOPTER JACKING (SHEET 2 OF 2)

SECTION 30 - SLINGING

MAINTENANCE INFORMATION

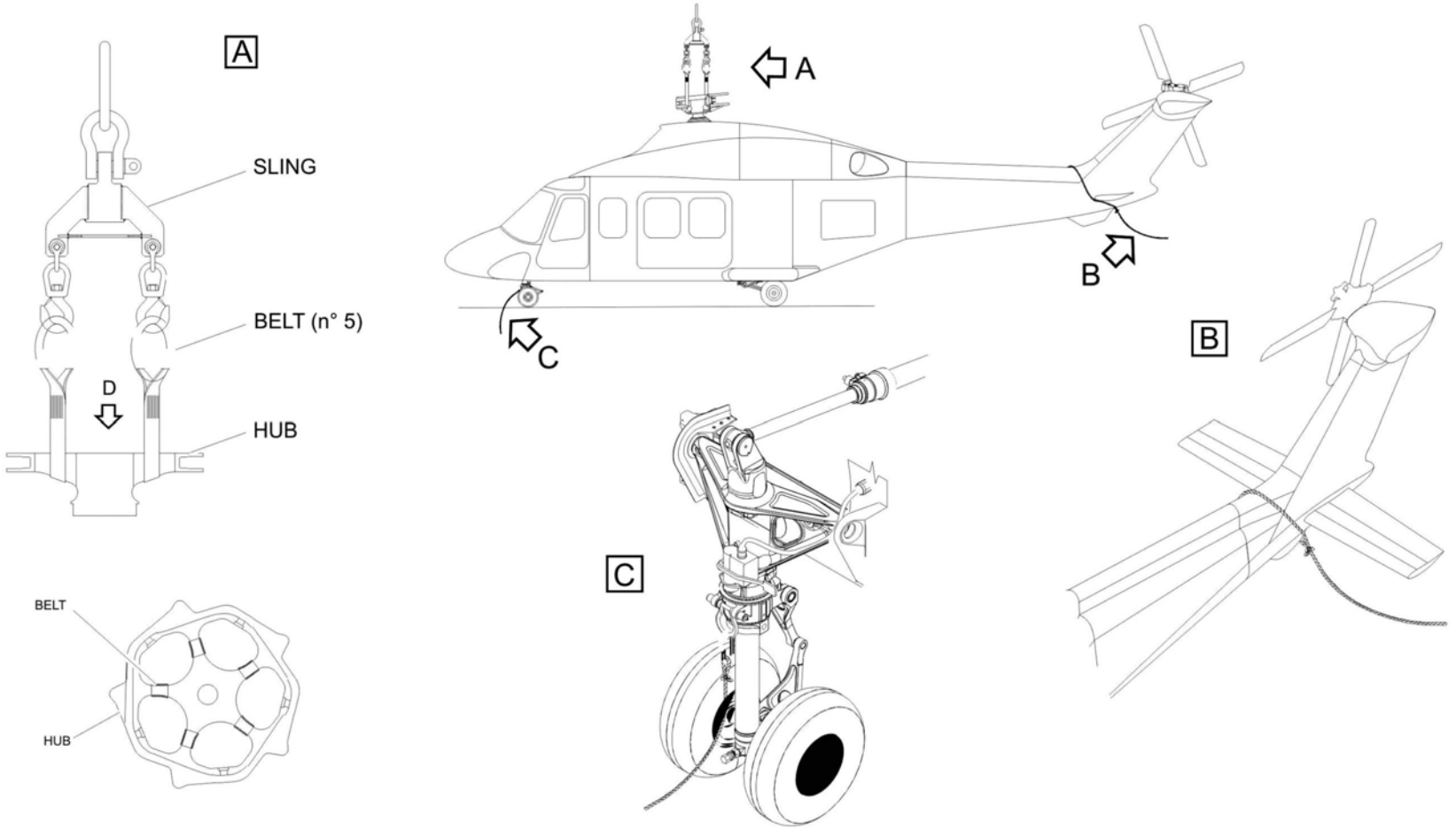
- Make sure that the five belts are at the same distance from the hub axle.
- Make sure that the helicopter is longitudinally stable.

SAFETY PRECAUTIONS

- Be careful when lifting or lowering the helicopter.
  Make sure all the persons who are not necessary for the procedures are clear of the helicopter.
  Do not stay near or on the helicopter during operation.
- Make sure that there is sufficient clearance around the helicopter before you start the lifting operation.
  Make sure that no part of the helicopter touches other objects while is lifted or lowered.
  Damage to the helicopter can occur if a part of it touches other objects.
- Make sure that:
  - The lifting device is serviceable.
  - A locally approved person operates the lifting device.
  - One person is in a position adjacent to the tail to make the helicopter stable during the lift operation.
  - All the tools and the equipment are moved clear of the helicopter.

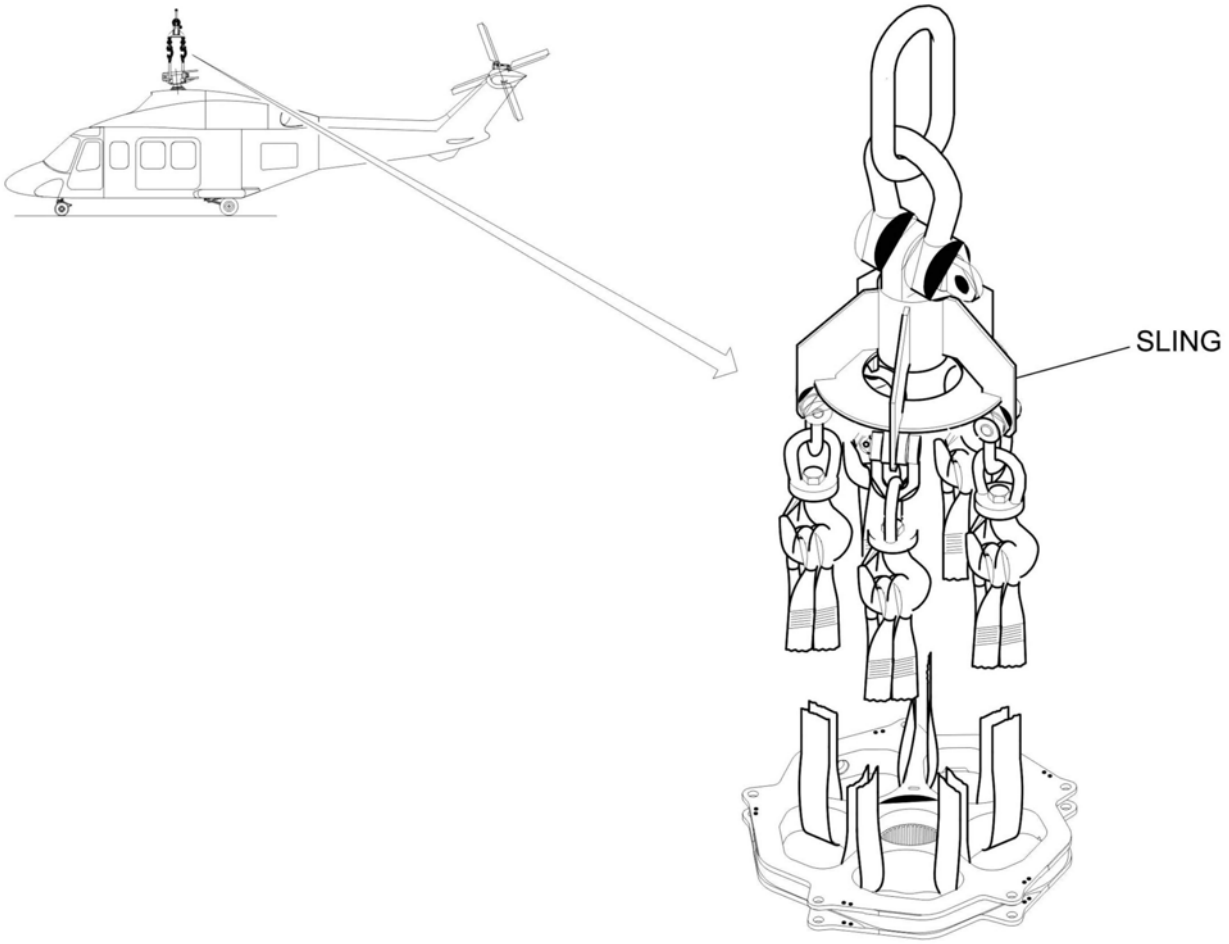
SPECIAL TOOLS / TEST EQUIPMENT

- Sling



ICN-39-A-073000-G-A0126-00505-A-01-1

HELICOPTER SLINGING



ICN-39-A-073000-G-A0126-00506-A-01-1

SPECIAL TOOLS/ TEST EQUIPMENT

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CHAPTER
08
LEVELING AND WEIGHING

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SECTION 20 - LEVELING

MAINTENANCE INFORMATION

- Make sure that the helicopter is in an area (hangar) where the wind can have no effect on it.

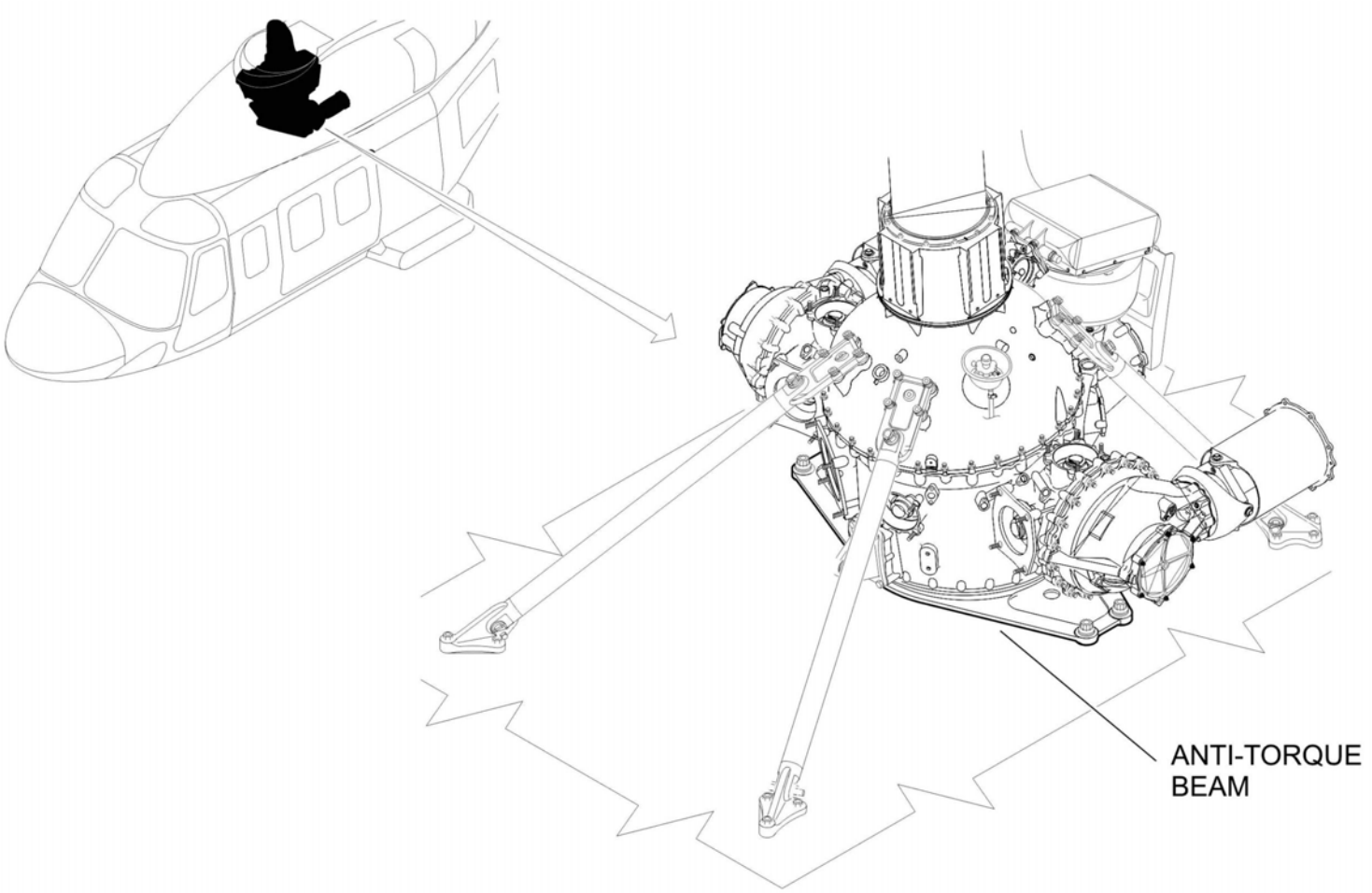
CAUTION

DO NOT LET PERSONS TO GO ON THE HELICOPTER DURING THE LEVEL PROCEDURE. IF PERSONS GO ON THE HELICOPTER THERE IS A RISK THAT THE PROCEDURE IS NOT ACCURATE.

- Lift the helicopter on jacks.
- Put the platform adjacent to left side of the fuselage.
- Open the access door 473AL.
- Get access to the anti-torque beam on the upper deck.
- Put the digital level on the anti-torque beam longitudinally and laterally. then adjust the height of the jacks to get the correct longitudinal and lateral level of the helicopter.
- Lock the ram of each jack with its ram locknut.
- When the procedure that made the helicopter level necessary is completed, lower the helicopter on the ground.

Alternative Level Procedure

- Put the digital level longitudinally and laterally on the seat guides between the STA 4789 and STA 5287 in the cabin.
- Adjust the height of the jacks to get the helicopter longitudinal and lateral level. Do the level check in many different positions.
- Lock the ram of each jack with its ram locknut.
- If necessary to do a check that the helicopter is in the correct level position, do the operations that follow:
  - Get access to the floor of the baggage compartment.
  - With the use of a digital level , do a check on the longitudinal axis of the helicopter. Make sure that the value you read on the digital level is 4.37 ± 0.25° pitch down.
  - When the procedure is complete that made the helicopter level necessary is completed lower the helicopter on the ground.



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HELICOPTER-LEVEL PROCEDURE-GENERAL

SECTION 30 - WEIGHING

GENERAL

Helicopter weighing is carried out in order to calculate the empty weight used for loading and CG data calculation.
The procedure is carried out following dedicated charts which are included in the Rotorcraft Flight Manual.
Charts take into account also items weighed but not part of basic weight and basic items not installed when weighed.
A specific chart takes also into account items removed/added to the basic weight to be compliant with mass and CG data.

NOTE A

In order to avoid stress to the helicopter structure do not insert loads in the forward compartment.

MAINTENANCE INFORMATION

- The helicopter must be compliant with the following conditions before weighing:
- Defuel the Number 1 and Number 2 tank.
- Check and if necessary refill:
  - Engine oil (both Engine)
  - Main Gearbox
  - Intermediate Gearbox
  - Tail Gearbox
  - PCM (both Engine)
  - Brake reservoir
- Set the rotor brake to OFF and align the Number 1 main rotor blade with the tailcone; then set the rotor brake On.
  After previous procedure it is possible to lift the helicopter on jacks.

WARNING

BE CAREFUL WHEN YOU LIFT OR LOWER THE HELICOPTER. MAKE SURE ALL THE PERSONS WHO ARE NOT NECESSARY FOR THE PROCEDURES ARE CLEAR OF THE HELICOPTER. DO NOT STAY NEAR OR ON THE HELICOPTER DURING LIFT OPERATION.

CAUTION

PUT ROPES AROUND THE WORK AREA AND PUT THE JACK WARNING SIGNS AT THE FOUR END SIDES OF THE HELICOPTER.

For further information about the helicopter weighing procedure ref to AMP.

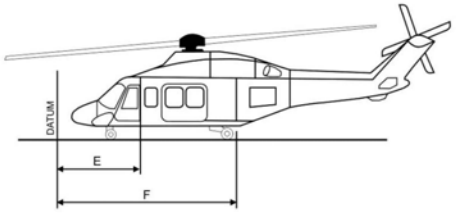
SPECIAL TOOLS / TEST EQUIPMENT

- Weighing adapters kit.
- Forward jacking bracket.
- Weighing cells
- Hydraulic jacks.

CHART B - HELICOPTER WEIGHING RECORD							Sheet 1 of 2
MODEL:	S/N:		REGISTRATION MARKS:				
DATE:	PLACE:		SIGNATURE:				
Reason for weighing:							
Scale type:							
JACKPOINTS	SCALE READING	TARE	NET WEIGHT	STA (1)	LONGITUDINAL MOMENT	BL (2)	LONGITUDINAL MOMENT
	(Kg)	(Kg)	(Kg)	(mm)	(Kgmm)	(mm)	(Kgmm)
FORWARD			3160			0	
LH AFTERWARD			6700			-905	
RH AFTERWARD			6700			905	
TOTAL (as weighed) (to Sht. 2 of 2)							

Note 1
The forward Lower Central Cabin is provided by two FWD jack points, only one FWD jack point is assured by using a proper tool which collect both jacking points.
The Station Reference Datum (STA 0) is located 3160 mm forward the FWD jack points. Therefore the STA are positive.
E = Distance from the reference datum (STA 0) to the FWD jackpoint Station of 3160 mm.
F = Distance from the reference datum (STA 0) to the LH and RH aft jackpoint Station of 6700 mm.

Note 1
The Buttock Line Reference datum (BL 0) is located on the fuselage Center Line.
Therefore the BL are negative on the left Hand side and positive on the Right Hand side.



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CHART B - HELICOPTER WEIGHING RECORD					Sheet 2 of 2
DESCRIPTION	NET WEIGHT (Kg)	STA (mm)	LONG MOMENT (Kgmm)	BL (mm)	LAT MOMENT (Kgmm)
TOTAL (as weighed) (from Sht. 1 of 2)					
SUBTRACT (from Tab.1)					
ADD (from Tab.2)					
BASIC AIRCRAFT (to Chart C)					
TABLE 1 ITEMS WEIGHED BUT NOT PART OF BASIC WEIGHT					
DENOMINATION	NET WEIGHT (Kg)	STA (mm)	LONG MOMENT (Kgmm)	BL (mm)	LAT MOMENT (Kgmm)
TABLE 2 BASIC ITEMS NOT INSTALLED WHEN WEIGHED					
DENOMINATION	NET WEIGHT (Kg)	STA (mm)	LONG MOMENT (Kgmm)	BL (mm)	LAT MOMENT (Kgmm)
TOTAL					
Reason of the weighing:		Type Scales:			

NOTE:
Remove the weight of the mission equipment items, which are pointed out in Chart A [ ]

ICN-39-A-083000-G-A0126-00508-A-01-1

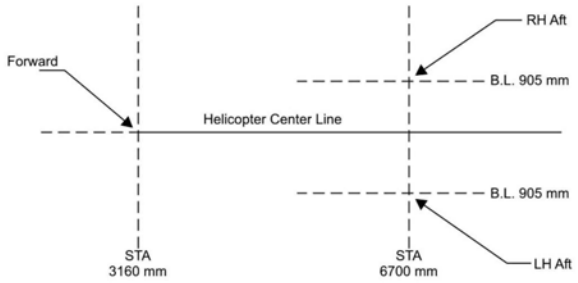
CHART B- HELICOPTER WEIGHING RECORD

CHART C BASIC WEIGHT AND BALANCE RECORD														Page N°
MODEL			S/N			REGISTRATION MARKS								
DATE	Item n° see chart A		DENOMINATION	LOADING CHANGE					BASIC WEIGHT, MOMENT & COG					SIGNAT.
	IN	OUT		WGT (3) (Kg)	STA CG (mm)	LONG MOMENT (Kgmm)	BL CG (mm)	LATERAL MOMENT (Kgmm)	WGT (Kg)	LONG MOMENT (Kgmm)	STA CG (mm)	LATERAL MOMENT (Kgmm)	BL CG (mm)	
	(1)	(2)												

CHART D - DATA FOR HELICOPTER WEIGHT & BALANCE COMPUTATION

MODEL AW139

JACKPOINTS LOCATION



Longitudinal STA. O (Datum) is 3160 mm. Forward of the front jackpoint.

TYPICAL DATA

		ARM		WEIGHT (Kg)
		STA (mm)	BL (mm)	
FUEL	Capacity		0	
	Unusable	6206	0	16
ENGINE OIL	Capacity	6878	0	19.7
	Undrainable	—	—	0
	Baggage	7700		

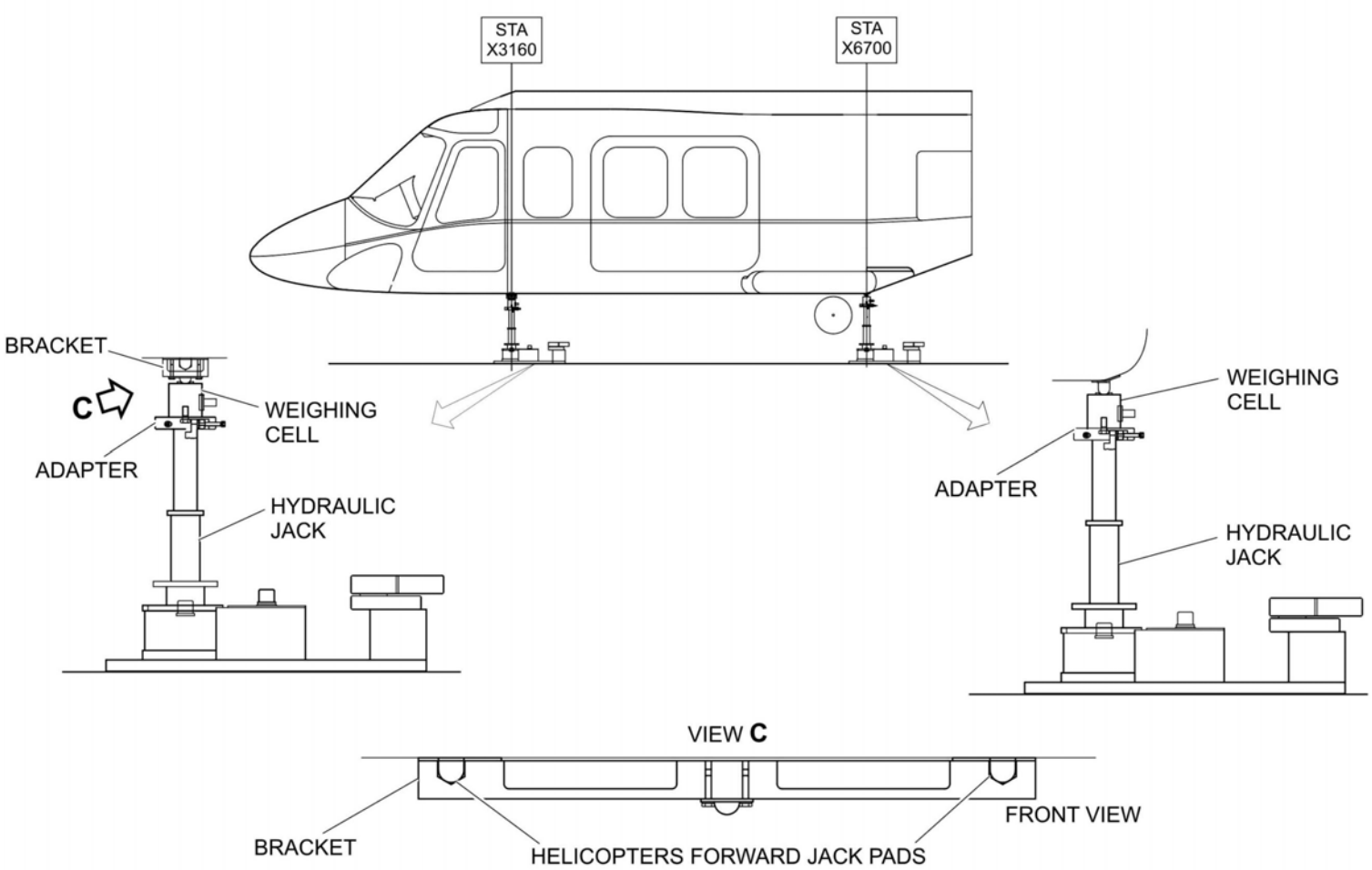
NOTE 1 - For pilot and crew station and for further information see Section 6 of this Flight Manual.

NOTE 2 - For further information see specific section of this Flight Manual.

NOTE 3 - Empty weight includes undrainable oil unusable fuel.

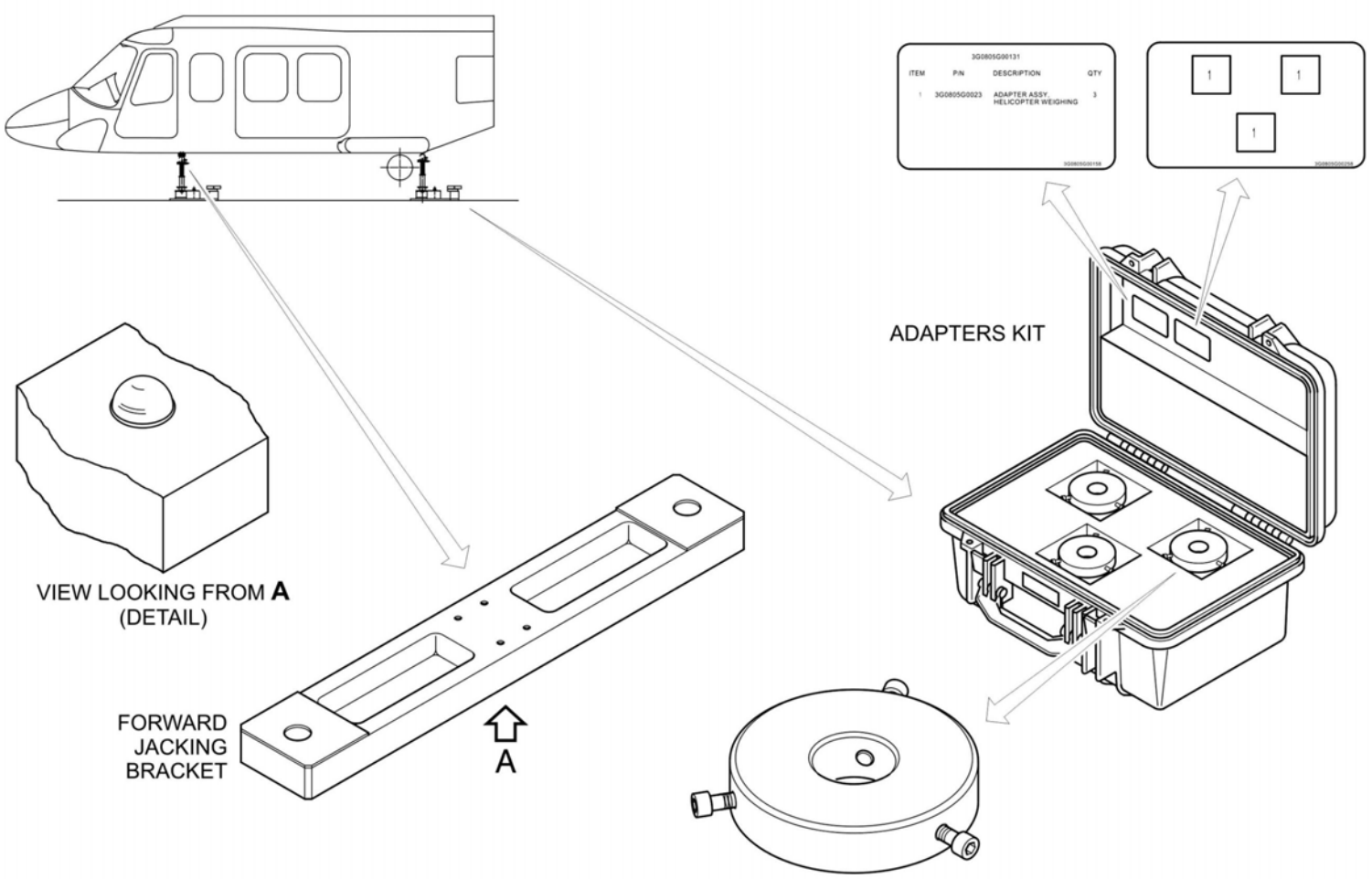
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CHART D - DATA FOR HELICOPTER WEIGHT AND BALANCE COMPUTATION



ICN-39-A-083000-G-A0126-00511-A-01-1

HELICOPTER WEIGHING INSTALLATION



ICN-39-A-083000-G-A0126-00512-A-01-1

SPECIAL TOOLS /EQUIPMENT

SECTION 40 - MASS AND C.G. DATA

GENERAL

Information for the weight and balance computation is provided by Rotorcraft Flight Manual in Section 6.
It is the pilot's responsibility to ensure that the helicopter is properly loaded to maintain the centre of gravity within the limitations for the duration of the flight as defined in Section 1 of the Rotorcraft Flight Manual.
Figures, charts and examples are provided to assist the pilot in computing the proper loading condition.
These charts will also provide for continuous control of weight and balance of the helicopter.
Agusta inserts all helicopter identifying data on dedicated charts that constitute the basic weight and balance data of the helicopter to which the Rotorcraft Flight Manual is assigned.
The operator shall keep this data up-to date by recording all changes made to the configuration of the helicopter.

WEIGHT AND BALANCE DATA



ICN-39-A-158000-A-A003-00039-A-01-1

HELICOPTER AW139

SERIAL NUMBER . . . . . .

REGISTRATION MARKS . . . . . .

ICN-39-A-084000-G-A0126-03469-A-01-1

HELICOPTER WEIGHING

The helicopter must be weighed when one of the following conditions exist:

1 When major modifications or repairs are made, or kits are installed or removed;

2 When the basic weight data is suspected to be an error;

3 At time of major overhaul;

4 In accordance with EASA instructions.

Even if the helicopter must be prepared for weighing according to AMP Chapter 08, instructions for weight and balance determination are available into the Rotorcraft Flight Manual (RFM) Section 6 with enclosed instruction for the charts to enable the operator to obtain all necessary data as to basic helicopter configuration, empty weight and centre of gravity. The required charts and forms are identified as follows:

Chart A – Equipment List

Chart B – Helicopter Weighing Record

Chart C– Basic Weight and Balance Record

Chart D–Data for Helicopter Weight and Balance Computation

Chart E – Weight and Balance Computation

DEFINITIONS

EMPTY WEIGHING

The empty weight considered is in accordance with JAR 29, chapter 29. It is the weight of the aircraft with its normal airborne equipment, including engine oil, trapped oil and fluids and unusable fuel but without mission equipment.

CENTER OF GRAVITY

The c.g. for each component and the aircraft c.g. are referred to a reference frame.

The longitudinal reference datum is an imaginary vertical plane forward of the aircraft nose from which all horizontal distances are measured for balance purposes.
It is a pre-determined distance from a fixed point on the aircraft (i.e. the centre line main rotor datum is Stn 5006.82 mm).
The longitudinal reference datum is Stn 0.
The lateral reference datum is a vertical plane down the longitudinal centre line of the aircraft and is Bl0: distances when viewed from above looking forward are positive to starboard of the centre line and negative to port.
The vertical reference datum is an imaginary horizontal plane beneath the aircraft.
It is a pre-determined distance from a fixed point on the aircraft (i.e. the upper deck floor is Wl 2670 mm), the reference datum is Wl 0.

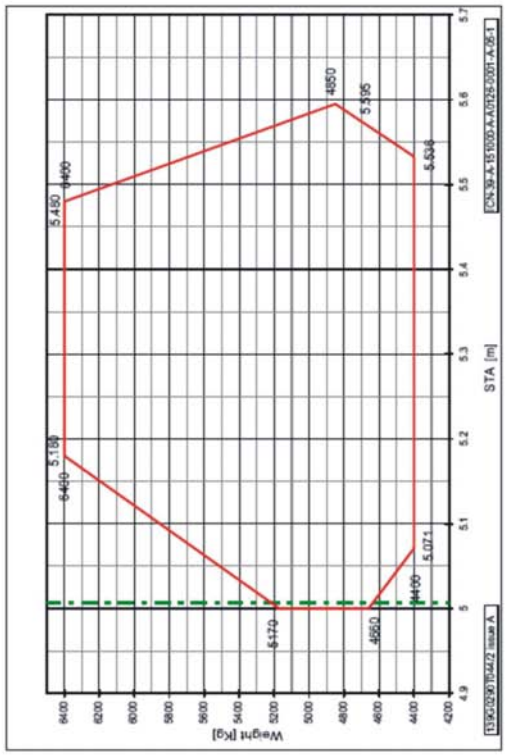


Figure 1-1 Weight and Longitudinal CG Limitations

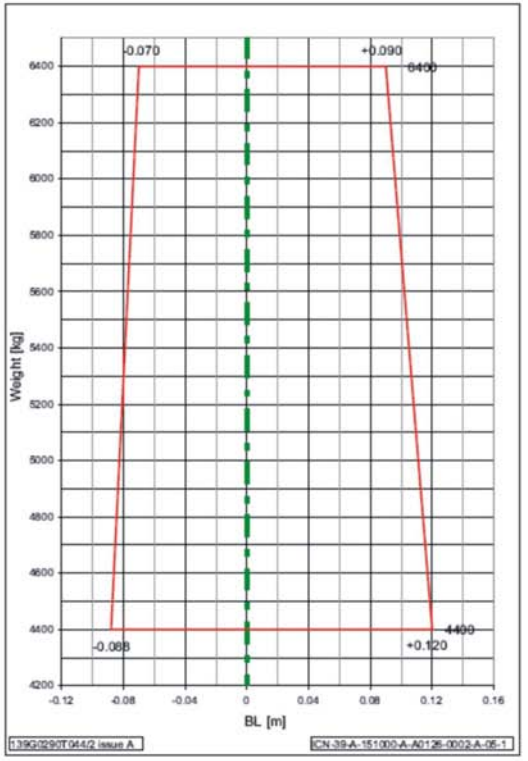
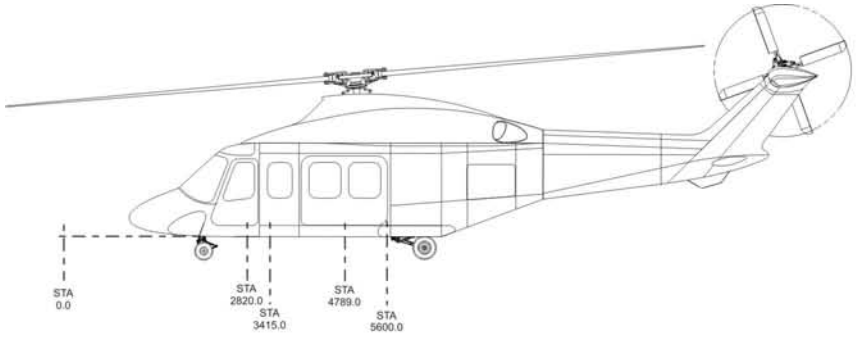
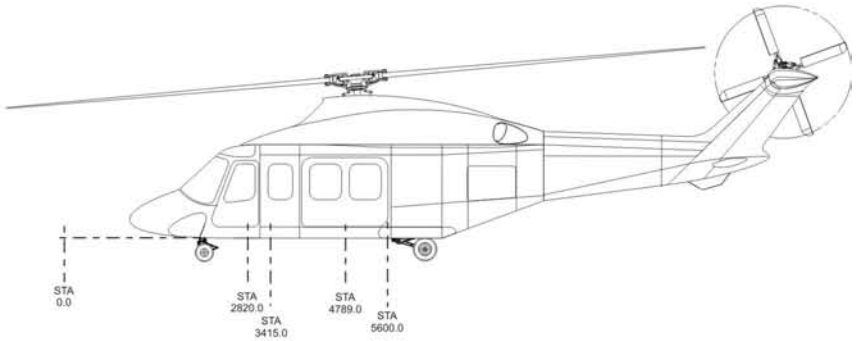
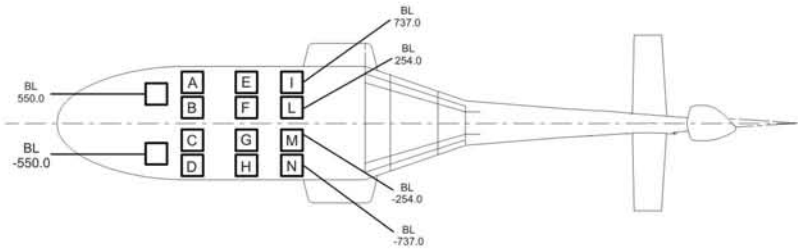


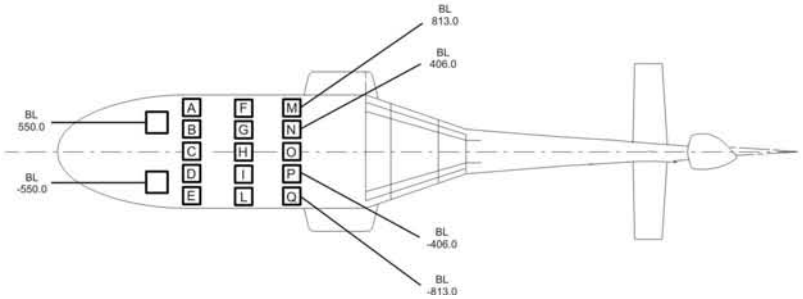
Figure 1-2 Weight and Lateral CG Limitations

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ICN-39-A-084000-G-A0126-00515-A-01-1

HELICOPTER STATION DIAGRAM (12 PASSENGERS)/(15 PASSENGERS)

Use Of Charts And Forms

CHART A

The Chart A gives the weight, arm and moment of all the standard and optional equipment installed on board at the moment of first weighing performed by Agusta and called “Basic Configuration.
A check (V) in the columns headed “in Helicopter” indicates the presence of the item in the helicopter, and a zero (0) indicates its absence.
When one of the following condition exist:

1 When the helicopter undergoes modification, major repair or overhaul;

2 When changes in equipment are made for a different type of operation;

3 When the helicopter is reweighed.

Subsequent check list inventories shall be carried out.

For all pages of Chart A:

— √ Means installed on the helicopter

— O Means not installed on the helicopter

Page 1 of 57

CHART A - EQUIPMENT LIST

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RECORD OF CHECKING (Date and Signature)

MODEL: AW139 | REGISTRATION MARKS

ITEM NUMBER	POWER PLANT SYSTEM		Q.ty	WEIGHT (Kg)	ARM (mm)	MOMENT (Kgmm)	Basic configuration	In helicopter	Chart C entry	In helicopter	Chart C entry	In helicopter	Chart C entry
	P/N	DENOMINATION											
1	3045379	Free turbine turboshaft PT6C-67C	2	200.000	6434	1286800							
2	1152546-2	DC generator	2	13.800	7139	98518							
3	3G7800A00112	Ejector duct installation	1	25.348	7345	186186							
3A	3G7800A00113	Ejector duct installation	1	33.848	7378	249724							

ICN-39-A-084000-G-A0126-00516-A-01-1

CHART A

CHART B

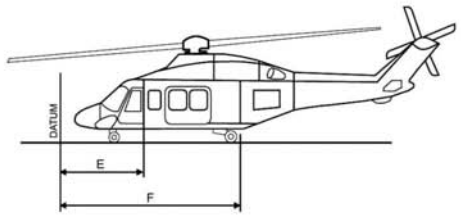
Enter the actual scale readings in the dedicated column for each jackpoints and then subtract tare, if any, from the scale readings to obtain the net weight.
Multiply the net weight by their respective arms to obtain respectively the longitudinal and lateral moments.
Obtain the total net weight and the total moments.
Divide the total moment by the net weight to obtain “as weighed” CG position.
Transfer the total weight arm and moment to the sheet 2 of Chart B.
Subtract the total weight and moment of equipment weighed but not part of the basic helicopter.
Add the weight and moment of unusable fuel.
Add the total weight and moment of the basic items not in helicopter when weighed. Added items shall be market on Chart A
Enter the new basic weight and moment on Chart C.

CHART B - HELICOPTER WEIGHING RECORD			Sheet 1 of 2
MODEL:	S/N:	REGISTRATION MARKS:	
DATE:	PLACE:	SIGNATURE:	
Reason for weighing:			
Scale type:			

JACKPOINTS	SCALE READING	TARE	NET WEIGHT	STA (1)	LONGITUDINAL MOMENT	BL (2)	LONGITUDINAL MOMENT
	(Kg)	(Kg)	(Kg)	(mm)	(Kgmm)	(mm)	(Kgmm)
FORWARD			3160			0	
LH AFTERWARD			6700			-905	
RH AFTERWARD			6700			905	
TOTAL (as weighed) (to Sht. 2 of 2)							

Note 1
The forward Lower Central Cabin is provided by two FWD jack points, only one FWD jack point is assured by using a proper tool which collect both jacking points.
The Station Reference Datum (STA 0) is located 3160 mm forward the FWD jack points. Therefore the STA are positive.
E = Distance from the reference datum (STA 0) to the FWD jackpoint Station of 3160 mm.
F = Distance from the reference datum (STA 0) to the LH and RH aft jackpoint Station of 6700 mm.

Note 1
The Buttock Line Reference datum (BL 0) is located on the fuselage Center Line.
Therefore the BL are negative on the left Hand side and positive on the Right Hand side.



ICN-39-A-156000-A-00003-00029-A-01-1

CHART B - HELICOPTER WEIGHING RECORD					Sheet 2 of 2
DESCRIPTION	NET WEIGHT (Kg)	STA (mm)	LONG MOMENT (Kgmm)	BL (mm)	LAT MOMENT (Kgmm)
TOTAL (as weighed) (from Sht. 1 of 2)					
SUBTRACT (from Tab.1)					
ADD (from Tab.2)					
BASIC AIRCRAFT (to Chart C)					

TABLE 1 ITEMS WEIGHED BUT NOT PART OF BASIC WEIGHT

DENOMINATION	NET WEIGHT (Kg)	STA (mm)	LONG MOMENT (Kgmm)	BL (mm)	LAT MOMENT (Kgmm)

TABLE 2 BASIC ITEMS NOT INSTALLED WHEN WEIGHED

DENOMINATION	NET WEIGHT (Kg)	STA (mm)	LONG MOMENT (Kgmm)	BL (mm)	LAT MOMENT (Kgmm)
TOTAL					

Reason of the weighing:	Type Scales:

NOTE:
Remove the weight of the mission equipment items, which are pointed out in Chart A [ ]

ICN-39-A-084000-G-A0126-00517-A-01-1

CHART B – HELICOPTER WEIGHING RECORD

CHART C

This chart is a continuous history of the basic weight and moment resulting from modifications and equipment is considered the current weight and balance status of the basic helicopter.
Make additions or subtractions to the basic weight and moment in Chart C in order to show always the actual total empty weight with its longitudinal and lateral moments.

NOTE A

If any equipment is not listed on Chart A, determine its weight and arm, and list corresponding data on Charts A and C.

CHART C BASIC WEIGHT AND BALANCE RECORD														Page N°
MODEL			S/N			REGISTRATION MARKS								
DATE	Item n° see chart A		DENOMINATION	LOADING CHANGE					BASIC WEIGHT, MOMENT & COG					SIGNAT.
	IN	OUT		WGT (3) (Kg)	STA CG (mm)	LONG MOMENT (Kgmm)	BL CG (mm)	LATERAL MOMENT (Kgmm)	WGT (Kg)	LONG MOMENT (Kgmm)	STA CG (mm)	LATERAL MOMENT (Kgmm)	BL CG (mm)	
	(1)	(2)												

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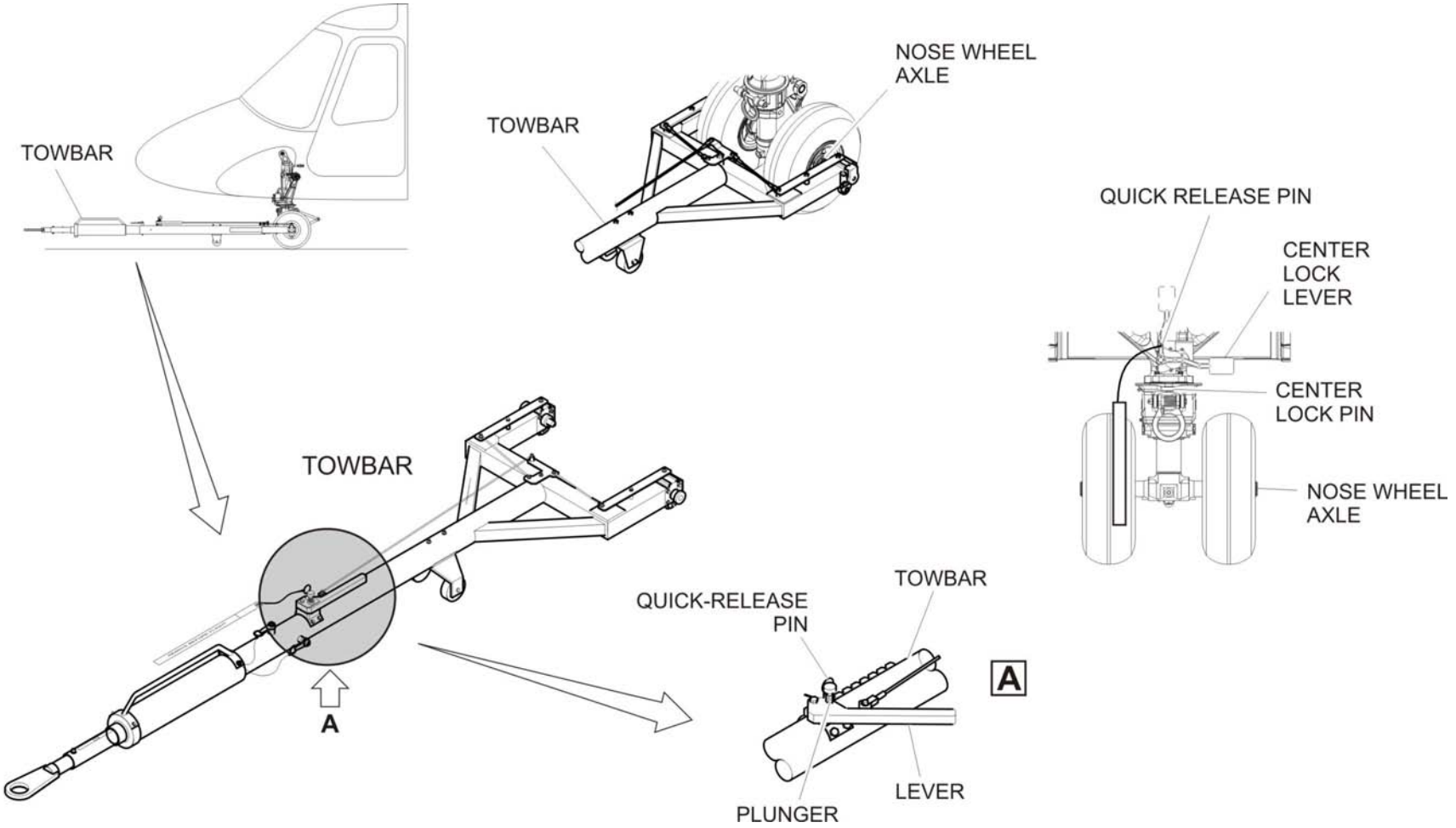
CHAPTER
09
HANDLING AND TAXIING

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SECTION 10 - HANDLING

MAINTENANCE INFORMATION

- Make sure that no part of the helicopter touches other objects while is moved.
  Be very careful with the main rotor blades and tail rotor blades.
  Damage to the helicopter can occur if a part of it touches other objects.
- Make sure that:
  - A person is in the cockpit to operate the wheel brakes.
  - There are sufficient safety persons to check for clearance around the helicopter during the tow operation.
    The safety persons must check the clearance from the tips of the main and tail rotor blades.
  - One person with two chocks is in a position adjacent to each main wheel.
  - All the tools and the equipment are moved clear of the helicopter.
  - Then towbar must be engaged to the nose wheel axle using the lever and then locked with the plunger.
  - The nose wheel center lock lever must be rotated to a vertical position (counterclockwise) to disengage the nose wheel center lock pin before towing the helicopter.
    The lever must be locked, using the quick-release pin stowed on the towbar, during towing.
  - Before disconnecting the towbar, make sure to align the nose wheel such that the center lock lever can be rotated to the horizontal position (clockwise) to engage the nose wheel center lock pin.
  - Prevent sudden starts and stops during tow operation.
  - Apply rotor brake before tow operation.
  - Apply parking brake after tow operation.



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SPECIAL TOOLS / TEST EQUIPMENT

CHAPTER 10

PARKING, MOORING, STORING AND RETURN TO SERVICE

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SECTION 10 - PARKING

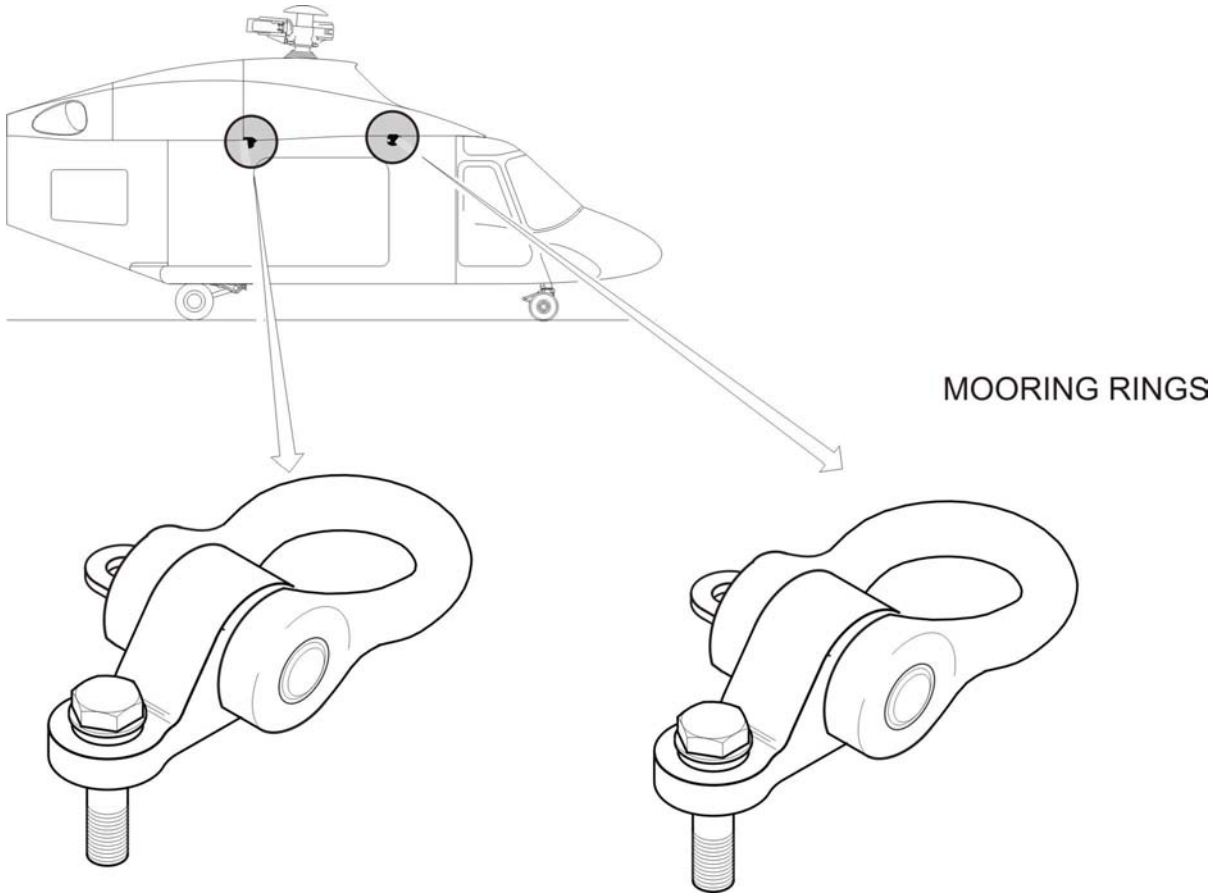
MAINTENANCE INFORMATION

- Helicopter parking is accomplished by the following conditions:
  - Park brake ON
  - Rotor brake ON
  - Nose center lock engaged
  - Main wheel chocks in front and behind each wheel
  - Cyclic stick in center position
  - Collective stick fully down
  - Pitot covers installed
  - Engine exhaust nozzle covers installed.
- Blades tie-dows must be installed if the wind velocity is more than 20Kts or a gust spread of 15 Kts.
  The blades correct position to install the tie-downs is such that one MR blade is perpendicular to the longitudinal axis.
  The tie-downs must be secured on the main landing gears. The main rotor tie downs are socks which fit over the end of each blade with cords which attach to mooring points on the fuselage.
- The pitot tubes will be hot after the engine stops. Let decrease the temperature of the pitot tubes at least five minutes before installing the covers. The pitot covers are attached with a red streamer stenciled in white letters “REMOVE BEFORE FLIGHT”.
- The engine components will be hot after the engine stops. Let decrease the engine components temperature at least thirty minutes before installing the covers.

  Each covers is attached with a red streamer stenciled in white letters “REMOVE BEFORE FLIGHT”.

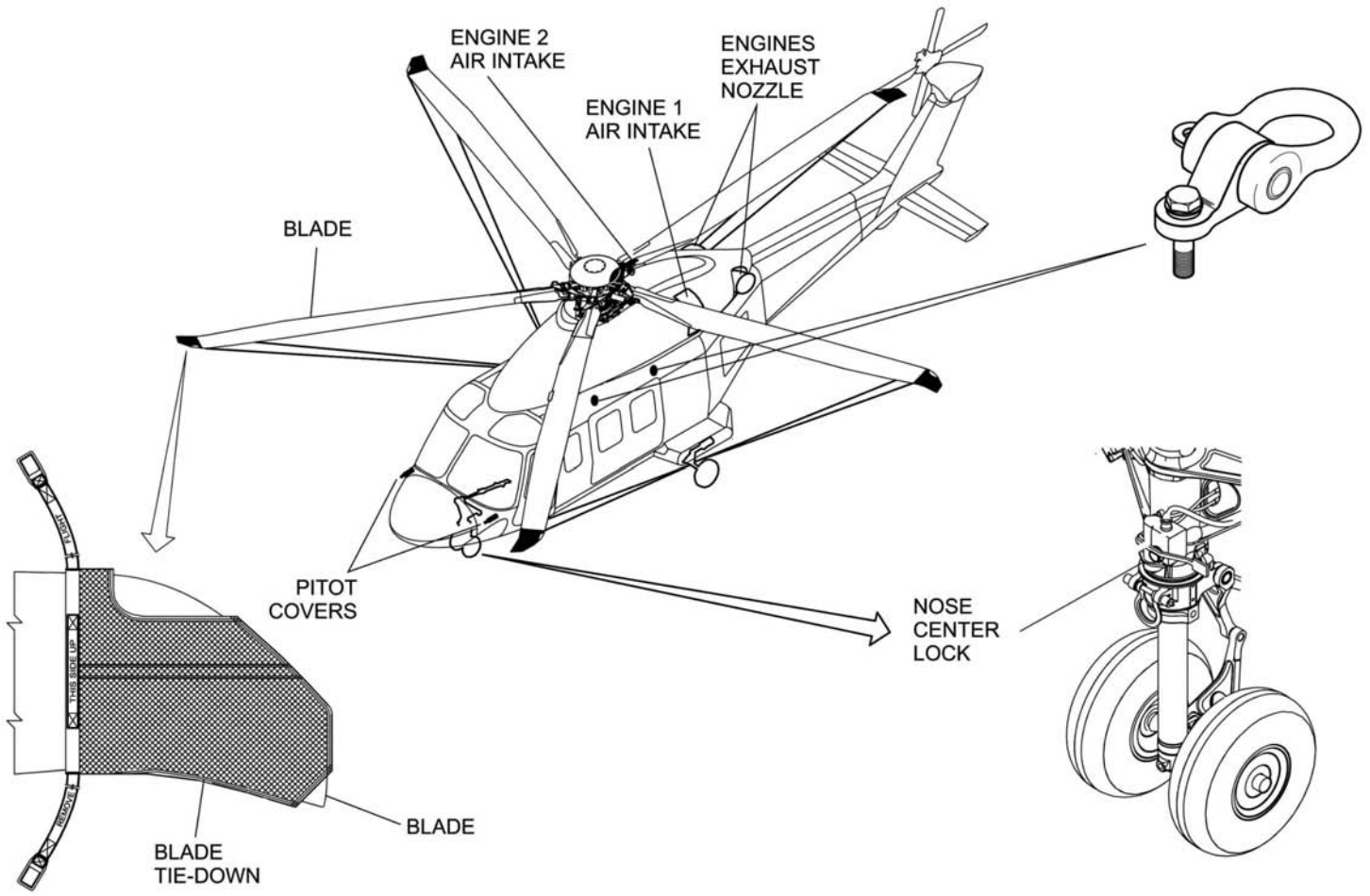
SPECIAL TOOLS / TEST EQUIPMENT

- Pitot covers
- Engine air intake covers
- Engine exhaust nozzle covers
- MR blades tie-down
- Chocks
- Mooring rings.



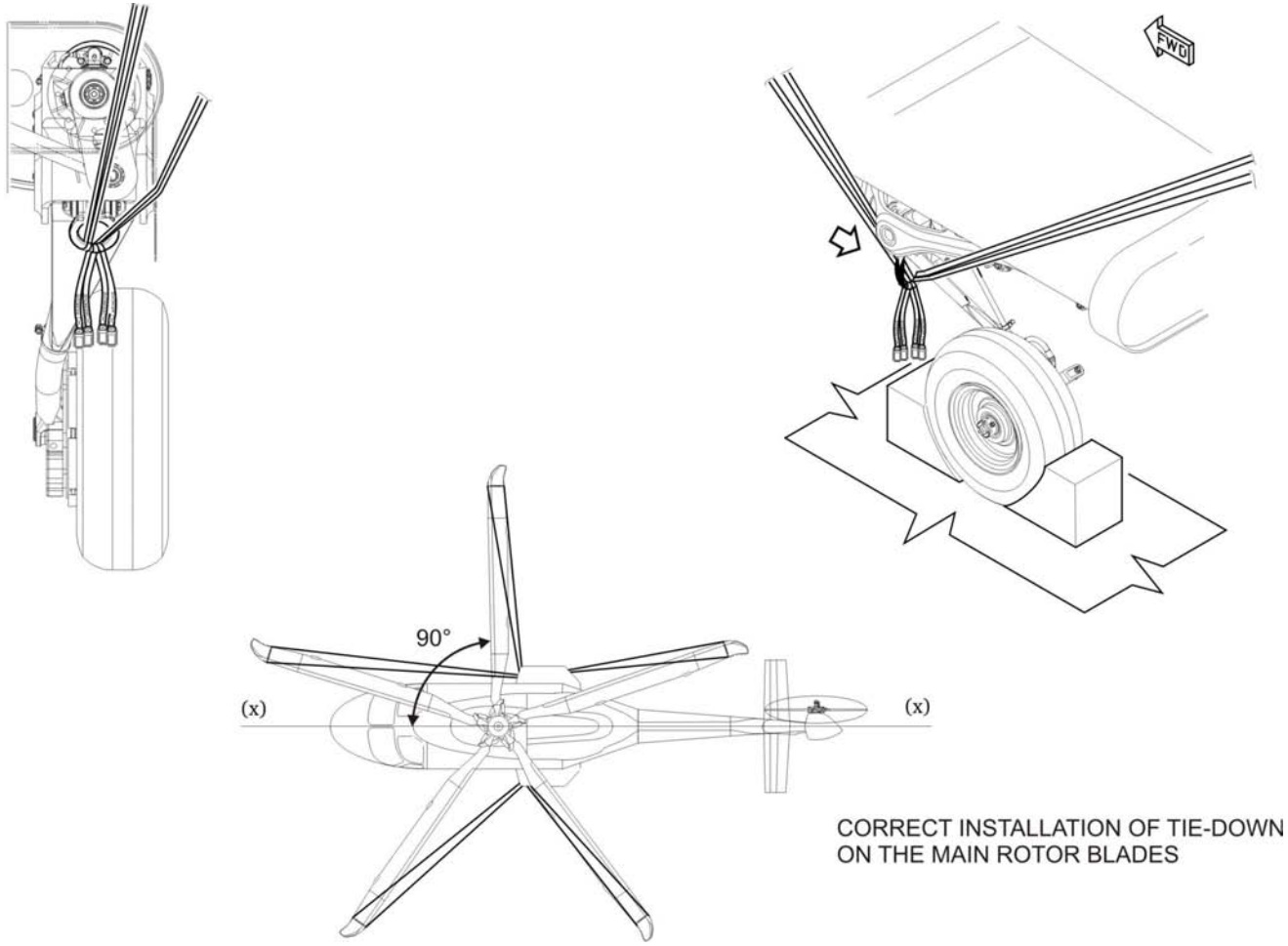
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SPECIAL TOOLS/ TEST EQUIPMENT



ICN-39-A-101000-G-A0126-00521-A-01-1

HELICOPTER PARKING (SHEET 1 OF 2)



ICN-39-A-101000-G-A0126-00522-A-01-1

HELICOPTER PARKING (SHEET 2 OF 2)

CHAPTER

12

SERVICING

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SECTION 00 - SERVICING

GENERAL

This section gives the procedures and data necessary to do servicing on the helicopter.
The servicing section includes:

- Replenishing and depleting section
  (39-A-12-10-00-00A-000A-A)
- Scheduled servicing section
  (39-A-12-20-00-00A-000A-A)
- Unscheduled servicing section
  (39-A-12-30-00-00A-000A-A)
- Support equipment connect/disconnect procedures section
  (39-A-12-40-00-00A-000A-A)

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SECTION 10 - REPLENISHING AND DEPLETING

FUEL

MAINTENANCE INFORMATION

- Tank defueling is carried out by disconnecting the supply hose to the corresponding engine, connect the supply line to a suitable container and operate the tank booster pump.
- When defueling a fuel tank 285 l (228 kg) fuel remains in the other tank.
- When defueling is complete, 15 l thru 20 l of fuel remains in each tank. To drain the tank completely you must operate the water drain valve.

Leading Particulars

- Tank maximum capacity..... 1588 l / 1270 Kg
- Unusable fuel in the two fuel tanks..... 20 l / (16 kg)
- Unusable fuel when cross feeding..... 285 l / 228 kg
- Low fuel caution..... 120 ±5 l / 96 ±4 Kg

Authorized fuels

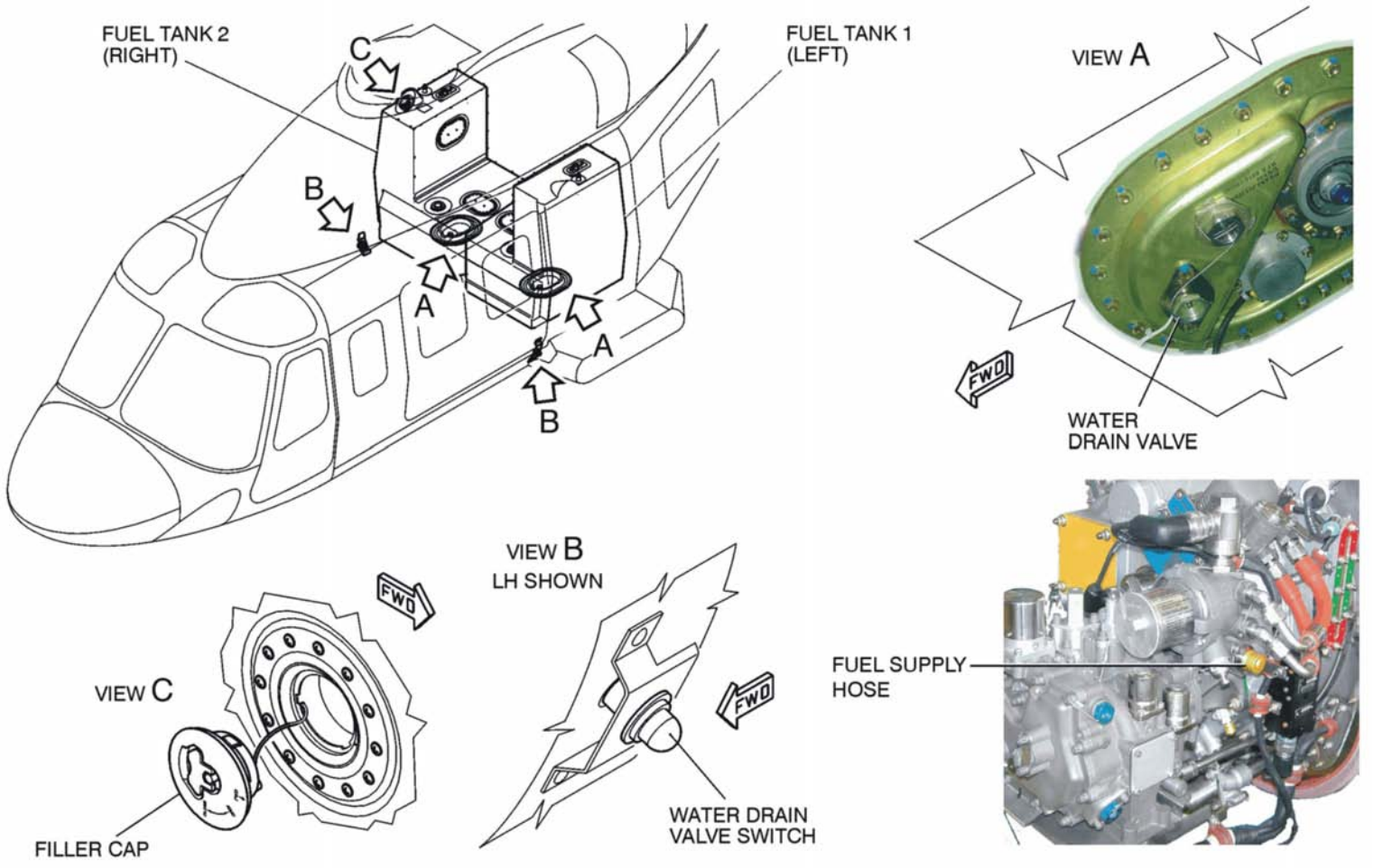
Fuel Type	Applicable Specification
JET A JET A-1	ASTM D1655 ASTM D1655
JP5	DEF STAN 91-86 AVCAT/FSII MIL-PRF-5624F NATO CODE F-44
JP8	DEF STAN 91-87-2002 AVTUR/FSII MIL-T-83138D NATO CODE F-34
JP8+100	Aeroshell Performance Additive 101

SAFETY PRECAUTIONS

- The radio hazard is dangerous.
  Make sure you get the necessary radio-hazard safety precautions.
- Make sure that all equipment ground cables are connected to the applicable earth point of the helicopter.

SPECIAL TOOLS / TEST EQUIPMENT

- Fuel hoses (part of fuel system test stand).



ICN-39-A-121000-G-A0126-03286-A-01-1

FUEL TANKS-REFUEL/DEFUEL

LUBRICANT

Maintenance Information

- The oil must be removed immediately from the painted surfaces because it can cause damage to the paint.
- The transmission oil filler cap is locked if you don't see the circumferential red mark on it.
- The MGB oil is drained using a tool connected to the coupling located in front of the MGB. Suction is provided by a motor driven pump.
- The IGB/TGB oil is drained using a special tool connected to the chip detector valve (chip detector must be removed).
- Engine oil draining is achieved using a pan and removing drain hoses below engine accessory gearbox.

Authorized engine oil:

- Type I/3 cs..... (i.e. BP Turbo Oil 2389. Mobil Avrex S Turbo 256)
- Type II/5 cs..... (i.e. Aeroshell Turbine Oil 500, Castrol 5000, Mobil Jet Oil II, Royco Turbine Oil 500, BP Turbo Oil 2380, Turbonycoil 525-2A)
- Third generation/5 cs..... (i.e. Aeroshell Turbine Oil 560, Royco Turbine Oil 560, Mobil Jet Oil 254)

NOTE A

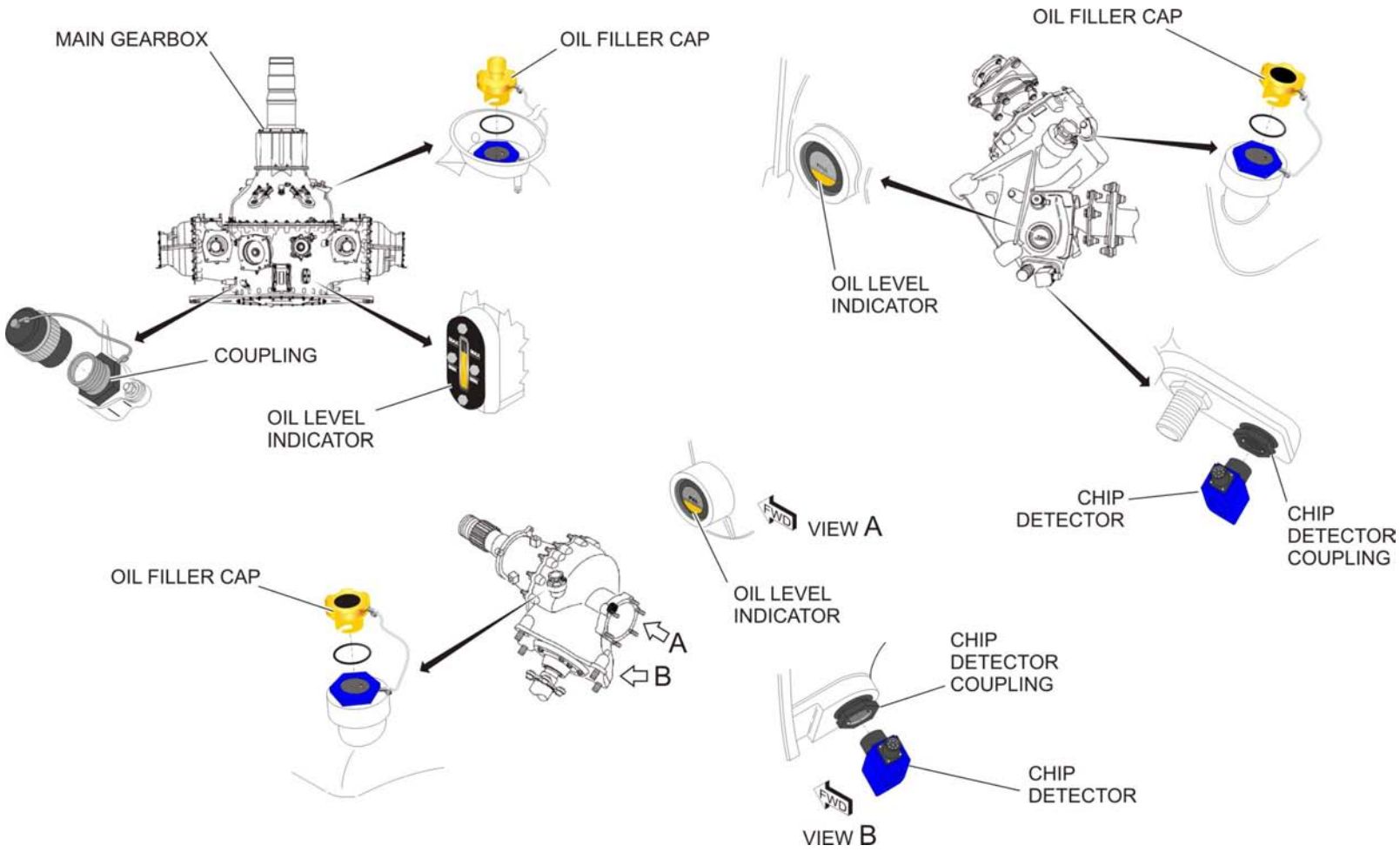
a. Type II oils are not approved for use at ambient temperature less than – 40°C.
b. Mixing of any oils by type, specification or brand name is prohibited.
c. Changing to a "Third generation" oil is recommended only if the engine is new or recently overhauled.

SAFETY PRECAUTIONS

- The oil is a dangerous material.
  Make sure that you know all the safety precautions and first aid instructions for the oil.
- The transmission and engine components are hot after the engine stops. The hot components can cause injury to personnel.

SPECIAL TOOLS / TEST EQUIPMENT

- MGB oil draining pump
- MGB oil draining tool (part of servicing adapters kit)
- IGB / TGB oil draining tool (part of servicing adapters kit)



ICN-39-A-121000-G-A0126-03288-A-01-1

HYDRAULIC

Maintenance Information

Parking Brake System

- Make sure that the parking brake is released before gravity refilling the wheel brake reservoir.

PCM#1 and PCM#2

- In order to fill the PCM reservoir, disconnect reservoir drain hose and connect a transparent tube to monitor air bubbles in the oil when the bleed valve is open.

Rotor Brake system

- Make sure that the rotor brake is released before draining rotor brake reservoir.

AUTORIZED HYDRAULIC OIL

- MIL-PRF-83282..... (i.e. Aeroshell Fluid 31)
- MIL PRF-5606..... (i.e. Aeroshell Fluid 41)

NOTE B

MIL-PRF-5606 can be used for enhanced performance of hydraulic system in low temperature environments below -30°C

SAFETY PRECAUTIONS

- The hydraulic fluid is a dangerous material.
  Make sure that you know all the safety precautions and first aid instructions for the hydraulic fluid.
- Remove all hydraulic fluid from painted surfaces immediately because it can cause damage to the paint.

Special Tools / Test Equipment

Parking Brake System

- Wheel brake reservoir wrench (part of servicing adapters kit)

PCM#1 and PCM#2

- Quick disconnecting coupling, PCM oil filling.
- Manual hydraulic pump

Rotor Brake system

- No special tools required.

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SECTION 40 - SUPPORT EQUIPMENT CONNECT / DISCONNECT

GENERAL

The support equipment connect / disconnect procedures are related to:

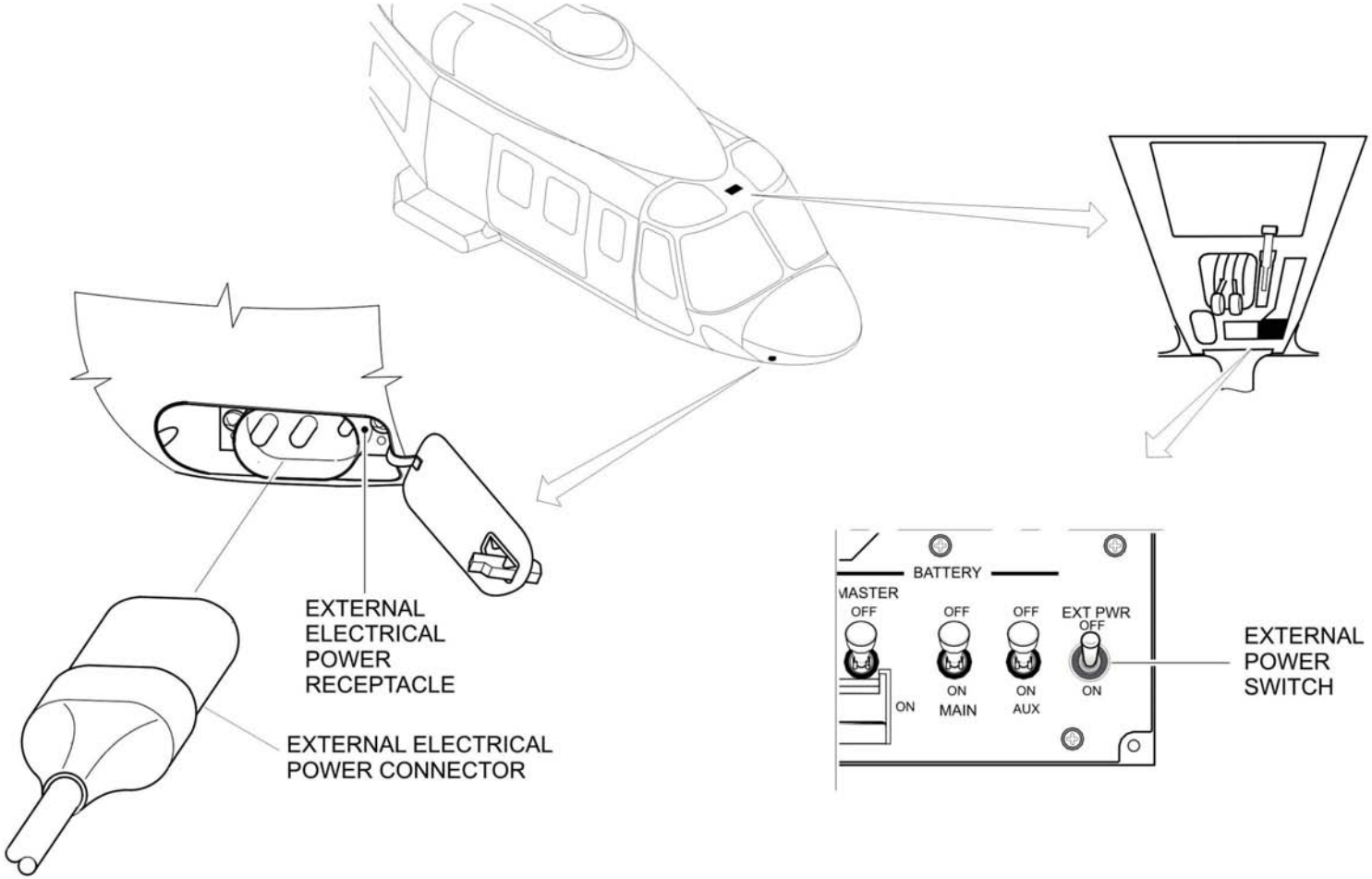
- External electrical power
- External hydraulic power

MAINTENANCE INFORMATION

- The pressure / return adapter couplings are used when connecting the hydraulic test bench to the PCM with the forward sliding fairing open (not removed) to prevent interference with pressure / return lines.

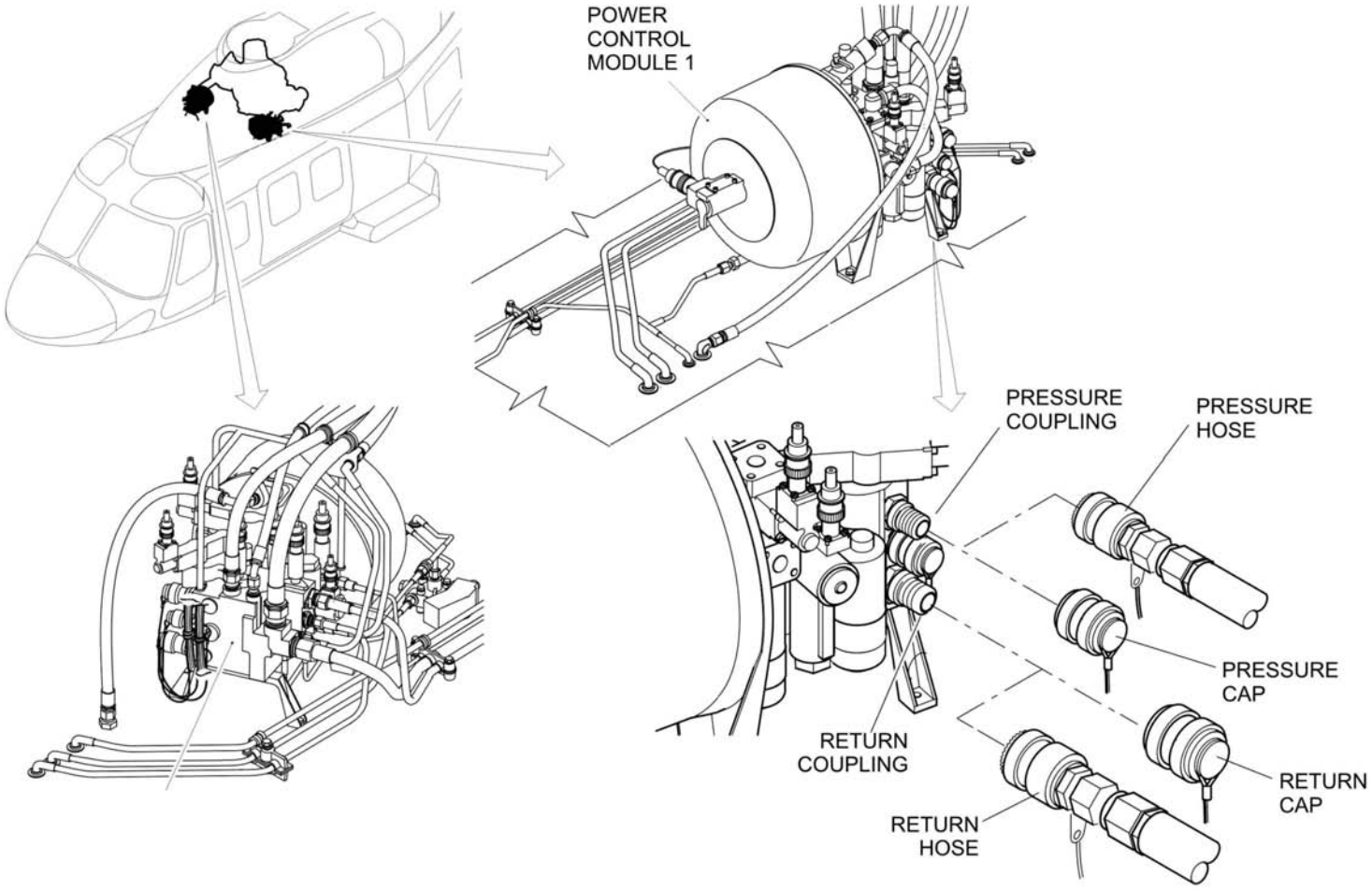
SPECIAL TOOLS / TEST EQUIPMENT

- Hydraulic test bench
- PCM adapters kit



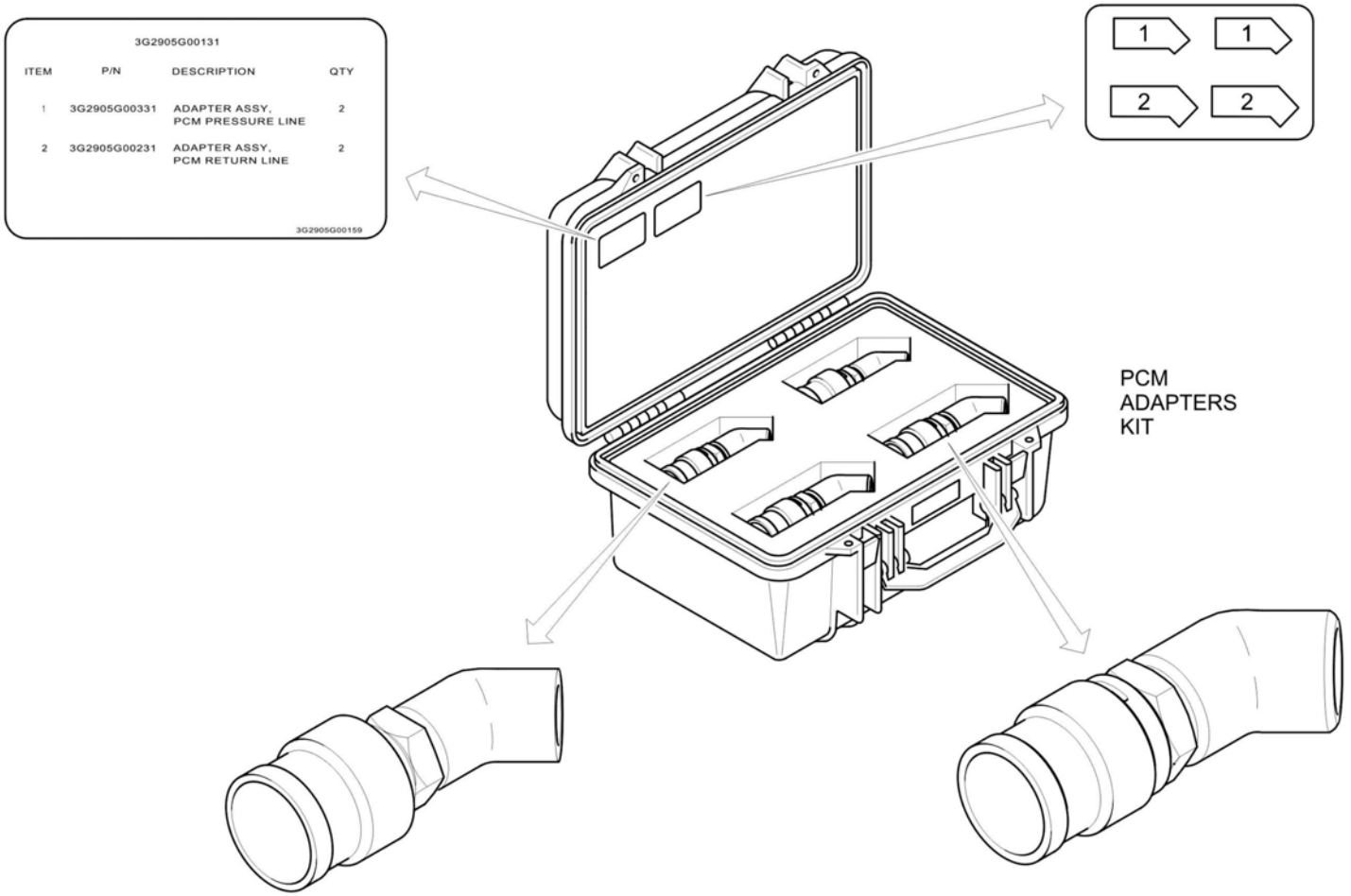
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EXTERNAL ELECTRICAL POWER CONNECT/ DISCONNECT



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EXTERNAL HYDRAULIC POWER CONNECT/DISCONNECT



ICN-39-A-124000-G-A0126-00547-A-01-1

SPECIAL TOOLS/TEST EQUIPMENT

CHAPTER 18

VIBRATION AND NOISE ANALYSIS AND ATTENUATION

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SECTION 10 - VIBRATION ANALYSIS

GENERAL

Mechanics are primarily interested in vibrations felt during in-flight or ground operations as felt in the cabin.
Most vibrations are always present in the helicopter at low magnitudes. It is when the magnitude of any vibration increases that it becomes of concern. The main problem is deciding when a vibration level has reached the point of being excessive. Extreme low frequency, and most medium frequency vibrations are caused by the rotor or dynamic controls. Various malfunctions in stationary compartments can affect the absorption of damping of the existing vibrations and increase the overall level felt by the pilot. A number of vibrations are present which are considered a normal characteristic of the machine.
The N per revolution (N/rev) vibration is the most prominent of these, with N +1/rev or N-1/rev the next most prominent. There is always a small amount of high frequency present. Flight experience is necessary to learn the normal vibration levels.
Even experienced pilots sometimes make the mistake of concentrating on feeling one specific vibration and conclude that the vibration level is higher than normal when actually it is not, it seems so because the pilot is concentrating on it.
For simplicity and some sort of standardization vibrations are arbitrarily divided into general frequencies as follows:

- Extreme Low Frequency..... Less than 1/rev, Normally limited to suspended transmission mount installation such as in the Bell helicopter types pylon rock.
- Low Frequency..... 1/rev or 2/rev type vibration.
- Med. Frequency..... Generally 4/rev or 5/rev, commonly is a N7rev vibe in multibladed systems
- High Frequency..... Tail rotor or faster.

EXTREME LOW FREQUENCY

Extreme Low Frequency vibration is virtually limited to pylon rock. This is encountered with suspended transmission systems which is typical in Agusta Bell helicopters such as the AB 212-412 series.

LOW FREQUENCY VIBRATION

Low frequency vibrations, 1/rev and 2/rev are caused by the rotor itself. 1/rev vibrations are of two basic types, vertical or lateral. A 1/rev is caused simply by one blade developing more lift at a given point than the other blade develops at the same point.
A lateral vibration is caused by a span wise unbalance of the rotor due to a difference of weight between the blades, difference in Span Moment Arms, the alignment of the CG of the blades with respect to the spanwise axis which affects chordwise balance, or unbalance of the hub or stabilizer bar.
Rigidly controlled manufacturing processes and techniques, eliminate all but minor differences between blades, resulting in blades which are virtually identical.
The minor differences which remain will affect flight but are compensated for by adjustments of trim tabs, pitch settings and Dynamic balance weights.
Smoothing of l/rev verticals is essentially a trial and error process although most rotor heads behave reasonably predictably. Dynamic Balance Equipment manufacturers use a sample head which is in a known, good condition and “map” out the various adjustments and responses obtained. This is done by applying a known adjustment (eg 1lb of mass on the Dynamic weight station at the hub end of the blade) and observe the magnitude of vibration created or reduced and in what direction the movement took place. After much data

collection, any change to any Dynamic adjustment can be reasonably accurately and reliably forecasted. This is then amalgamated into a software package and becomes the heart of any RTB equipment on the market today. It replaces the older manual clock angle charts which most would be familiar with. It enables multi- adjustments to be made reliably and predictably. If the charts do not produce the expected results, the only conclusion that can be drawn (assuming the operator is competent and skilled in operating the respective RTB equipment) is that there is either a problem with the hardware i.e. the blades, the head, or parts thereof assuming the accelerometers and cabling have been installed in the correct direction/position and are functioning correctly.
Once in a while it will be found to be impossible to get two blades flying satisfactorily together and it will be necessary to remove and replace one blade this is more than likely a span moment arm problem.
The only option to date has been to swap blades and try and match sets of blades to try and get an acceptably smooth aircraft.
Laterals: Should a rotor, or rotor component, be out of balance, a 1/rev vibration called a lateral will be present.
Laterals existing due to an unbalance in the rotor are of two types; spanwise and chordwise. Spanwise unbalance is caused by one blade or hub being heavier than the other (i.e. an unbalance along the rotor span) or the Span moment arm of one blade being different from the other blade's.
A chordwise unbalance means there is more weight toward the trailing edge of one blade than the other.

MEDIUM FREQUENCY VIBRATION

Medium frequency vibration at frequencies 4/rev and 6/rev are another inherent vibration associated with most rotors. In two bladed teetering head systems, an increase in the level of these vibrations is caused by a change in the capability of the fuselage to absorb vibration, or a loose airframe component, such as the skids vibrating at that frequency. The most common cause is loose skids caused by worn, loose or incorrect skid retaining straps.

In multi-bladed systems, medium frequency vibrations are normally Nper rev and harmonics thereof. N being the number of rotor blades in the particular system being examined. it is quite common to have numerous absoption devices to minimize the component. it's cause is discussed in depth in N per rev.

HIGH FREQUENCY VIBRATION

High frequency vibrations can be caused by anything in the helicopter that rotates or vibrates at a speed equal to or greater than of the tail rotor. This includes many unusual situations such as hydraulic line buzzing, or starter relay buzzing, to the most common and obvious causes; loose elevator linkage at the swashplate horn, loose elevator, or tail rotor balance and track.

NPER REV VIBRATION

N per Rev vibration is a function of the number of blades within the rotor system. For example a 4 bladed system would have an inherent 4 per rev freq, a 5 bladed system, a 5 per rev and so on. Associated with these, are the N-1, N+1 and harmonics of all these combined to from a cocktail of medium freq vibrations.
Some people often believe that N per rev vibration is a function of the RTB or rotor smoothing exercise. This is incorrect. The N per smoothing is a separate operation but is often carried out at the same time or in conjunction with an RTB exercise for obvious reasons that the vibration equipment is already fitted. It is true that the smoother a rotor system is, then it tends to unmask the Nper rev vibration such that they appear a lot worse to the aircrew inside even though no adjustments were done to the absorbers. This is because the higher levels of 1 per rev vibration tends to mask the N per rev vibration. As the 1 per rev levels are reduced toward zero, the N per rev vibe becomes apparently more noticeable. If you were to take actual IPS level readings, you would see that the actual IPS magnitude has not altered – merely become more noticeable.

The common explanation used is that it “is an inherent vibration within the rotor system”.
Nper rev and its associated harmonic derivative frequencies are product of two main sources:

1 Rotor Blade flapping to equality. The greatest contributor is the action of the rotor blade flapping to equality to equalize lift across the disc throughout its rotation. It is caused by energy transfer from rotor blade to airframe via the transmission. This energy is created in the rotor in the form of a “whiplash” action induced along the blade as the tip of the blade reverses direction of travel. This change in direction of the tip is caused by the ever-changing creation of lift as the local angle-of-attack is constantly altering as the local relative wind ever-changes with constantly varying local airflows throughout the rotation of the rotor blade.
These changes are most prevalent with increasing Airspeed. Hence most compensating devices are tuned for best effect at cruise or high IAS.

2 Airframe interference. Airframe interference is also a contributor but is really only prevalent at slow Airspeed. It is caused by the absorption of energy from the “pulse” of air that each blade pushes downward as it passes over a large, flat surface. This surface may be a larger flat stabilator, an empennage (these are mostly rounded and semi-aerodynamic however), winglet, or upper structure covering the roof/ transmission area. The roof and transmission area is close to the root of the blades which generally has far less “pulsed” downwash than the midspan to tip portion of the blade. A lot of the inboard portion of the blade is producing little lift if any through most of its flight regime unless in the hover. It is often either stalled or in flow reversal for a good portion of the flight envelope.
I can be seen that airframe interference will really only be of any consequence at slow speed flight or in the hover.

DAMPENING DEVICES

N per revs are broken into both an Nper vertical and an Nper lateral component. Depending the aircraft make/model, the severity of each component varies. the severity determines whether the Manufacturer will provide dampening for both vertical & lateral components or just dampening for the most noticeable or predominant component. The AW139 is provided with some cabin absorbers arrangement to reduce the vertical components.

NOTE A

If the helicopter has the Health and Usage Monitoring System (HUMS), you can do the blade track check of the main rotor through the HUMS. This chapter deals with the basic equipment C/H VIBREX 2000.

MAIN ROTOR TRACK AND BALANCE

Purpose of the M/R track and balance is to reduce the level of vibrations, relative to the 1 per REV transferred by the Main Rotor to the helicopter structure.
During the tracking it is necessary to adjust the Pitch Links and the Trim Tabs in order to reduce the vertical displacement of the tip of the blades.
Note that the purpose of the procedure is not to have the rotation of the blades on the same plane, but to reduce the level of the 1 per REV vertical vibration as much as possible.
In this procedure the scope is to bring the vertical inertial axe of the rotor as close as possible to the axe of rotation by putting masses in the Blade Retention Bolts; in this case the correction is on the lateral component of the 1 per REV.
For what concerns the Gross Weight during the trials:

- on ground, the lateral unbalance, is more evident with low Gross Weight.
- on ground the tracking with high gross weight is more penalized for the torque used.

We consider in any case a gross weight of more than 5000 Kg (11025 Lb).
The procedure is divided in:

- Dynamic balance and tracking on ground (MPOG).
- Tracking and Dynamic balance in hovering.
- Fine Dynamic balance on ground (MPOG)
- In flight tracking at various speeds.

CORRECTION

- The correction for lateral unbalance is made by adding or removing weights on blades bolts, equally distributed maximum 0.2 Kg (0.44 lb) per bolt.
- For the tracking, the correction is on the pitch links; a complete turn corresponds to 1 deg. of pitch change.
- The pitch link is divided in 15v parts giving a sensitivity of 1/15 of degree.
- The trim tabs are two, located on the trailing edge of the blade with a maximum bending angle of 8 degrees.

Acting on those Trim Tabs is possible to give a torsional deformation to the blade changing the pitch angle. The correction, if made on both Tabs, has to be given on the same direction for a maximum amount of 8 degrees up or down.

IN FLIGHT CORRECTION

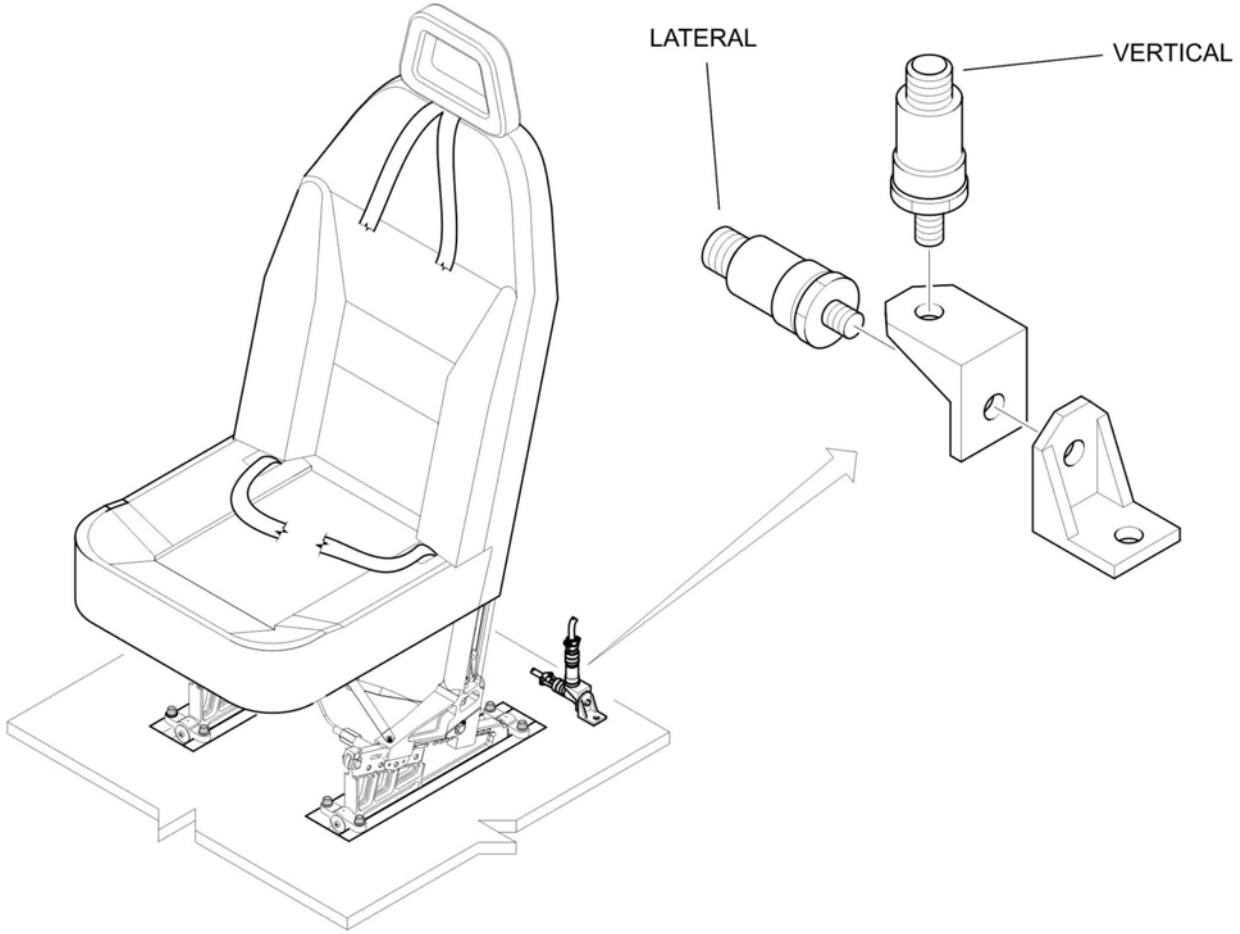
- Pitch Links: MPOG, hovering, low speeds (IAS ≤80 Kts)
- Outer Tab: low speeds (80 < IAS < 140 Kts)
- Inner Tab: High speeds (IAS ≥140 Kts)

There’s not a precise law that established the difference in between trim tab angle and blade pitch change.

For reference only, to have an effective correction an angle of a least 1 or 2 degrees should be given to the external one and 2 or 3 degrees on the internal ones.

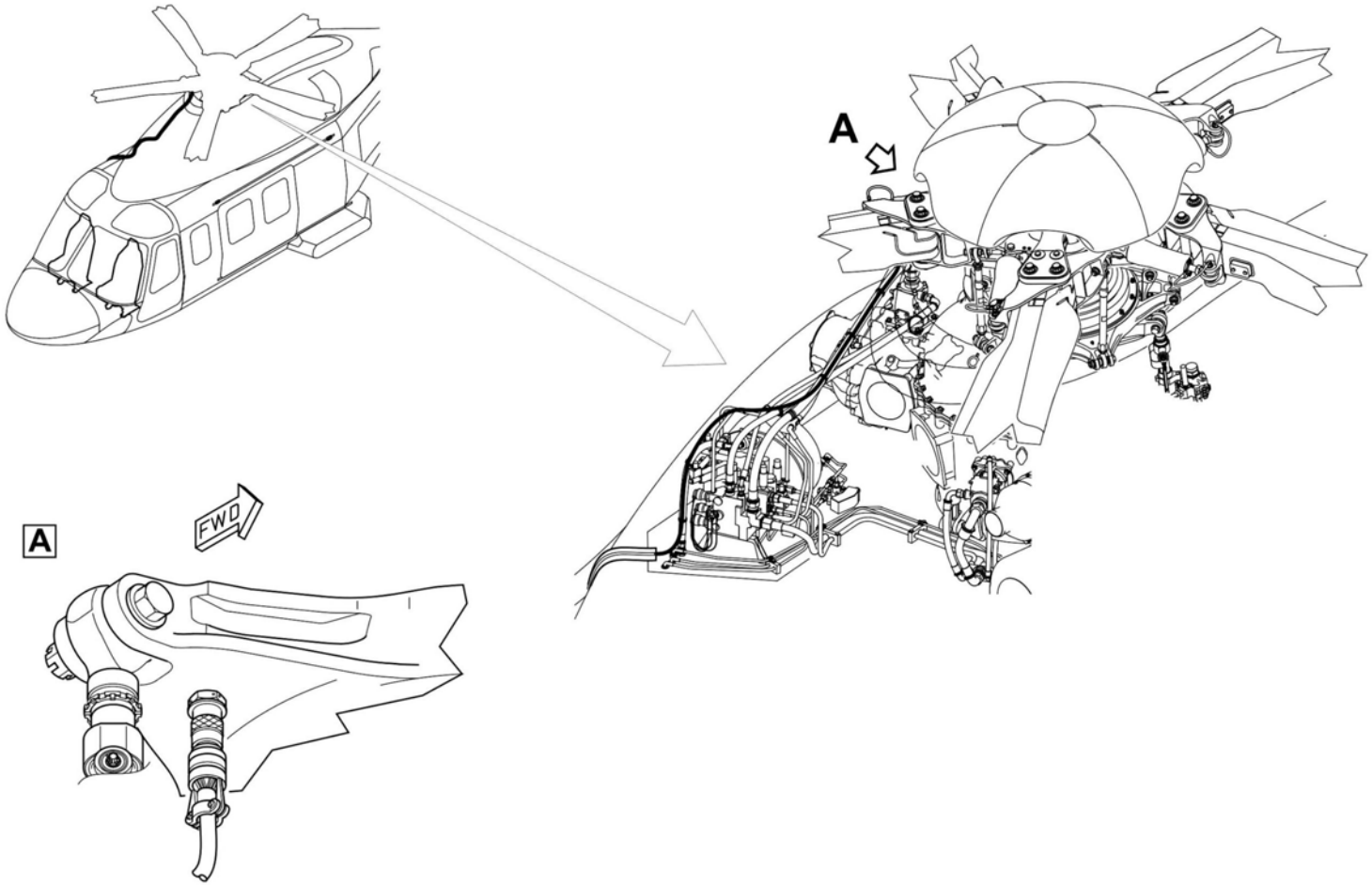
NOTE B

The correction have to be made on the same directions



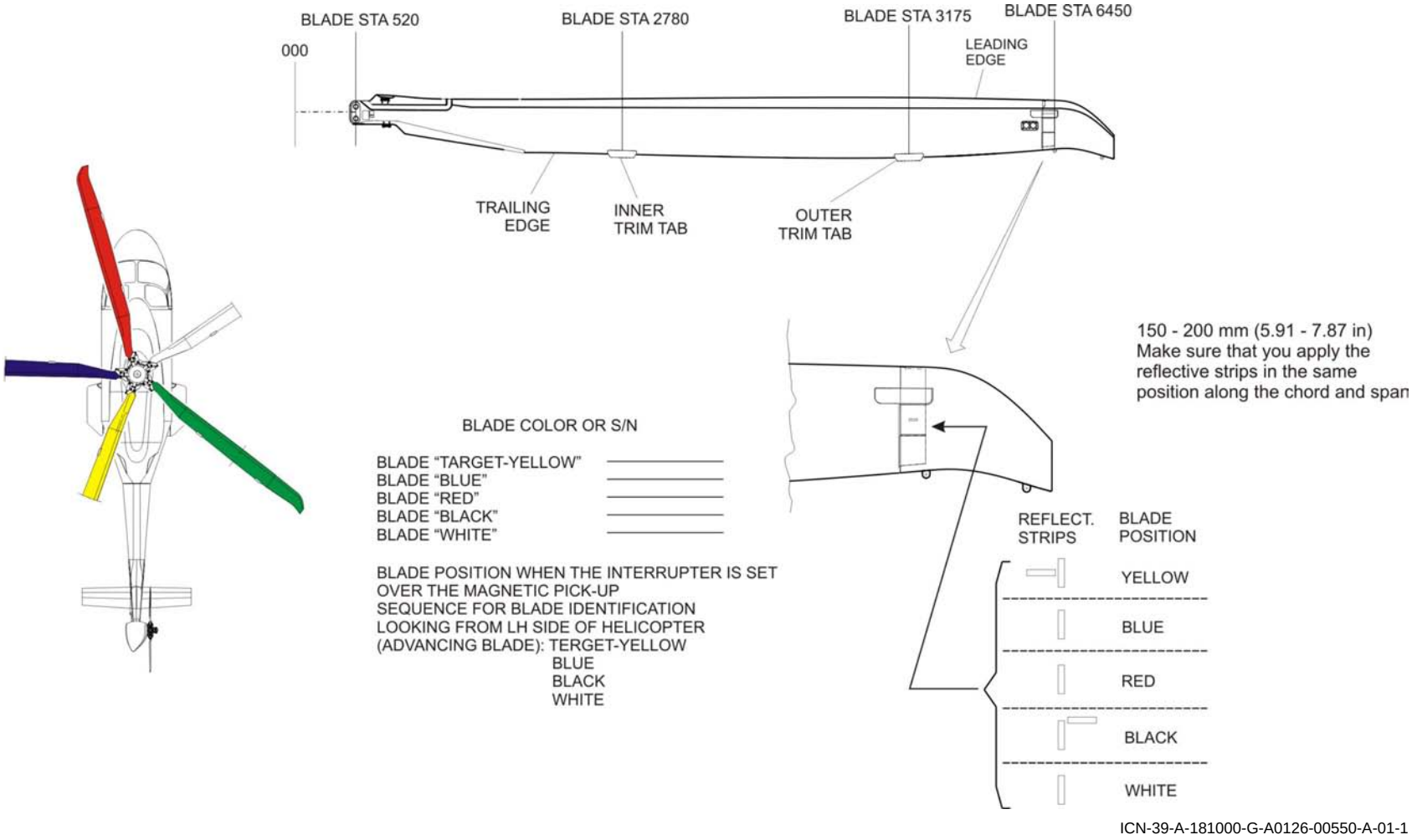
ICN-39-A-181000-G-A0126-00548-A-01-1

COCKPIT VELOCIMETERS INSTALLATION

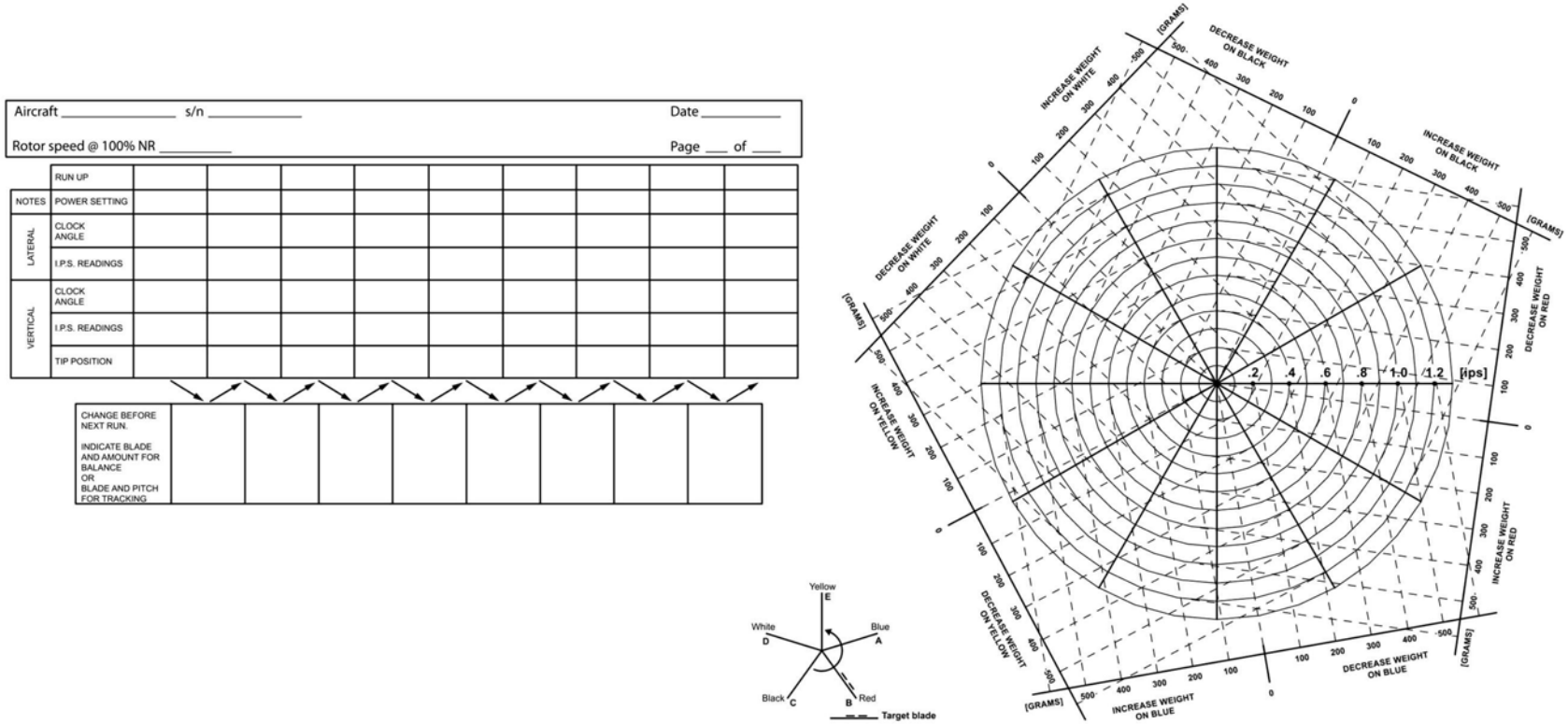


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M/R- MAGNETIC PICK UP MECHANISM



REFLECTIVE STRIPS



ICN-39-A-181000-G-A0126-00551-A-01-1

M/R BALANCE CHART

TAIL ROTOR TRACK AND BALANCE

The track and balance procedure is divided as follows:

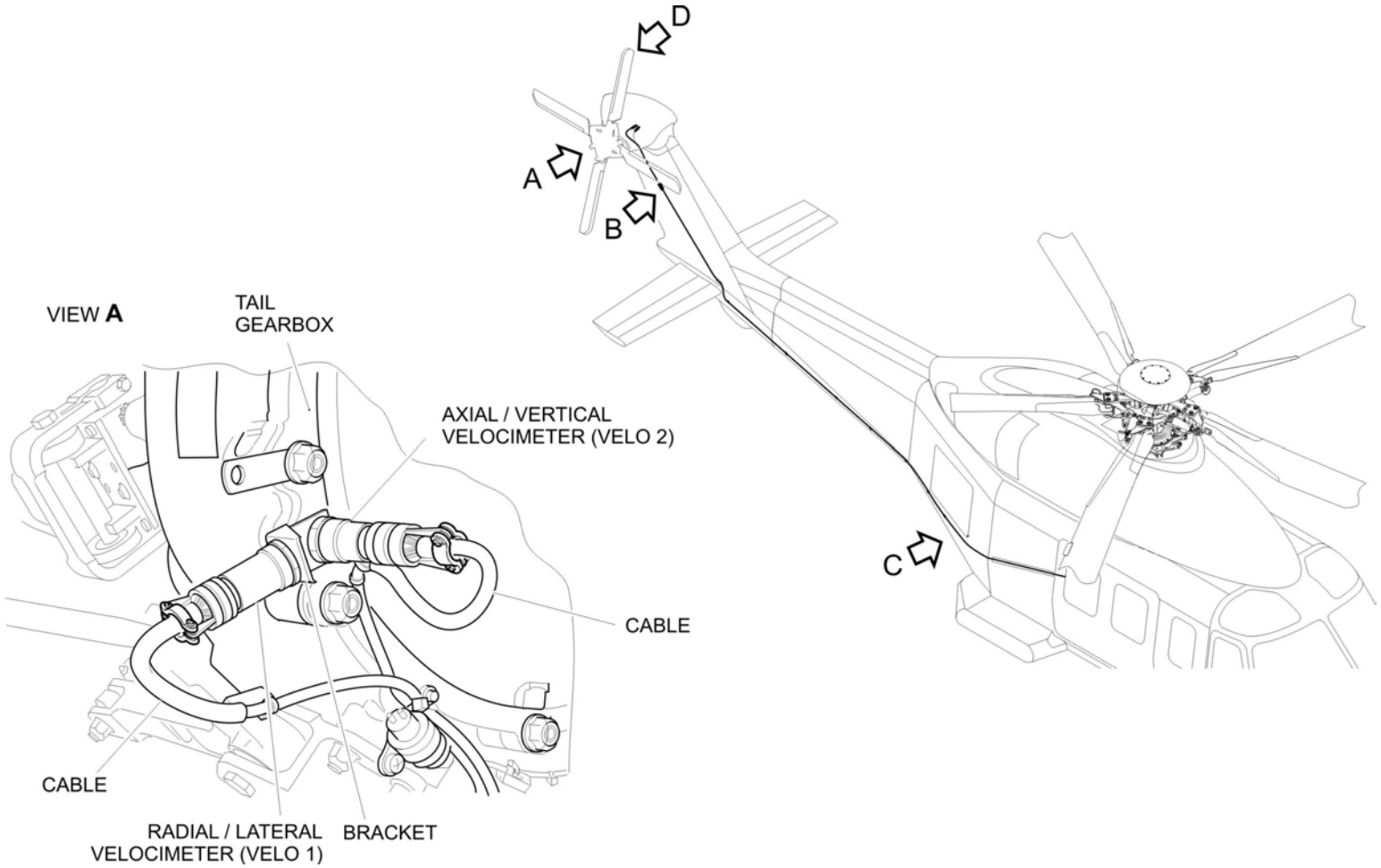
- track and balance at MPOG;
- track and balance in hover OGE:
- in flight tracking, followed, in case of important corrections of the length of the pitch links, by a dynamic balance in hover OGE.

CORRECTIONS

- The corrections for dynamic unbalance are made by adding or removing weight from blade retention bolt.
- For the tracking are made by adjusting the pitch link length.

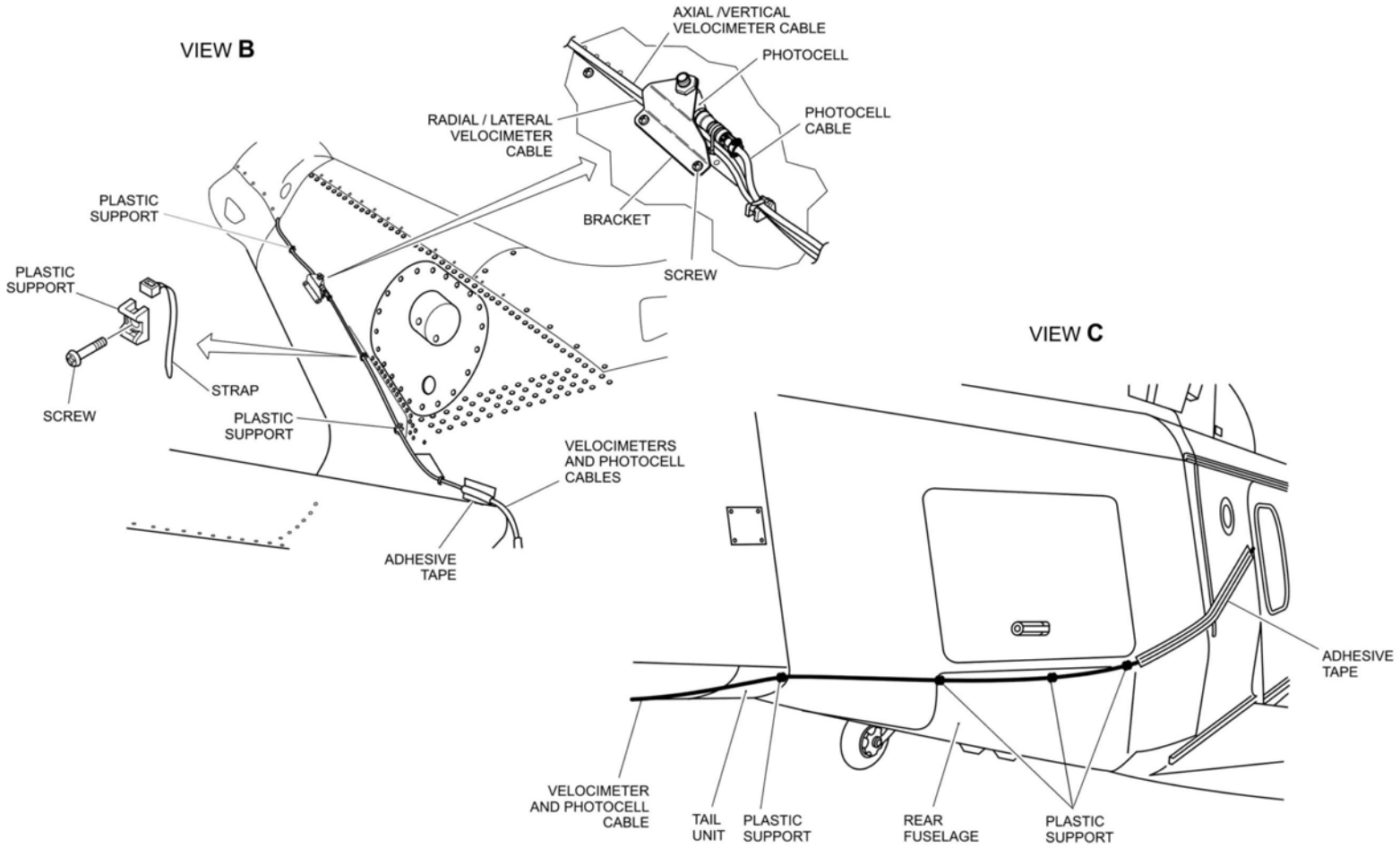
INSTRUMENT INSTALLATION

Photocell installation
Installed on the vertical fin to detect the RPM of the T/R together with a reflective strip that acts as an interrupter.
Velocimeter installation
The velocimeters are installed on the TGB, one is parallel to the axle of rotation of the T/R (Vertical one), the other is perpendicular to the T/R axle and is important to make sure that the prolongation of his axle passes through the center of the mast otherwise the signal acquired will compromise the result of the balance.



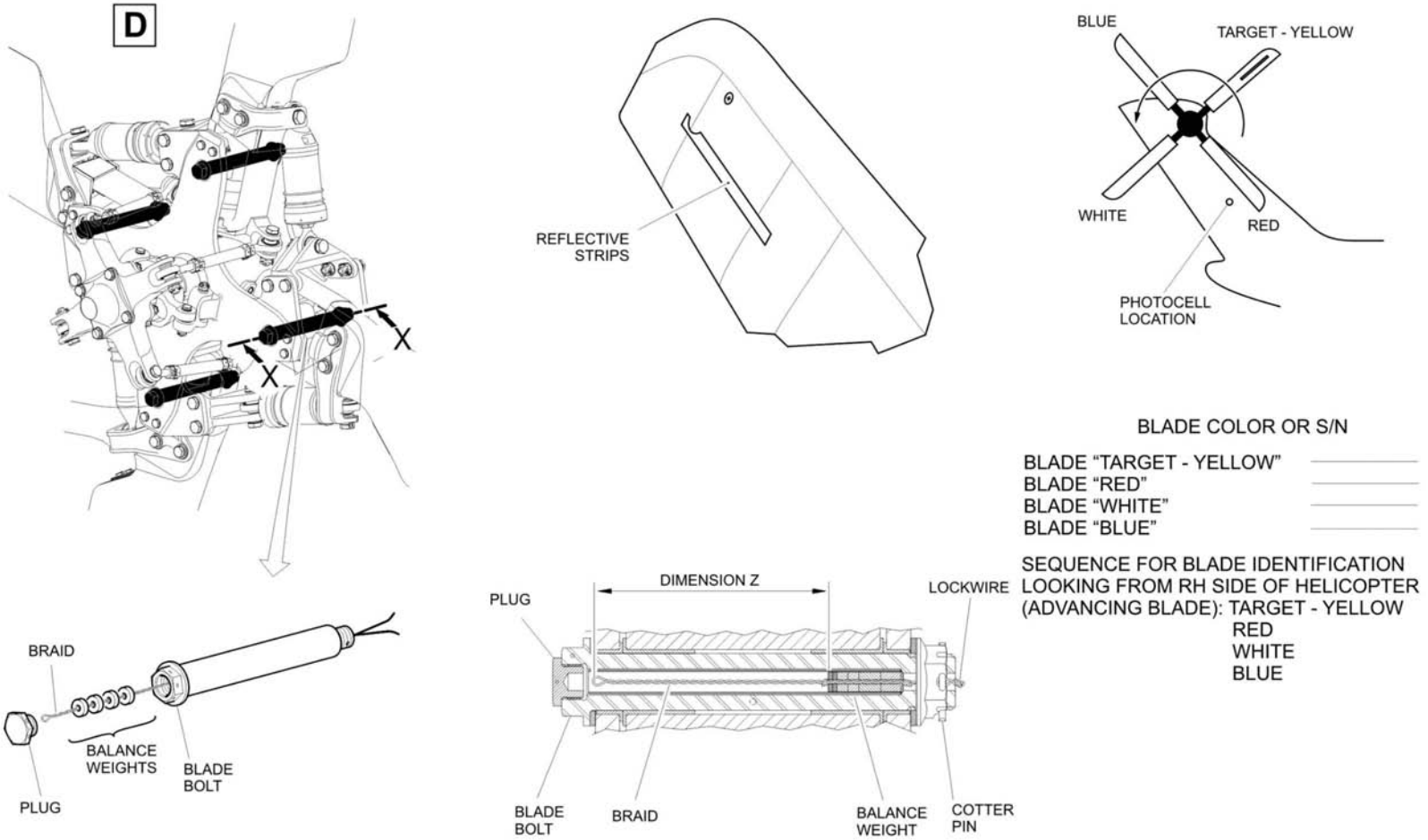
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TAIL ROTOR PHOTOCELL INSTALLATION (SHEET 1 OF 3)



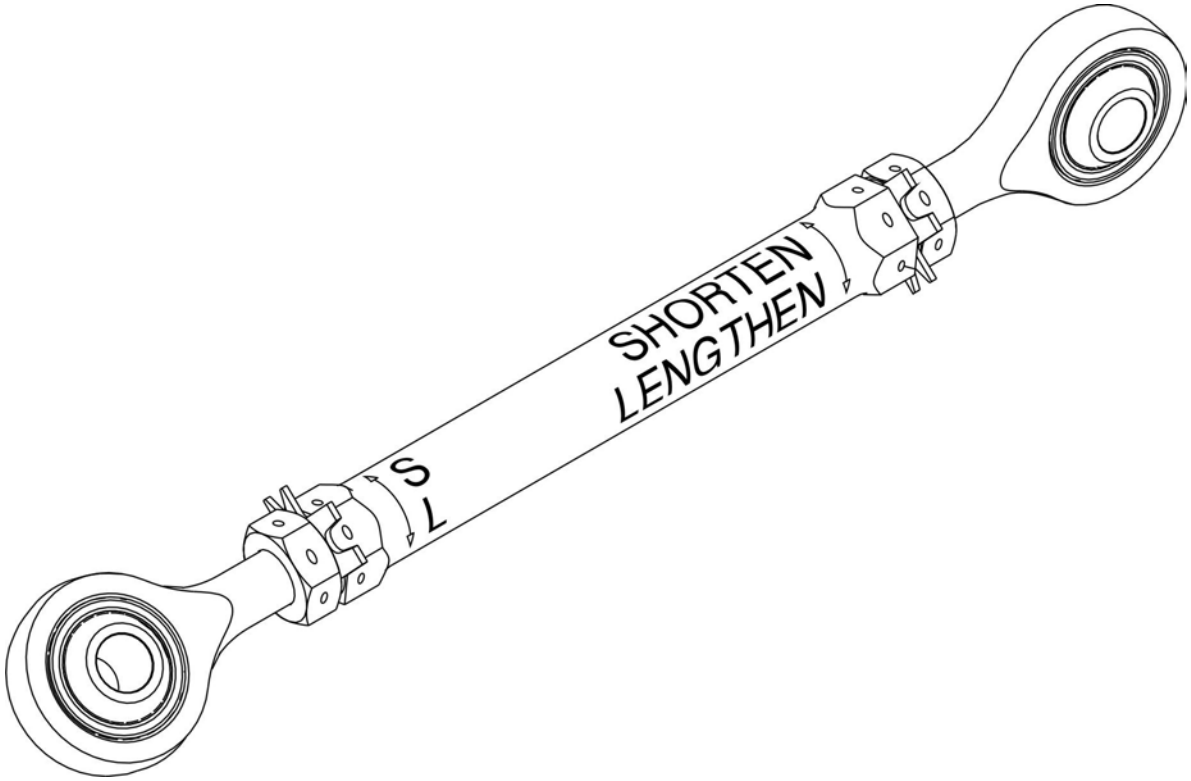
ICN-39-A-181000-G-A0126-00553-A-01-1

TAIL ROTOR PHOTOCELL INSTALLATION (SHEET 2 OF 3)



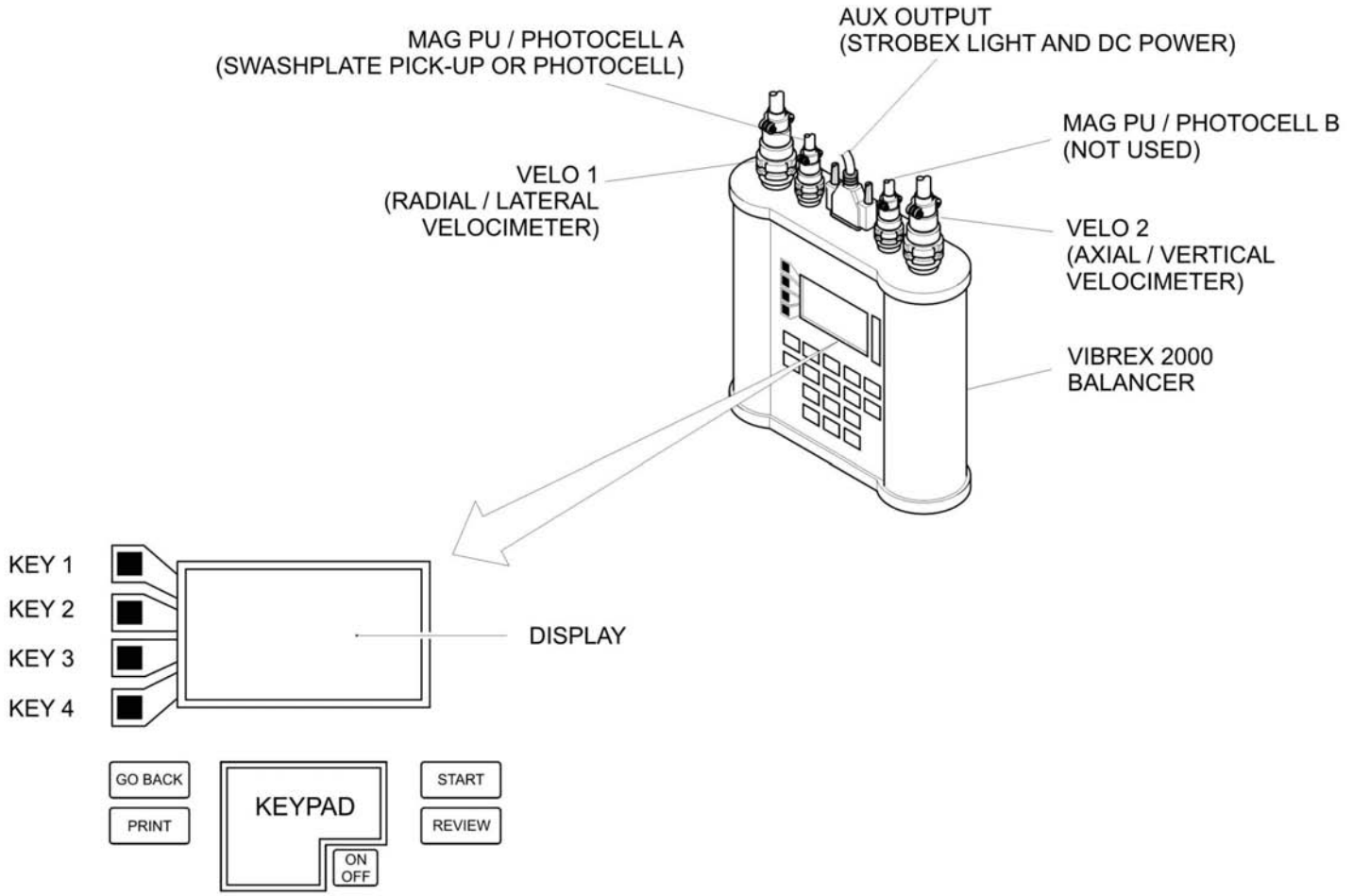
ICN-39-A-181000-G-A0126-00554-A-01-1

TAIL ROTOR PHOTOCELL INSTALLATION (SHEET 3 OF 3)



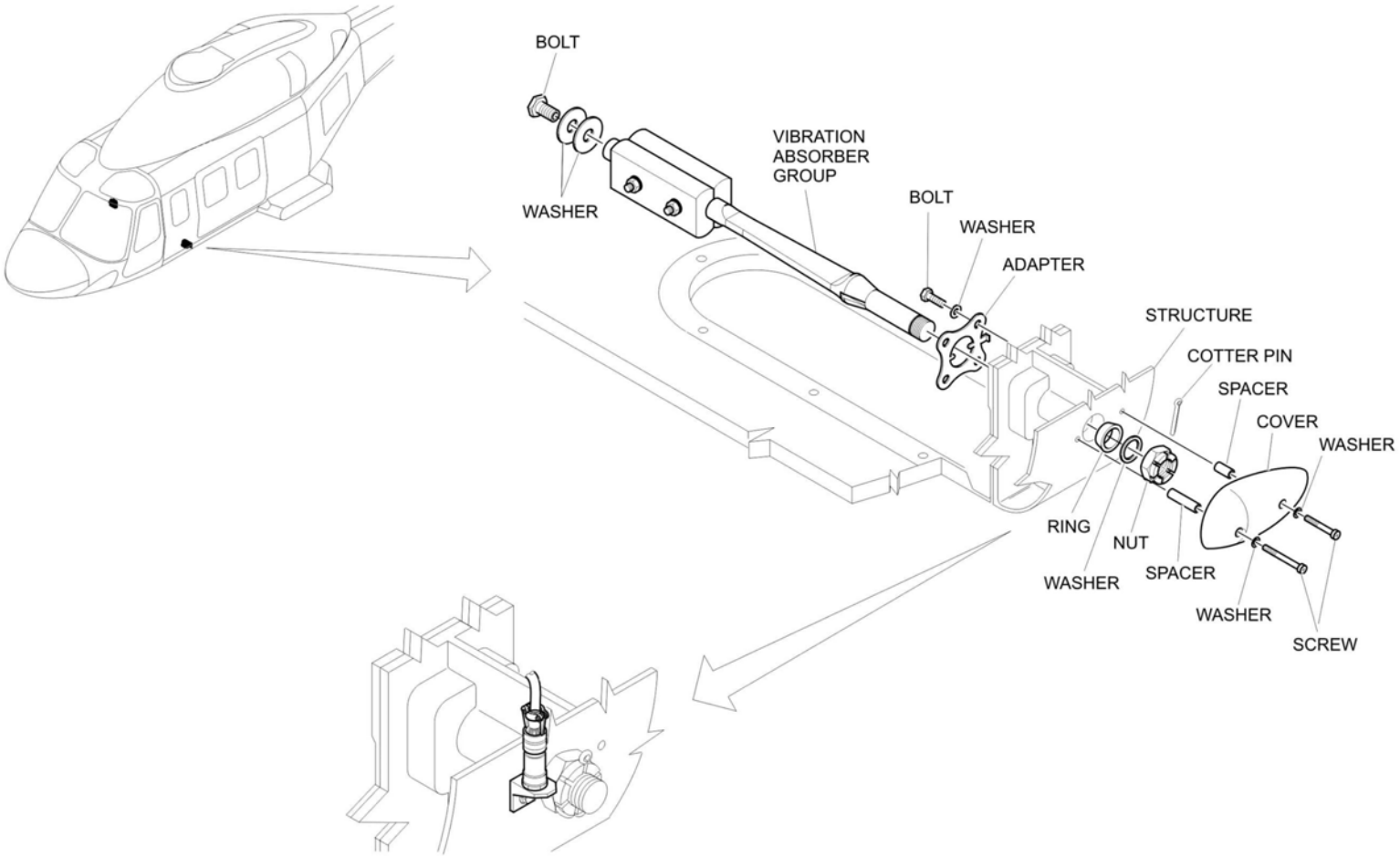
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TAIL ROTOR PITCH CHANGE LINK



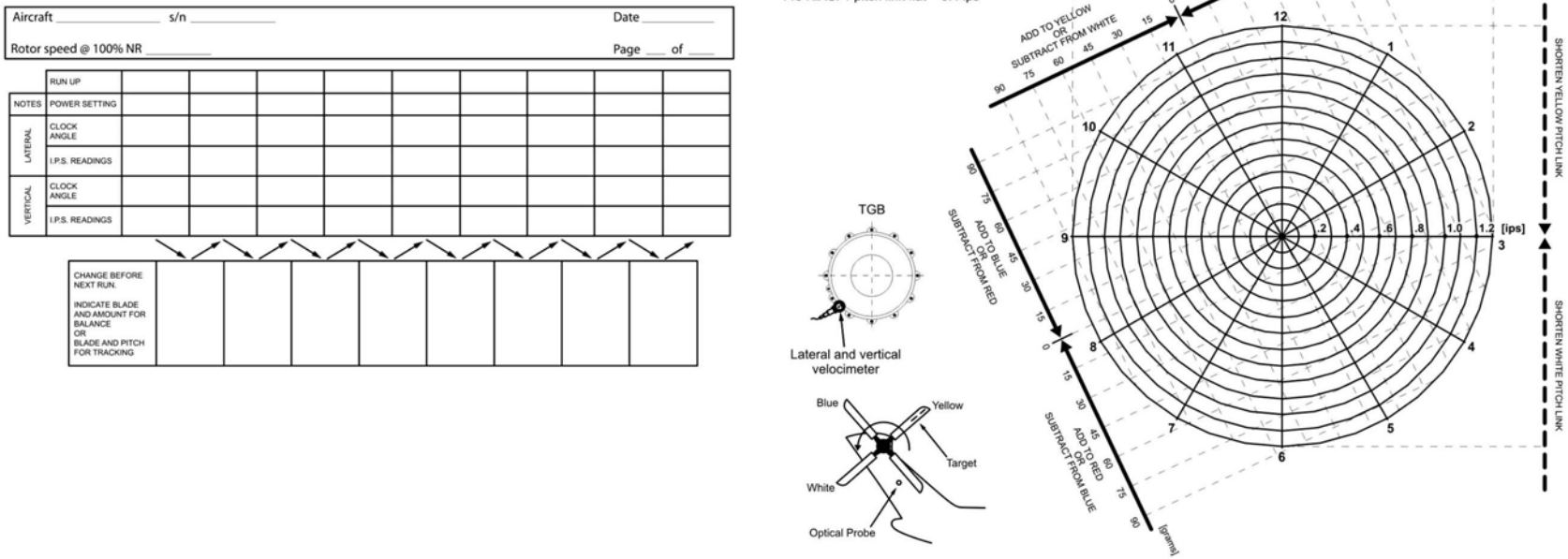
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BALANCER CONNECTION (CHADWICK VIBREX 2000)



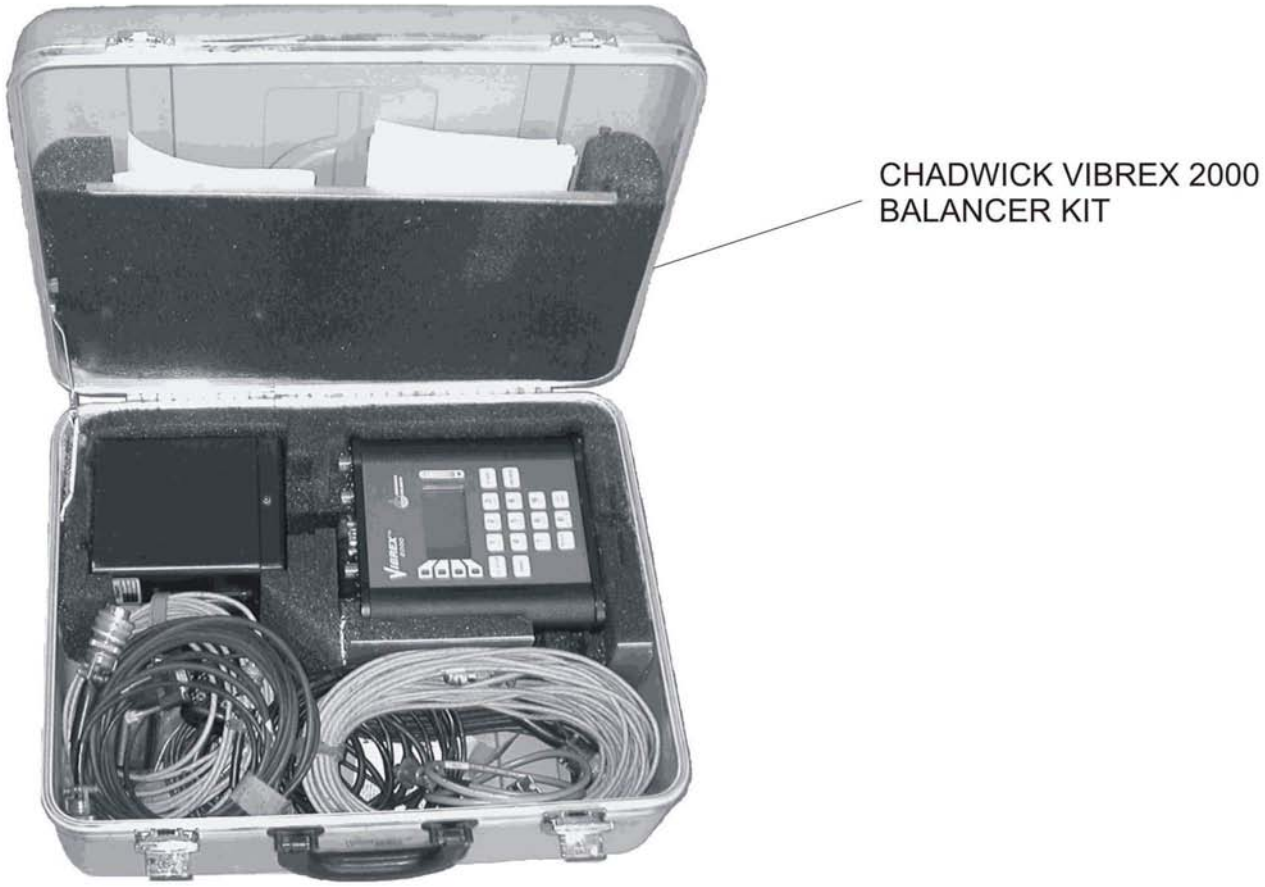
ICN-39-A-181000-G-A0126-00557-A-01-1

VIBRATION ABSORBER COMPONENTS



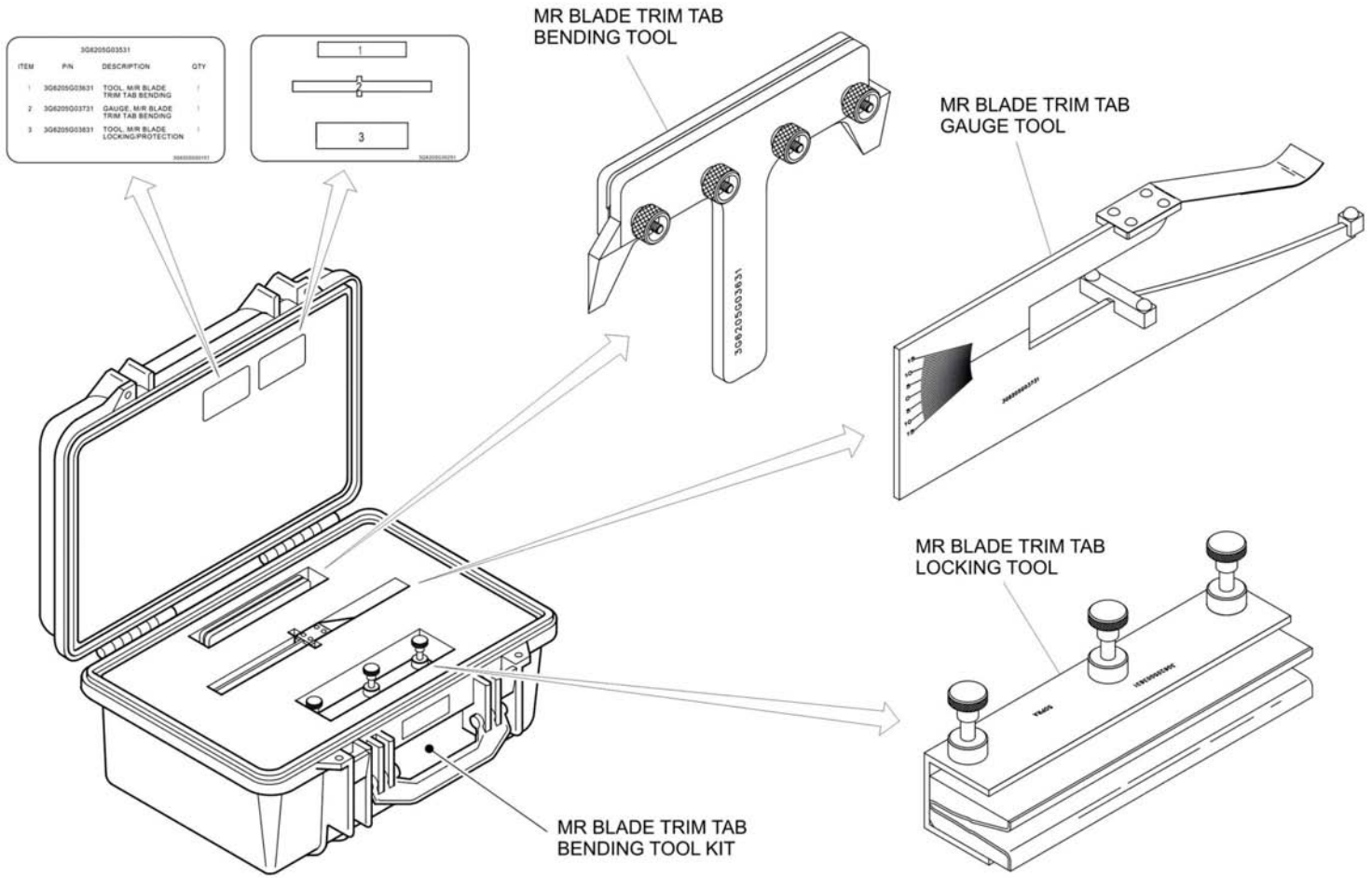
ICN-39-A-181000-G-A0126-00558-A-01-1

T/R BALANCE CHART



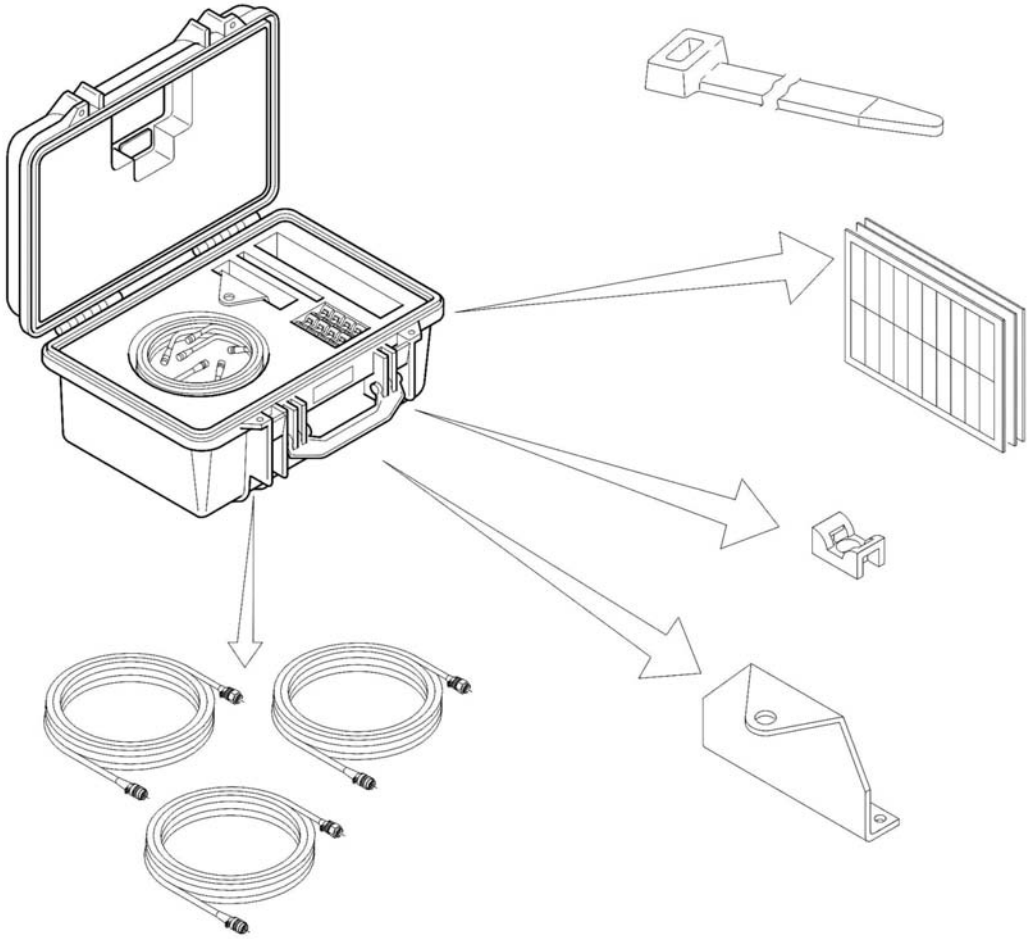
ICN-39-A-181000-G-A0126-00559-A-01-1

SPECIAL TOOLS / TEST EQUIPMENT (SHEET 1 OF 3)



ICN-39-A-181000-G-A0126-00560-A-01-1

SPECIAL TOOLS / TEST EQUIPMENT (SHEET 2 OF 3)



ICN-39-A-181000-G-A0126-00561-A-01-1

SPECIAL TOOLS / TEST EQUIPMENT (SHEET 3 OF 3)

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CHAPTER 21

ENVIRONMENT CONTROL

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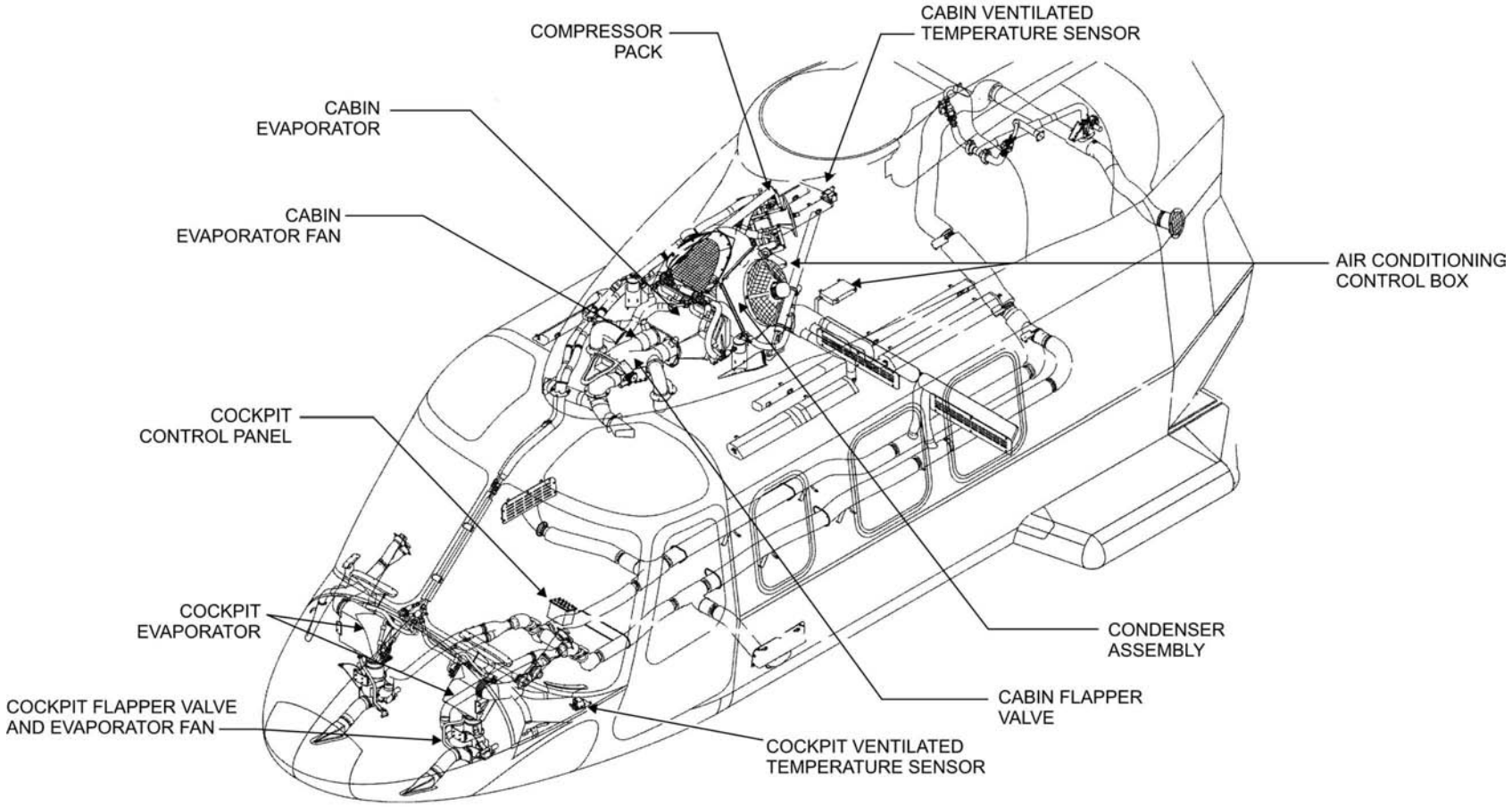
SECTION 00 - ENVIRONMENTAL CONTROL

GENERAL DESCRIPTION

The enviromental control system is composed by:

- Distribution system :
  - The cockpit ventilation;
  - The cabin ventilation.
- Heating system :
  - The air distribution;
  - The bleed ai.
- The integrated Environmental Control System (ECS) includes these primary components:

The heating system keeps the temperature in the cockpit and cabin at a satisfactory temperature. The system also removes the mist from the windshield and the cockpit windows.
The role of the system is to control the temperature ambient air as required by the crew and passenger.
As a kit added to the basic ventilation and heating fit, the air conditioning kit is required to allow maintenance of a safe level of oxygen content and prevent build-up of CO and CO_2 and to provide fresh air for removable of contamination, moisture and objectionable odurs.



ICN-39-A-210000-G-A0126-00273-A-01-1

CABIN VENTILATION GENERAL SCHEMATIC

INTERFACE

The ventilation, the heating and the integrated environmental control system are operated by the crew via the cockpit control panel. System control is as follow:

Cockpit Control Panel

The cockpit control panel is divided into two major which may be operated independently or together:

- VENT..... to control the cockpit and cabin ventilation with the flapper valves and the ventilation fans;
- COND / HEATER..... to manage the temperature of the air supplied.

There are two switches under the legend VENT

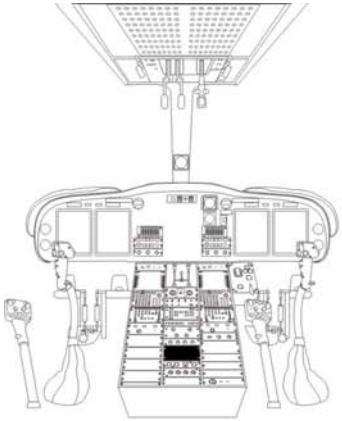
- Two rotary switches, VENT-CREW and VENT-PAX allow the pilot to control the cockpit and cabin flapper valves and fans:

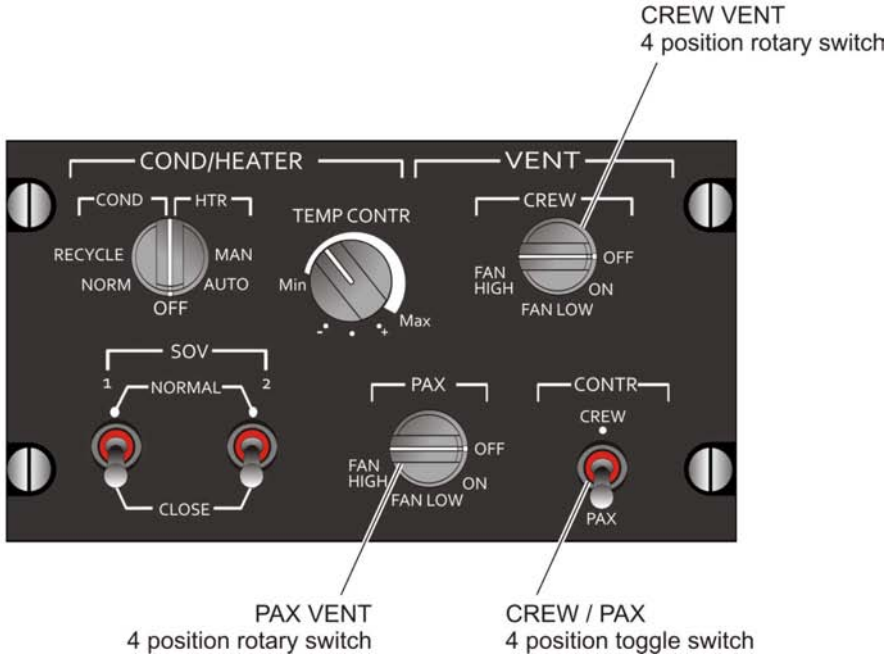
1 A 4-position rotary switch VENT-CREW controls the cockpit ventilation. OFF prevents the entry of external air by commanding the flapper valves to the recirculation position and switches OFF the fans, ON opens the flapper valves but fans are OFF (i.e. air supplied by ram air action alone), FAN LOW, and FAN HIGH control the fan speed accordingly.

2 A 4-position rotary switch VENT-PAX controls the cabin ventilation. OFF prevents entry of external air by commanding the flapper valves o the recirculation position and switches OFF the fans ON opens the flapper valves but fans are OFF (i.e. air supplied by ram air action alone), FAN LOW, and FAN HIGH control the fan speed accordingly.

NOTE A

When COND / HEATER-COND is selected to RECYCLE, both cockpit and cabin flapper are forced into the closed, or recirculation, position.





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CABIN VENTILATION INTERFACE

DESCRIPTION AND OPERATION OF MAJOR COMPONENTS

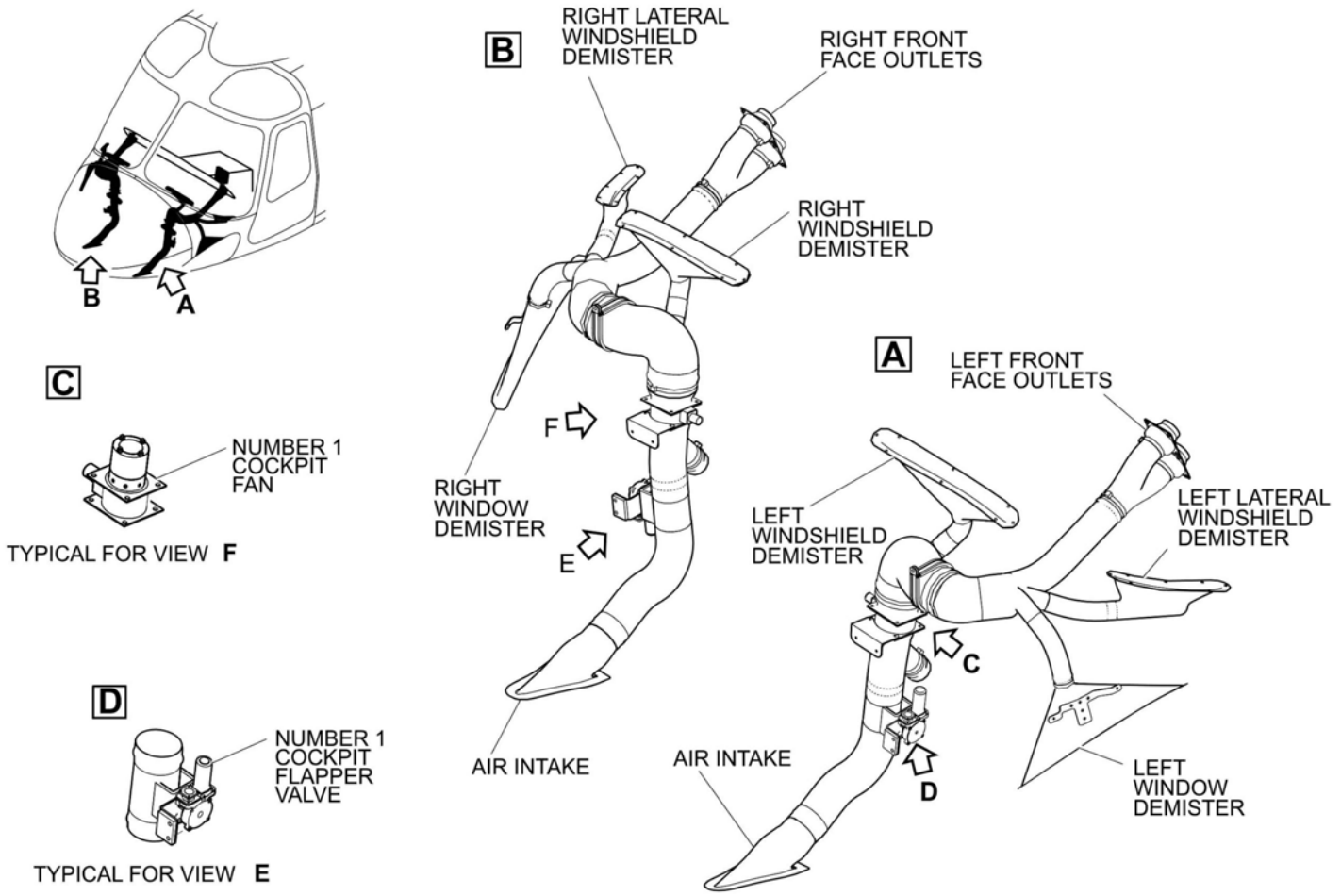
VENTILATION SYSTEM

Cockpit Ventilation

Two isolated circuits supply the airflow to the pilot and copilot stations. The circuits are the left circuit for the copilot, and the right circuit for the pilot. The components included in each of the two circuits are the same. The external ram air enters the two circuits through the air intakes installed under the nose section of the helicopter. The position of the air intakes is the farthest possible from the engine exhausts. Thus the exhaust gases cannot go into the ventilation system.

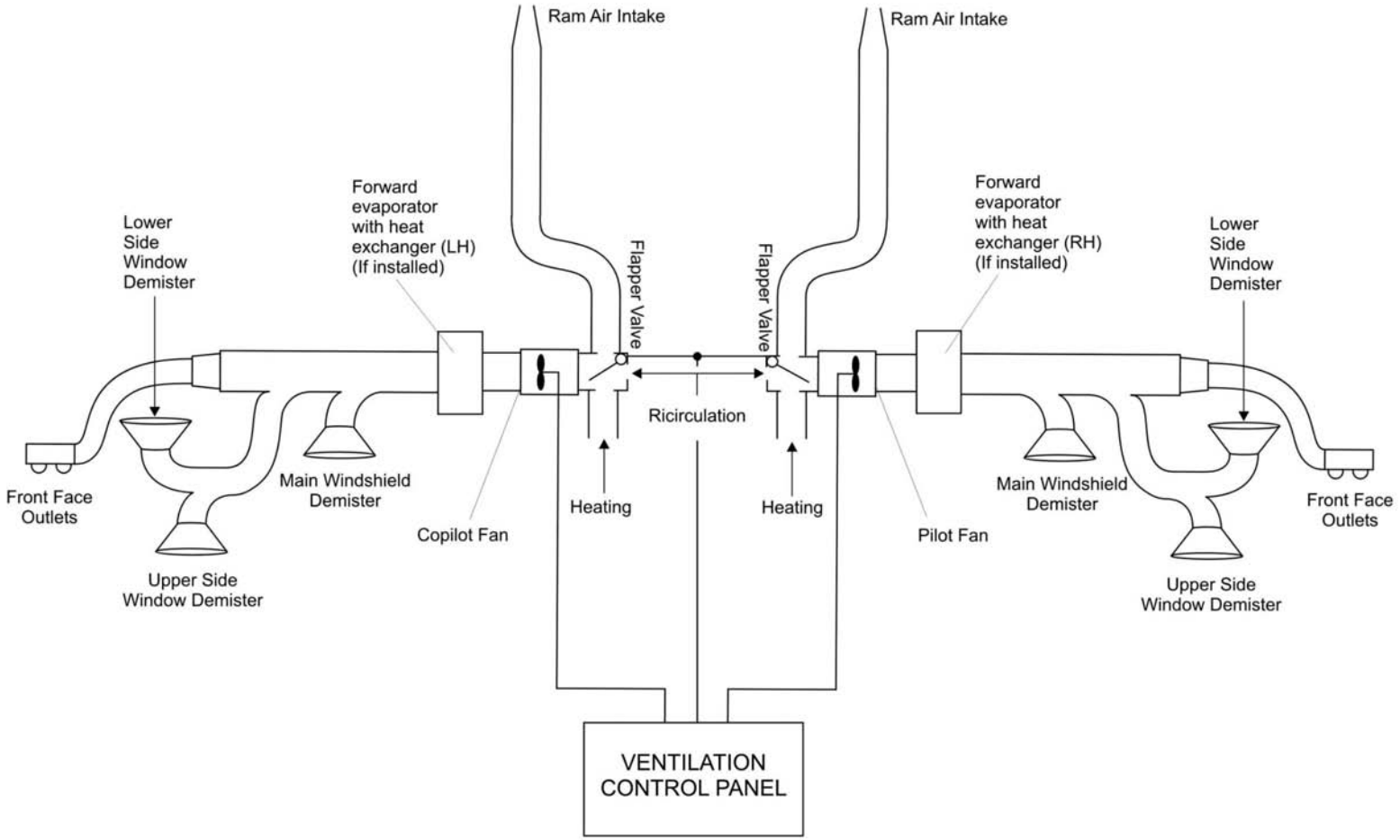
Cabin Ventilation

The air that flows to the cabin goes into the system through an air intake installed in the forward sliding cowling. The position of the air intake is the farthest possible from the engine exhaust, thus the exhaust gases cannot go into the ventilation system. The cabin flapper valve controls the airflow. The valve is operated electrically and the crew sets it to open or close with the PAX selector. The selector is installed on the ventilation control panel in the cockpit. Downstream of the flapper valve, the air flows through a hose and a duct to a water separator, and then into a manifold. From the manifold, the air is sent to the cabin air inboard and outboard diffusers.



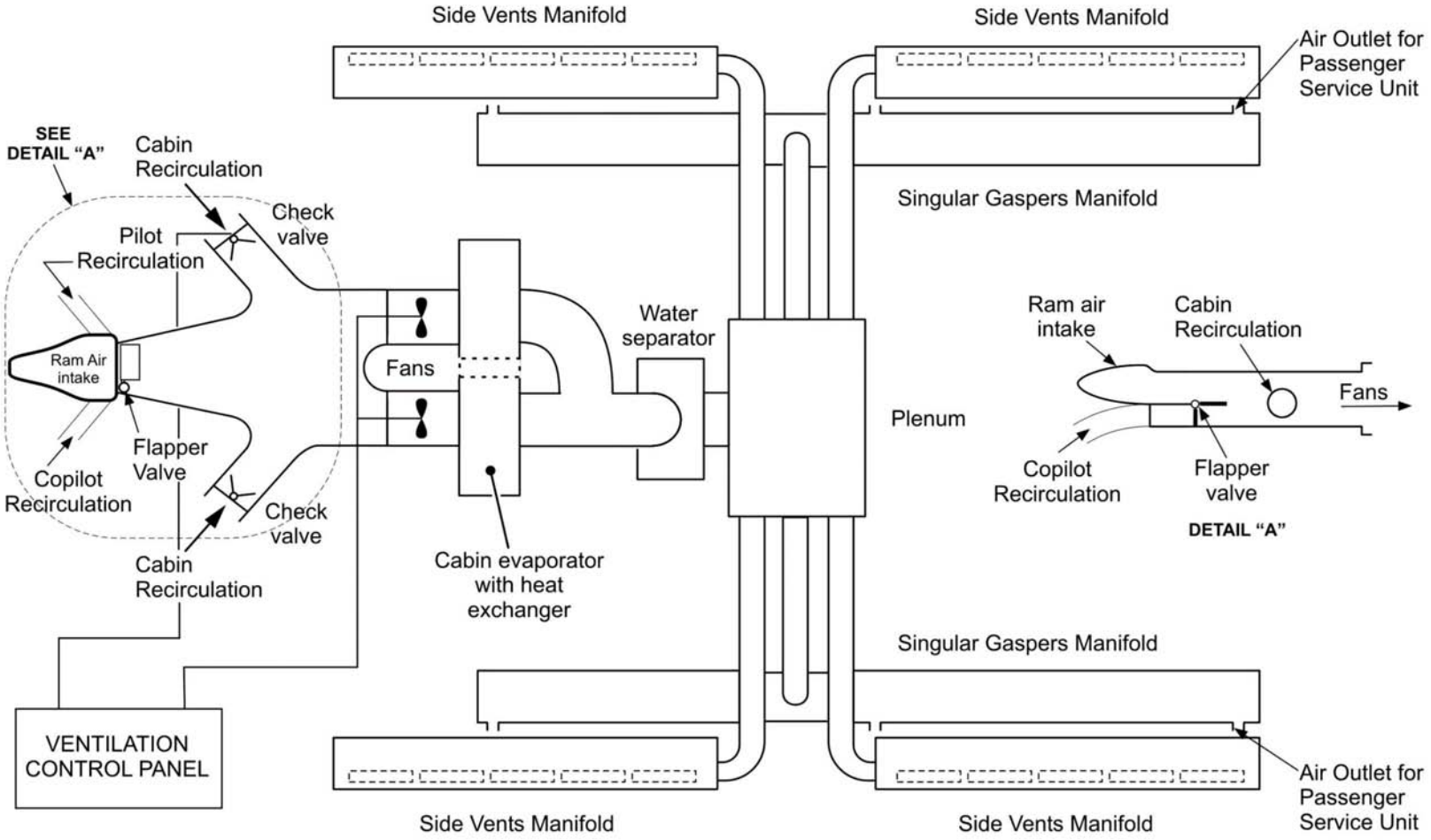
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PILOT VENTILATION SYSTEM



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PILOT VENTILATION SYSTEM SCHEMATIC



ICN-39-A-210000-G-A0126-00277-A-01-1

CABIN VENTILATION SCHEMATIC

HEATING SYSTEM

Air Distribution

The air distribution distributes the mixed air from the jet pump flows to the cockpit and cabin through flexible and rigid ducts.
The airflow from the jet pump goes to the two lines installed on each side of the cabin under the floor. Each line gets half of the airflow. Then the air enters the cabin through four diffusers that are installed near the floor. Downstream of the cabin air diffusers, the air flows into the cockpit through two air outlets that are installed on the left and right side of the pedestal, near the floor and adjacent the pilot and copilot pedal sets.
The heating control panel in the cockpit and the heating control box let the flight crew operate the heating system as necessary.
The heating control box also monitors the system and sends warnings to the flight crew when the air temperature becomes too high.
A duct temperature sensor and an overheat temperature switch give the data necessary for these functions. The switch is installed on the mixing unit of the venturi duct (jet pump).

Bleed Air

The air from the bleed port of each engine flows in the related stainless-steel pipe to one of the shut-off valves. There are two pipes in the helicopter, one for each bleed port, and their routing is through the engine compartments. Thus, these sections of the pipes go through the fire zone.
The bleed airflow is only 2 percent of the total engine airflow. The size of the opening on each of the flanges of the bleed pipes installed on the engines prevents higher bleed airflow.
Flexible sections in each pipe absorb the thermal expansion and the vibrations of the engines.
Downstream of the shut-off valves and out of the fire zone, each pipe goes to a check valve in the rear upper deck. The check valves prevent the flow of the bleed air from one engine to the other.
Downstream the check valves, the two pipes connect and become one stainless steel pipe. This pipe has a flexible section that absorbs the thermal expansion.
Through this pipe, the hot and high-pressure bleed air goes to a temperature control valve. This valve controls the quantity of bleed air that flows to the jet pump installed downstream of it. The hot bleed air and the external air from an inlet at the top of the cabin wall mix in the jet pump.
A secondary injector also supplies bleed air to the jet pump. The airflow from the secondary injector bypasses the temperature control valve when it is in failure.

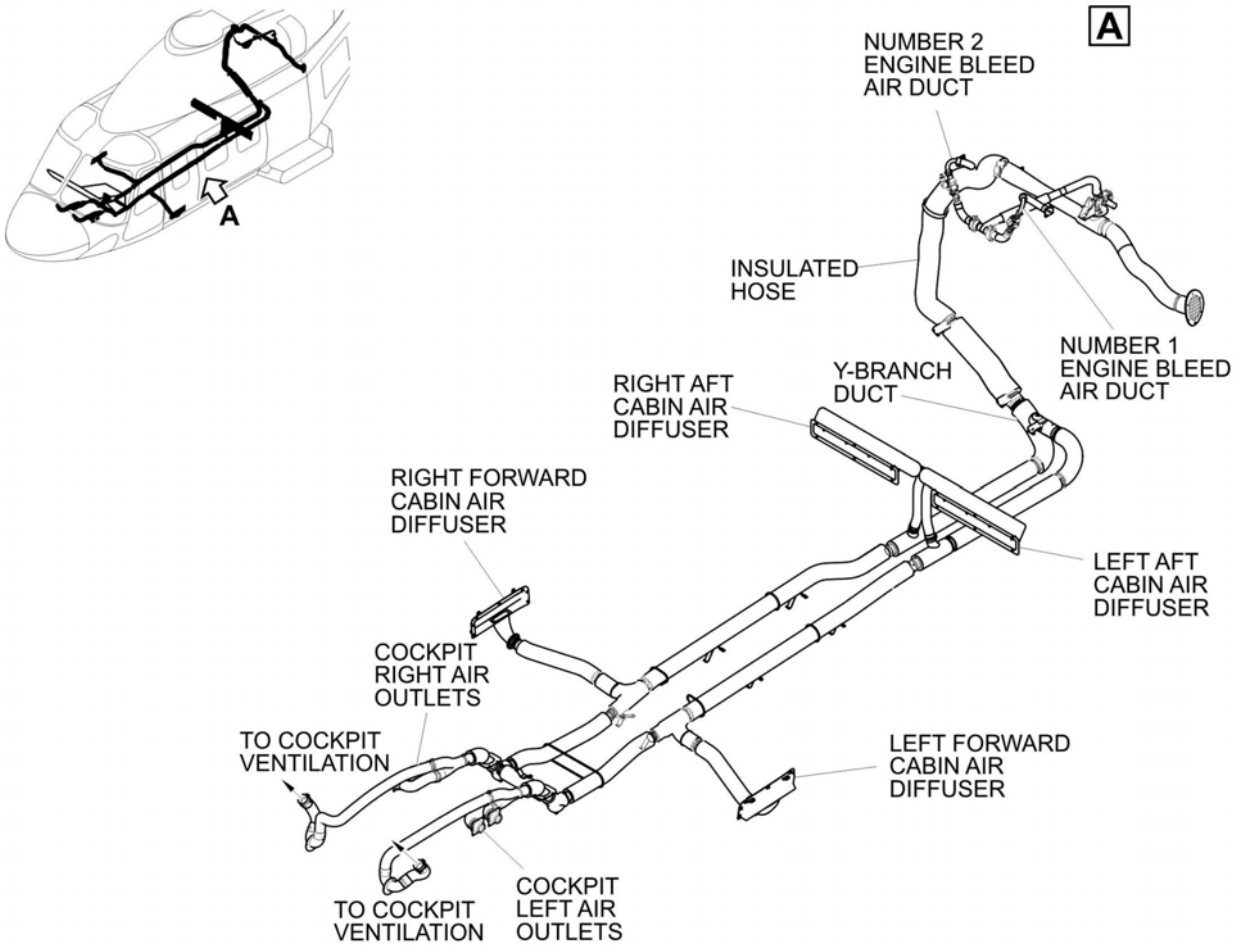
SYSTEM OPERATION

The heating system operates in those modes: the automatic mode and the manual mode.
In the automatic mode the heating control box controls the temperature of the mixed airflow. To do this, it compares the temperature of the mixed air supplied by the temperature sensor with the temperature set on the heating control panel.
The temperature control valve opens or closes to give the correct temperature. This temperature is the temperature at which the difference between: – The temperature the flight crew has set on the cockpit control panel and – The temperature measured at the duct temperature sensor at the exit of the venturi duct is minimum.
When the AUTO mode is set (HTR selector set on AUTO), the flapper valves of the cockpit and cabin ventilation sub-systems are closed. But the flight crew can use the related selector to manually operate them.
If the temperature sensor senses a too high temperature in the duct, the temperature switch operates and closes the Number 1 and Number 2 shut-off valves.
The heater control panel sends the HEATER AUTO MODE command to the ENG FIRE EXTING control panel before it sends it to the heater control box.

This disconnects the heating system if, on the ENGINE FIRE EXTING control-panel, the ENG 1 and ENG 2 are at set at ARM.
In the manual mode (HTR selector set on MAN), the heating control box is overridden. Then the TEMP CONTR selector, on the heating control panel, controls the temperature control valve.
This lets the crew change the position of the temperature control valve as necessary to have a direct control of the temperature of the air that goes in the cockpit and in the cabin. This is to remove the mist from the windshield and cockpit windows if necessary.
The Primary Flight Display (PFD) and the Multifunction Display (MFD) show the related indications for the flight crew.
The MFD shows, on the CAS window, the caution and advisory messages that follow:

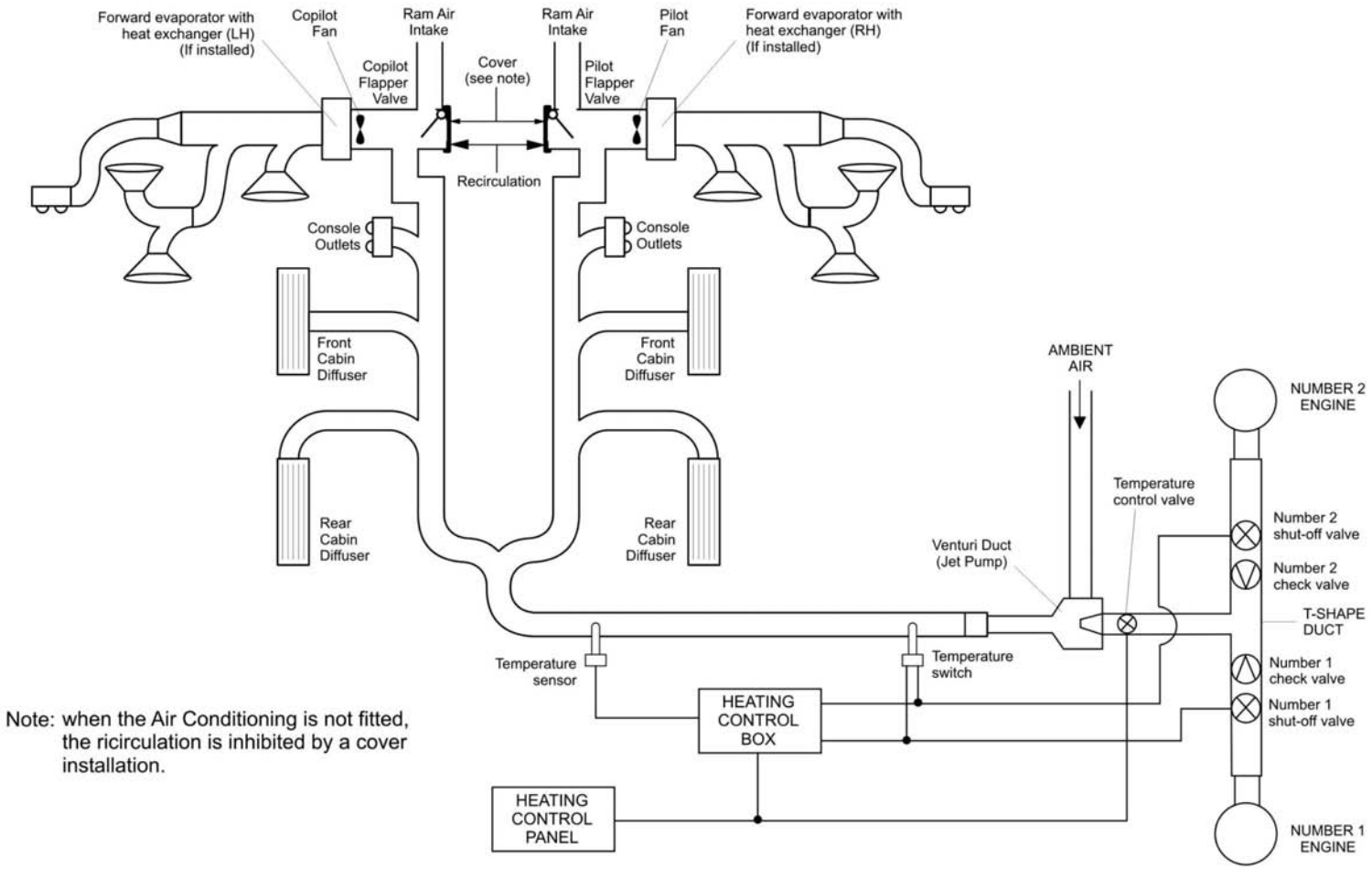
- HEATERFAIL..... The heating system is unserviceable
- HEATER VENT..... The Heating system ON.

The indications shown on the PFD and the MFD are sent from the Number 1 and Number 2 Modular Avionics Units (MAU1 and MAU2). This because they have an interface with the heating system.



ICN-39-A-210000-G-A0126-00278-A-01-1

HEATING SYSTEM



Note: when the Air Conditioning is not fitted, the ricirculation is inhibited by a cover installation.

ICN-39-A-210000-G-A0126-00279-A-01-1

HEATING SYSTEM

INTEGRATED ENVIRONMENTAL CONTROL SYSTEM

The functions of the integrated environmental control system are operated by the flight crew from the ECS cockpit control panel installed in the cockpit.

The ECS cockpit control panel is divided into two parts. One part has the label VENT, and gives control of the cockpit and cabin ventilation. The other has the label COND/HEATER, and lets the flight crew set the temperature of the air in the cockpit and the cabin.

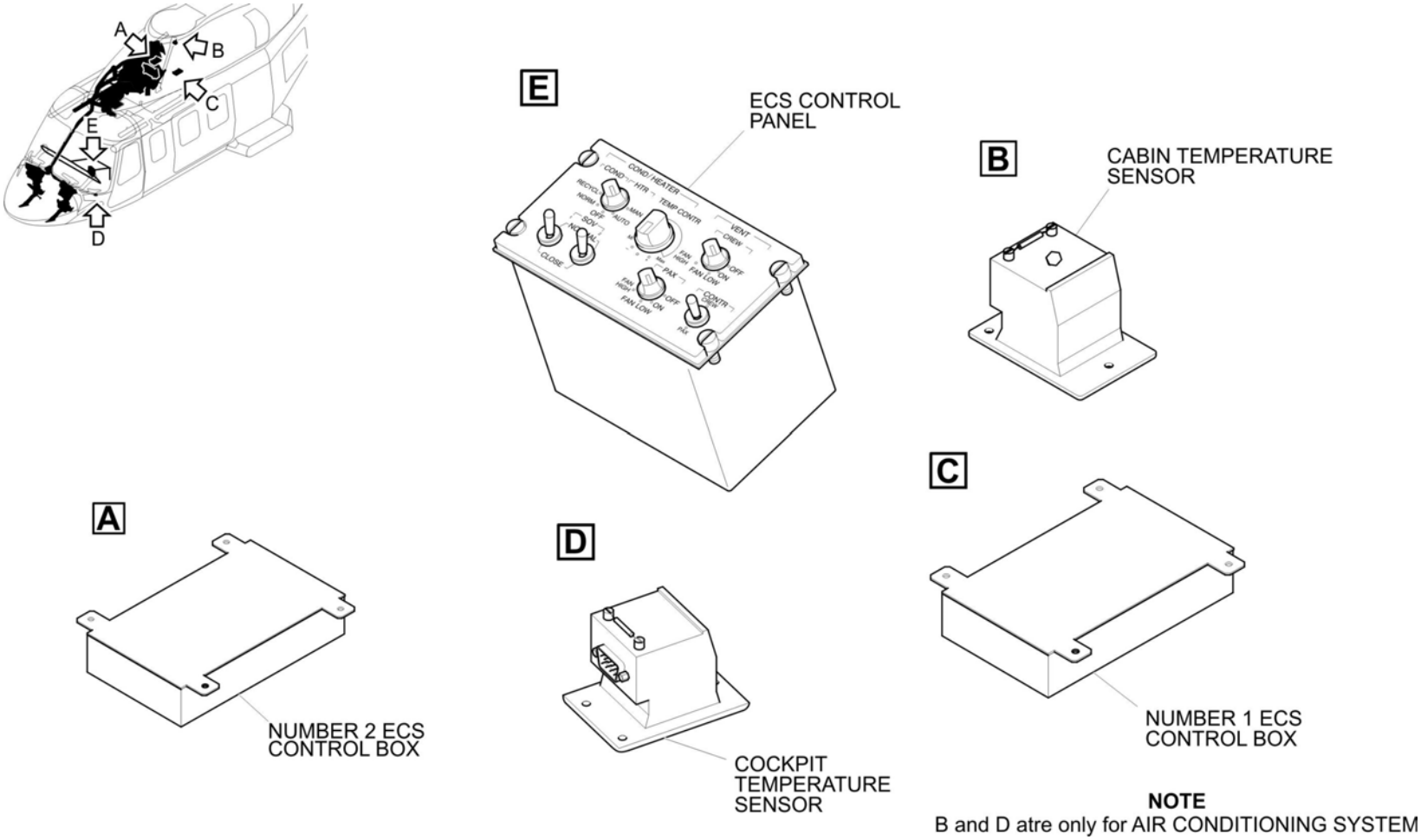
The VENT-CONTR toggle switch lets the flight crew give the control of the cabin air conditioning to the cabin control panel.

The cabin control panel controls the cabin functions when the two-position VENT-CONTR switch is at PAX. When the switch is at CREW, the cockpit control panel has the control of the cabin air conditioning.

Then there are the four-position VENT-CREW selector and the four-position VENT-PAX selector. These selectors let the flight crew control the related flapper valves and fans.

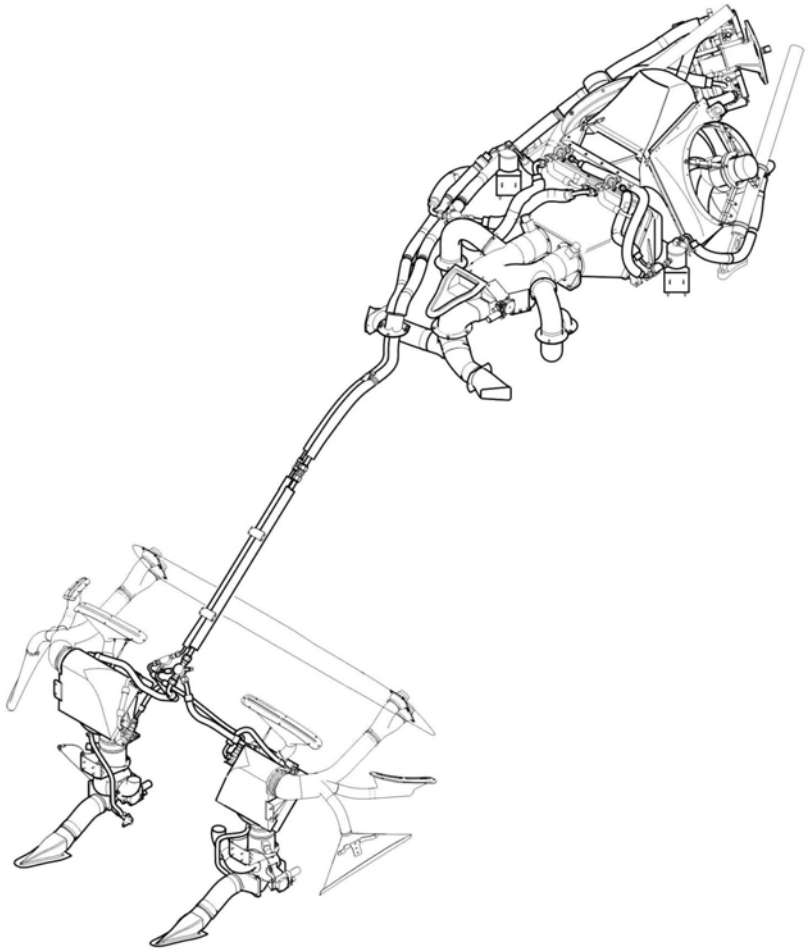
The VENT-CREW selector has the OFF, ON, FAN LOW and FAN HIGH positions. When the selector is at OFF the flapper valves are closed. The flow of external air into the cockpit is prevented, but not fully: a very small quantity of external air continues to flow around the internal flap of each valve. Also, the cockpit fans are off. When the selector is at ON the flapper valves are open, but the fans remain off. The external air that goes into the cockpit is ram air. When at FAN LOW or FAN HIGH, the selector controls the fans to turn at the related speed. Each axial-flow fan blows the external or re-circulation air to the related cockpit evaporator where the temperature of the air is decreased as necessary. Each fan has an under-speed detection device that prevents damage if the rotor becomes locked, and a thermal protection device. The thermal protection device opens the related circuit when the temperature is 120˚C to prevent damage to the electronics.

The VENT-PAX selector has the OFF, ON, FAN LOW and FAN HIGH positions. When the selector is at OFF the flapper valve is closed. The flow of external air into the cabin is prevented, but not fully: a very small quantity of external air continues to flow around the internal flap of the valve. Also the cabin fans are off. When the selector is at ON the flapper valve is open, but the fans remain off. The external air that goes into the cabin is ram air. When the selector is at FAN LOW or FAN HIGH, it controls the fans to turn at the related speed. Each axial-flow fan blows the external or re-circulation air to the cabin evaporator where the temperature of the air is decreased as necessary. Each fan has an under-speed detection device that prevents damage if the rotor becomes locked, and a thermal protection device. The thermal protection device opens the related circuit when the temperature is 120˚C to prevent damage to the electronics



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INTEGRATED ENVIRONMENTAL CONTROL SYSTEM



ICN-39-A-210000-G-A0126-00281-A-01-1

INTEGRATED ENVIRONMENTAL CONTROL SYSTEM

MONITORING AND DIAGNOSTIC

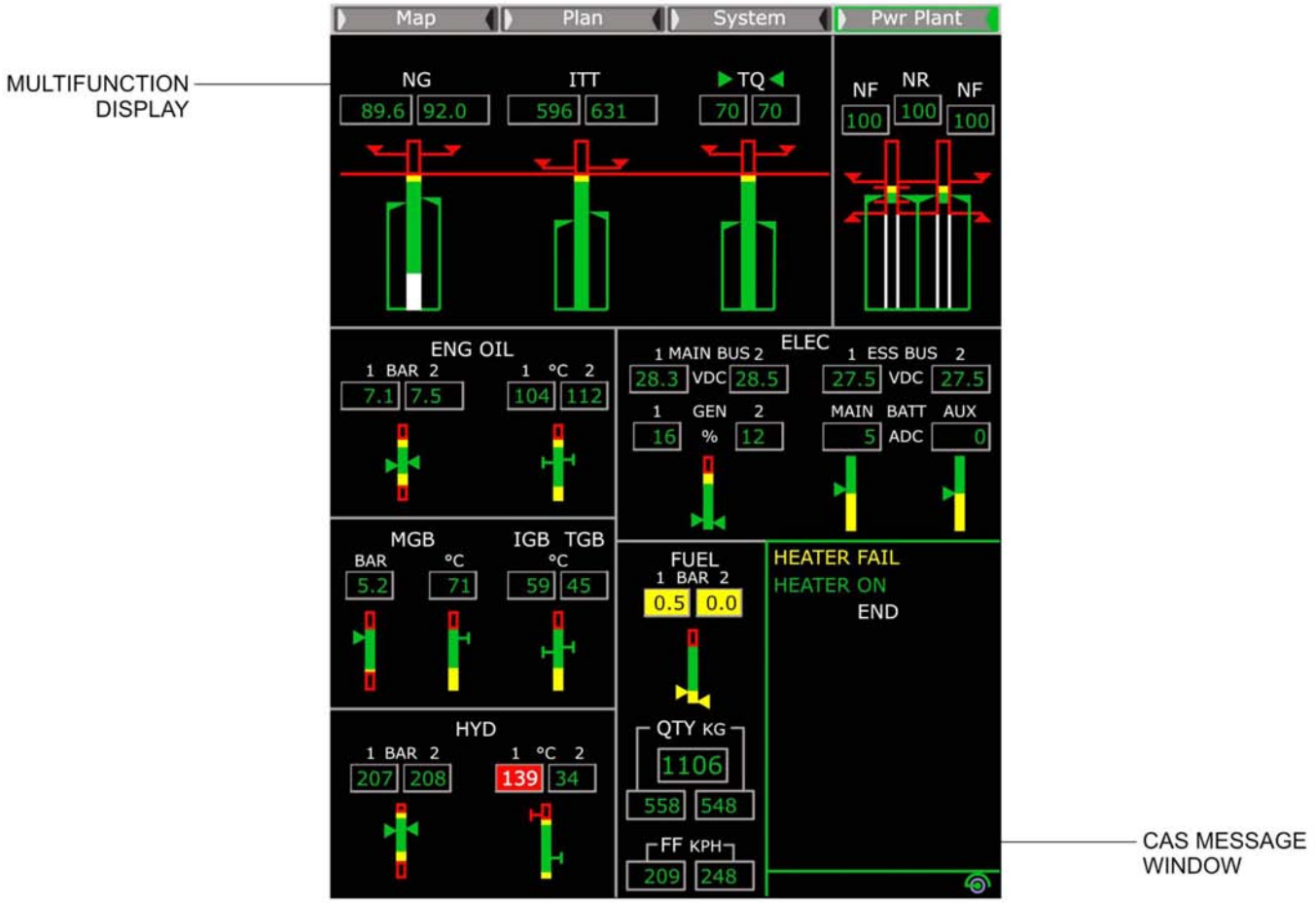
Displays

The primary Flight Display (PFD) installed in the instrument panel in front of the pilot and the Multifunction Display (MFD) installed in the central instrument panel, show the parameters of ECS operation and related crew alerting system (CAS) messages. ECS status is communicated to the flight by Caution and Advisory messages displayed on the MFD.

Cursor Control Device (Ccd)

The CCD Panel, installed in the central cockpit console, controls the selection of different pages of information on the MFD. The CCD is a controller that interfaces between the avionic system and the Displays Unit (DU) and is used by the DU processing for cursor positioning, scrolling of crew alerting display, display range selection and control of other display functions. The duplex modular avionics units (MAU 1 and MAU2) receive information from the air conditioning control system and communicate it to the CCD.
The following tables shows the CAS messages advising system status.

CAUTION MESSAGE	SYSTEM MALFUNCTION
FWD COND FAIL	Failure of the cockpit air conditioning system. May be caused by refrigerant over-pressure, refrigerant under pressure, low evaporator air temperature or failure to control internal ambient temperature as commanded by the control panel.
AFT COND FAIL	Failure of the cabin air conditioning system. May be caused by refrigerant over pressure, erfrigerant under pressure, low evaporator air temperature or failure to control internal ambient temperature as commanded by the control panel.



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INDICATING

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SECTION 20 - DISTRIBUTION

DESCRIPTION

The role of the Environmental Control System (ECS) is to distribute air, to provide comfortable conditions for the crew members and the passengers.
This is achieved by four systems which are:

- Cockpit Ventilation system pilot and copilot
- Cabin Ventilation system
- Heating system
- Air Conditioning system

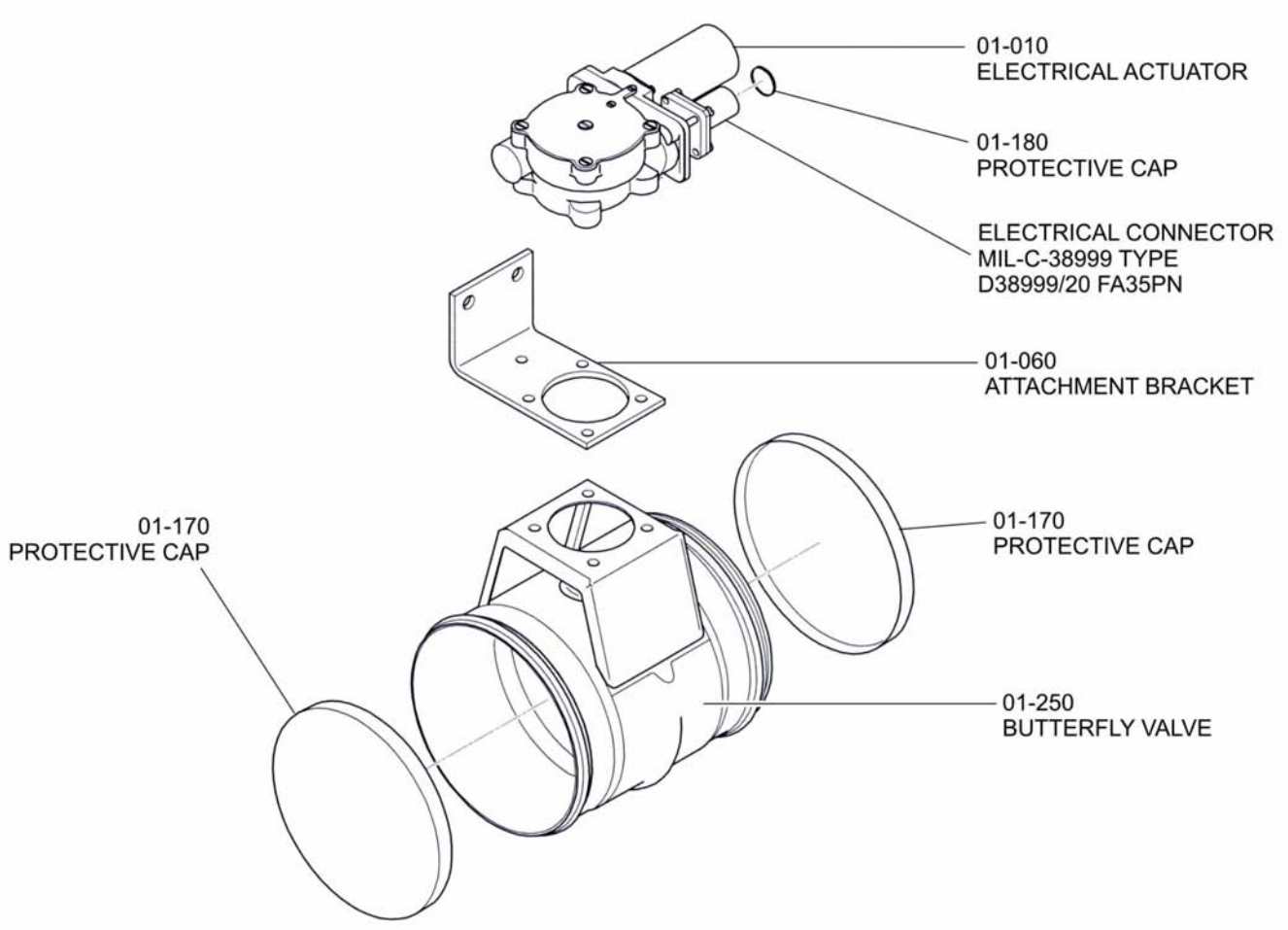
COCKPIT VENTILATION SYSTEM – DESCRIPTION

For the cockpit, two separate systems provide air for pilots.
Outside ram air enters the system through two air intakes positioned in the nose section.
Outside ram flow is controlled by an electrically actuated flapper valve, set by the crew with a 4 position rotary switch on the control panel.
Forced air for ventilation or defogging is provided by two speed axial fans, one per system.
The blowers are manually controlled by the crew.
Air is then distributed to the different outlets, main windshield and side windows diffusers.
Pilot and copilot ventilation systems are independent and arranged symmetrically. Each of them is composed of:

- A Ram Air Intake located in the lower part of the cockpit.
- A Flapper Valve, installed at the bottom of the instrument panel, forward of the control pedals.
- An Electrical fan for forced air operation, installed in aluminium casing. Each Electrical fan includes an impeller and a shaft. The impeller is made of ULTEM 2300/2310 (30% glass-filled polytherimide) and the shaft is made from stainless steel.
- Five outlets:
  - Two adjustable face outlets on the instruments panel.
  - Two free outlets for the windshield (main and side).
  - One free outlet for the lower window.

NUMER 1 AND NUMBER 2 COCKPIT FLAPPER VALVES

The cockpit ventilation has two flapper valves. The Number 1 and Number 2 flapper valves are installed at the bottom of the instrument panel, forward of the control panel. Each flapper valve includes a butterfly valve contained in a cylindrical body made of glass-fiber phenolic-resin.
An electrical actuator installed on a bracket attached to each body operates the butterfly valve.
The cockpit flapper valve is of the interlocked type. The CREW selector installed on the ventilation control panel operates them.
The actuator that moves the flapper valves includes an electronic motor and a worm screw. It also has microswitches that give an indication of the fully open or fully closed position.



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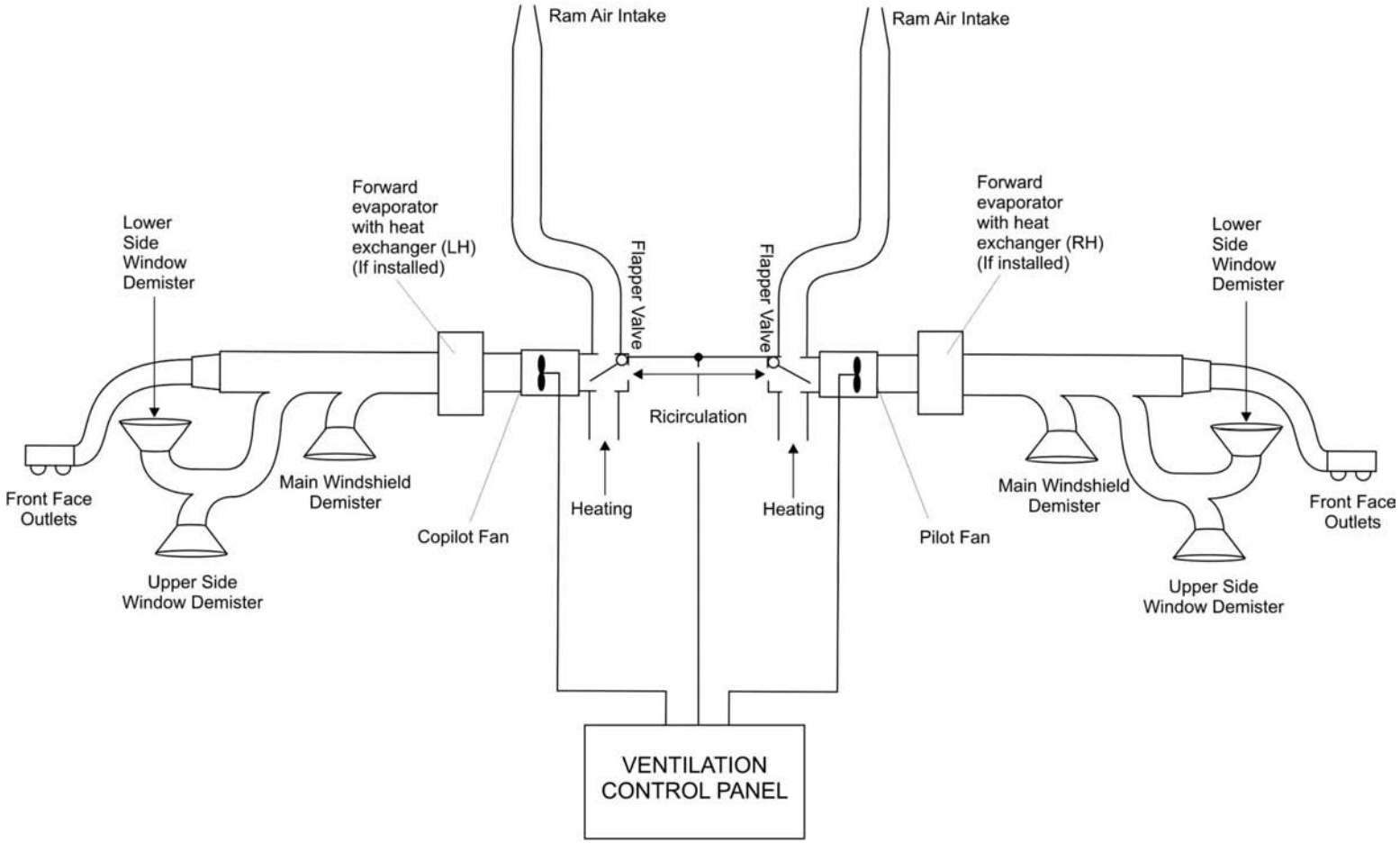
FLAPPER VALVE

NUMBER 1 AND NUMBER 2 COCKPIT FANS

The Number 1 and Number 2 cockpit fans and their motors are installed in an aluminium casing.
They include an impeller and a shaft. The impeller is made of ULTM 2300/2310 and the shaft is made from stainless steel.
A printed circuit board is installed in each assembly. And an integrated electronic box with an EMI filter is attached to the body of the fan.
The Number 1 and Number 2 cockpit fans send the air to their related ducts.
A printed board is installed in each assembly. An an integrated electronic box with an EMI filter is attached to the body of the fan. Each cockpit fans send the air to its related ducts. The 28 VDC Main Bus Bar No.2 supplies the electrical power to the ventilation cockpit fans No.2 and the cockpit control panel through the VENT circuit breaker VENT CKPT FAN 2.
The 28 VDC Non Essential Bus Bar No.1 supplies the electrical power to the ventilation cockpit fan No.1 through the VENT circuit breaker VENT CKPT FAN 1. This circuit breaker is installed on the circuit breaker panel on the overhead console.

NOTE A

The axial flow fans have two-speed brushless motors equipped with under-speed detection to prevent damage in the event of a locked motor and thermal protection to prevent overheat of the electronics.
The under-speed detection level will be the half of the low speed for both high speed and low speed selection.
The under-speed failure is subject to a time delay of 6 to 10 sec to prevent spurious fault indication, after which period, power supply to the fan will be interrupted and OFF/ON selection will have to be made.
The re-settable thermal protection device is set to trip at 120°C.



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COCKPIT VENTILATION GENERAL SCHEMATIC

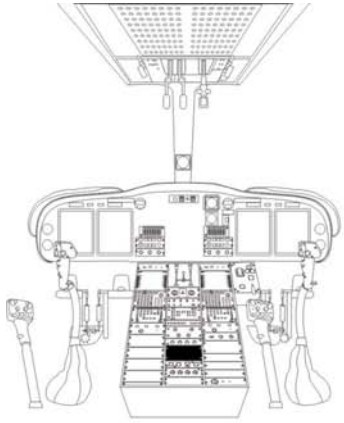
COCKPIT VENTILATION – OPERATION

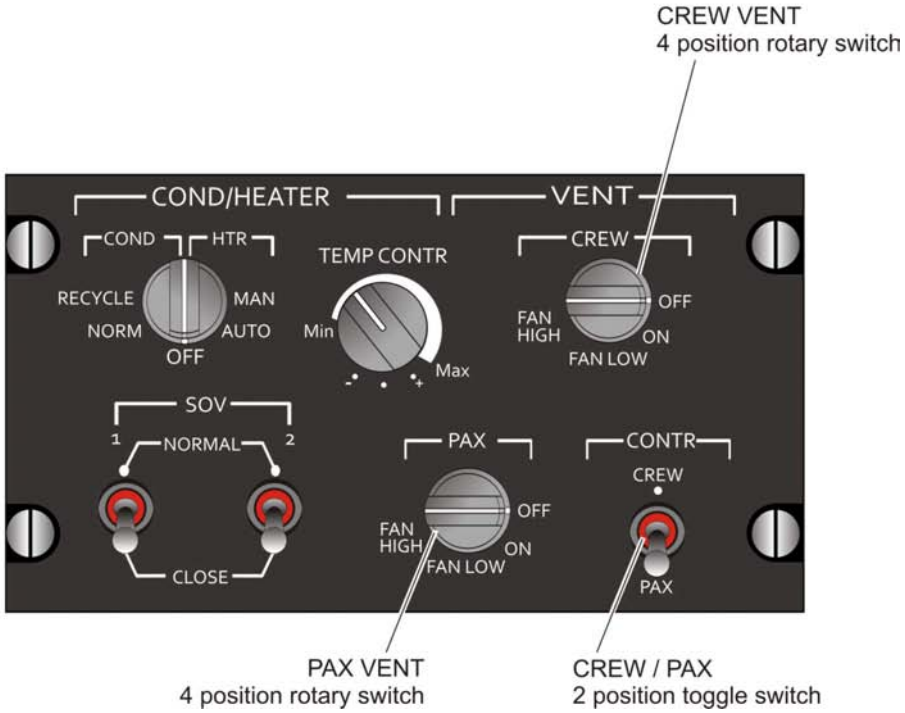
Two isolated circuits supply the airflow to the pilot and copilot stations. The circuits are the left circuit for the copilot, and the right circuit for the pilot. The components included in each of the two circuits are the same.
The external ram air enters the two circuits through the air intakes installed under the nose section of the helicopter. The position of the air intakes is the farthest possible from the engine exhausts. Thus the exhaust gases cannot go into the ventilation system.
The airflow in each circuit is controlled through the flapper valves that are operated electrically. The crew sets the valves to the open/close position with the CREW selector installed on the ventilation control panel in the cockpit.
Two axial fans supply the airflow for ventilation and the removal of mist.
The crew can set the fans to OFF, FAN LOW (speed) or FAN HIGH (speed) with the CREW selector installed on the ventilation control panel.
The ventilation control panel is installed into the interseats console in the cockpit. It gives the flight crew the control of the cockpit and cabin ventilation.
The panel has these controls and indications:

- CREW selector:..... controls the operation of both the Number 1 and Number 2 cockpit flapper valves and both the Number 1 and Number 2 cockpit fans.
- The PAX selector:..... controls the operation of the cabin flapper valve and if installed (optional) the cabin fan. In this condition the ventilation control panel becomes an optional item for the cabin fan installation.
- The CONTROL switch:..... let the functions of the CREW or PAX selector.

The CREW selector has four positions that show the OFF, ON, FAN LOW and FAN HIGH marks. To move the Number 1 and Number 2 cockpit flapper valves from the fully closed to fully open position, the selector must be set to the ON position. Then the selector can be set to FAN LOW (speed) or FAN HIGH (speed) to operate the Number 1 and Number 2 cockpit fans.
The PAX selector has two positions that show the OFF and ON marks. To move the cabin flapper valve to the fully closed to fully open position, the selector must be set to the ON position. When is installed the cabin fan (optional), the PAX selector has four positions that show also the FAN LOW and FAN HIGH marks.
The CONTROL switch has two positions: CREW and PAX.
The fans do not operate when the flapper valves are closed.
The duct of each circuit that supplies the air has a hose that drains and releases overboard the condensation.
The air goes into the cockpit through:

- The left and right front face outlets
- The left and right windshield demisters
- The left and right lateral windshield-demisters and
- The left and right window demisters.

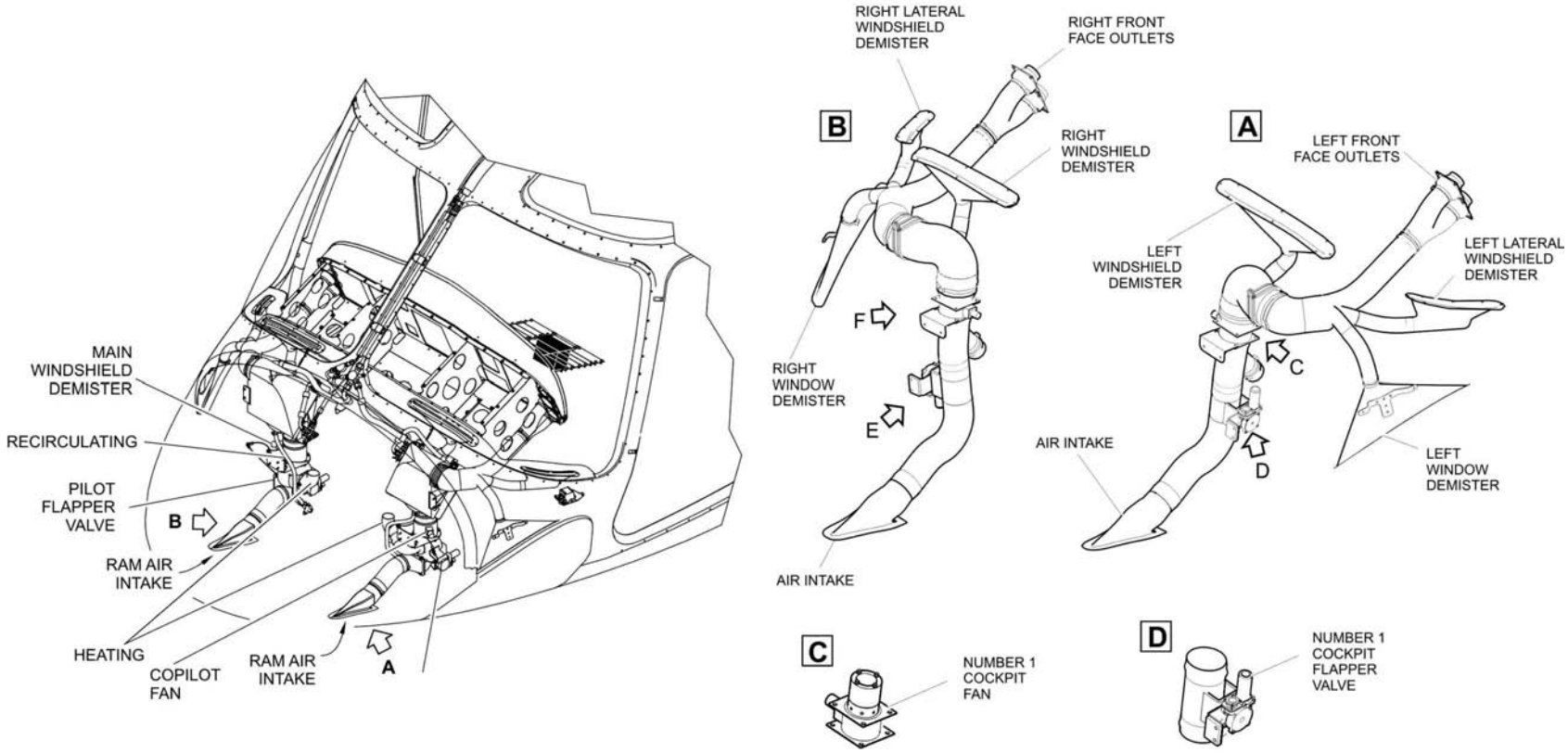




ICN-39-A-212000-G-A0126-00265-A-01-1

COCKPIT VENTILATION CONTROL PANEL

The normal operation and the malfunctions of the cockpit ventilation are monitored through the Primary Flight Display (PFD's) and the Multifunction Displays (MFD's).
During normal operation, the FWD VENT advisory message is shown on the CAS window of the MFDs and the PFD. This advisory message indicates that the cockpit ventilation is on.
In case of a malfunction, a VENT FAIL caution message is shown on the CAS window of MFD and the PFD.
The advisory and caution signals are sent to the PFD and the MFDs from the Number 1 and Number 2 Modular Avionics Units. This because they have an interface with the cockpit ventilation.



ICN-39-A-212000-G-A0126-00267-A-01-1

COCKPIT VENTILATION GENERAL LAYOUT

CABIN VENTILATION SYSTEM – DESCRIPTION

The Cabin ventilation system is a single independent system which provides fresh air to passengers and is composed of:

- a Ram Air Intake located on the upper deck fairing.
- two flapper valves (LH and RH), electrically controlled to be either fully open (ventilation on) or fully closed (ventilation off).
- two electrical fans for forced air operation.
- twelve adjustable outlets located in the PSUs (Passenger Service Units).

For the cabin, the outside ram air enters the system through one air intake in the forward pylon fairing of the cabin roof.
The air is then routed by ducts on the high sides of the cabin.
The outside ram air flow is controlled by a flapper valve electrically actuated, set by the crew with a 4 position rotary switch on the control panel.
Forced air for ventilation is provided by 2 speed axial fans.
The blower is manually controlled by the crew.
Air is then distributed via the plenum to both lateral manifolds, each one being separated.

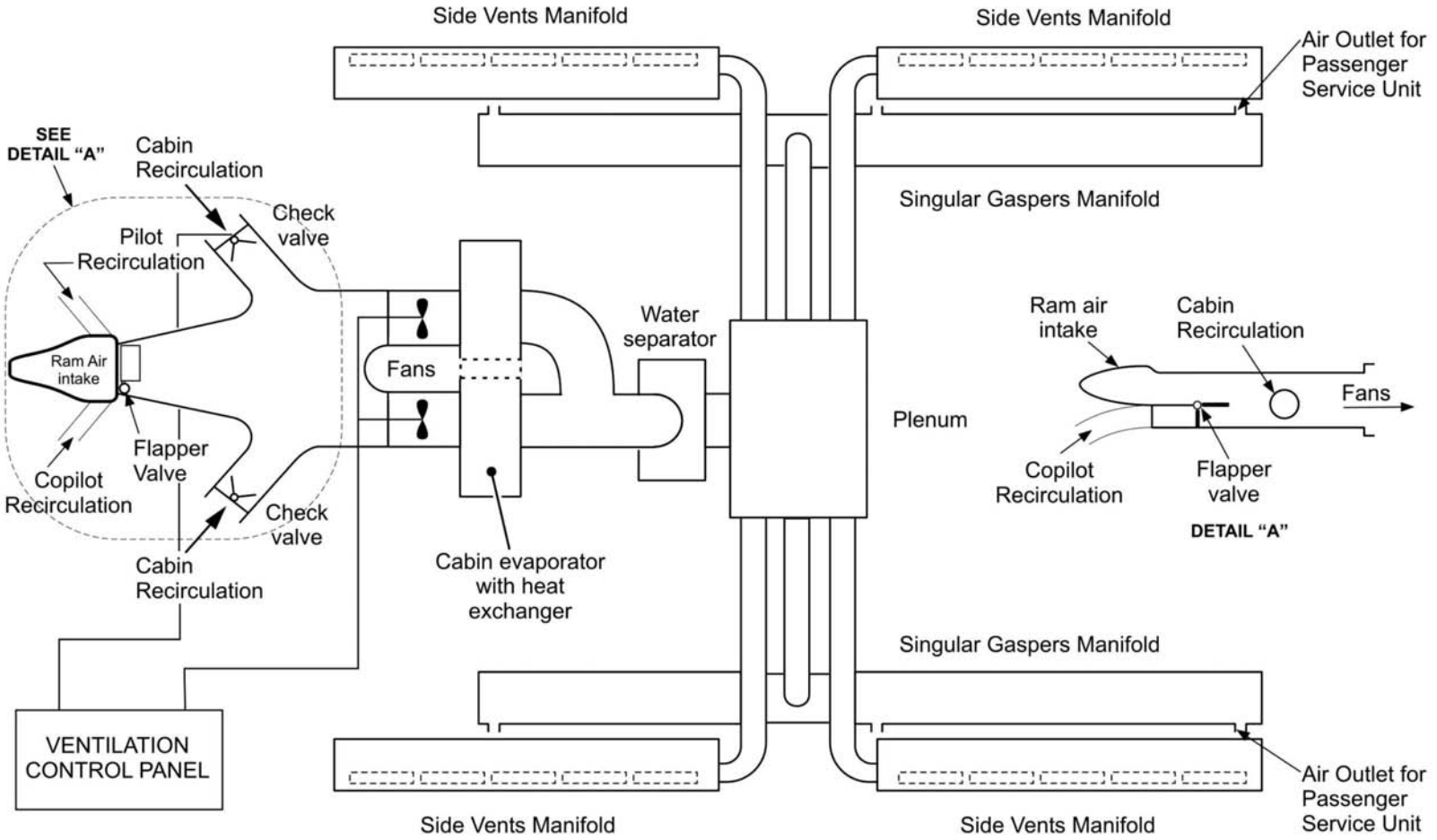
CABIN FLAPPER VALVE

The cabin flapper valve is installed in the cabin ventilation circuit in the forward sliding cowling. It lets the air from the air intake go into the system.
The flapper valve includes a butterfly valve contained in a cylindrical body made of glass-fiber phenolic-resin.
An electrical actuator installed on a bracket attached to the body operates the butterfly valve.
The actuator that moves the flapper valve includes an electronic motor and a worm screw. It also has microswitches that give an indication of the fully open or fully closed position.

The PAX selector, installed on the ventilation control panel, controls the flapper valve.

CABIN FAN (OPTIONAL)

When installed, the cabin fan is installed downstream the cabin flapper valve in the cabin ventilation circuit. The fan is of the axial type.
The fan and its motor are installed in an aluminium casing. They include an impeller and a shaft. The impeller is made of ULTM 2300/2310 and the shaft is made from stainless steel.
The PAX selector, installed on the optional ventilation control panel, controls both the cabin flapper valve and the cabin fan.
The 28 V DC Non Essential Bus Bar No. 1 supplies the electrical power to the ventilation fan through the CABIN FAN circuit breaker. This circuit breaker is installed on the circuit breaker panel on the overhead console.



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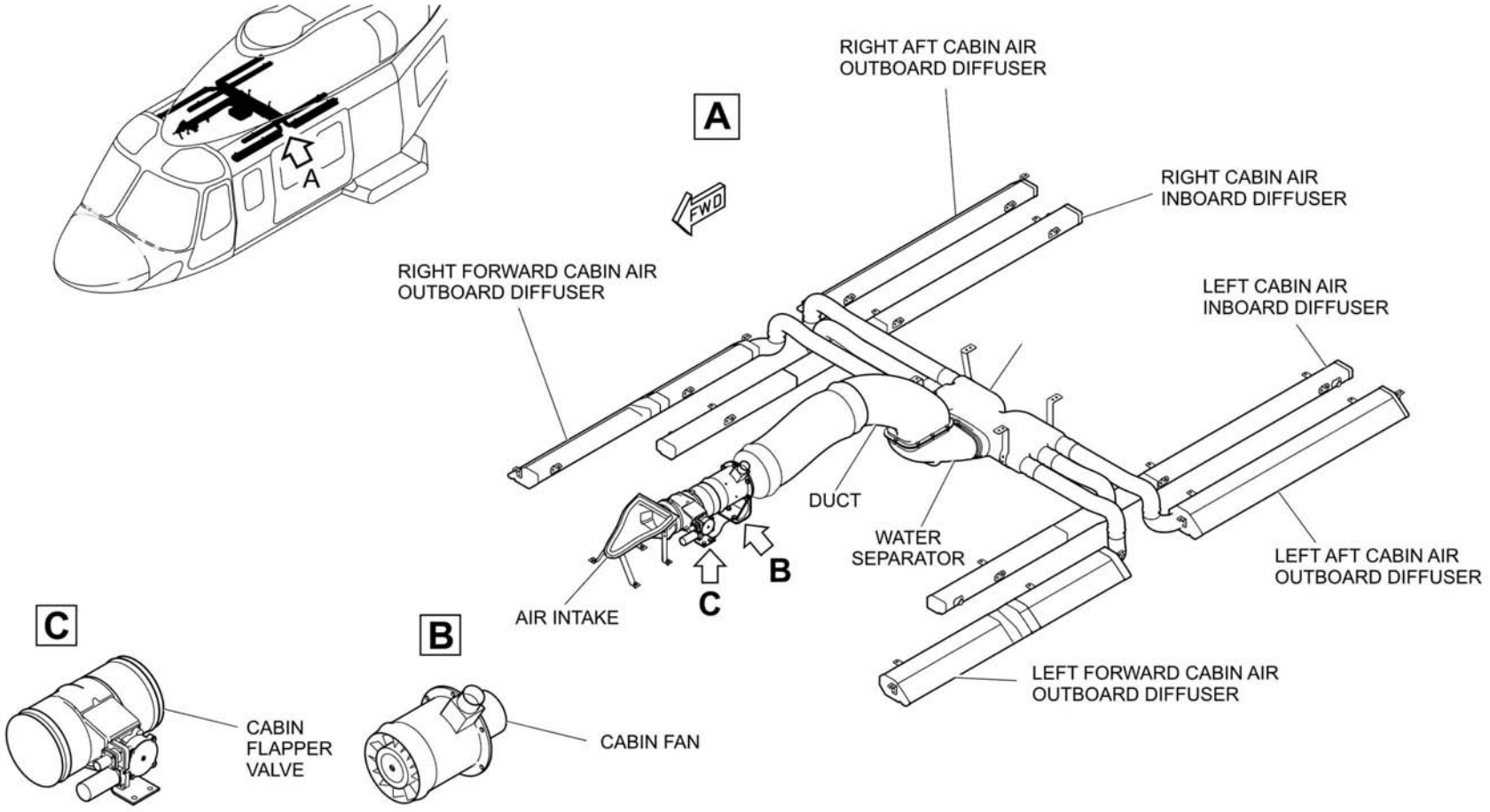
CABIN VENTILATION GENERAL SCHEMATIC

CABIN VENTILATION SYSTEM – OPERATION

The air that flows to the cabin goes into the system through an air intake installed in the forward sliding cowling. The position of the air intake is the farthest possible from the engine exhaust, thus the exhaust gases cannot go into the ventilation system.
The cabin flapper valve controls the airflow. The valve is operated electrically and the crew sets it to open or close with the PAX selector. The selector is installed on the ventilation control panel in the cockpit.
Downstream of the flapper valve, the air flows through a hose and a duct to a water separator, and then into a manifold. From the manifold, the air is sent to the cabin air inboard and outboard diffusers.
A fan is installed downstream of the cabin flapper valve to increase the airflow to the cabin when:

- The forced ventilation kit, or
- The heating and forced ventilation kit are installed in the helicopter.

The fan is of the axial type and turns at two speeds.
The fan does not operate when the flapper valve is closed.
The suction of the ventilation fan lets the external air flow into the helicopter through the air intake installed in the forward sliding cowling. Then the air flows through the ducts of the cabin ventilation and into the cabin.
The fan has an interface with the flapper valve in the duct in the upper transmission deck. The logic of the control prevents the operation of the fan when the flapper valves are fully closed.
The normal operation and the malfunctions of the cabin ventilation are monitored through the Primary Flight Display (PFD) and the Multifunction Displays (MFDs).
During normal operation, the AFT VENT advisory message is shown on the CAS window of the MFDs and the PFD. This advisory message indicates that the cabin ventilation is on.In case of a malfunction, the VENT FAIL caution message is shown on the CAS window of MFD and the PFD.
The advisory and caution signals are sent to the PFD and the MFDs from the Number 1 and Number 2 Modular Avionics Units. This because they have an interface with the cabin ventilation.



ICN-39-A-212000-G-A0126-00269-A-01-1

CABIN VENTILATION GENRAL LAYOUT

VENTILATION SYSTEM CONTROLS

The cockpit and cabin ventilation controls are housed on the ECS COCKPIT CONTROL PANEL that is located on the console.
The operative functions and the malfunctions are displayed in the CAS message windows of the MFD.
The ECS Cockpit Control Panel provides the following functions:

1 VENT CREW rotary switch:

- OFF: ..... Pilot and copilot flapper valves are closed (no airflow).
- ON: ..... Pilot and copilot flapper valves are open (Ram airflow).
- FAN LOW: ..... Pilot and copilot flapper valves are open and electrical fan operates at low speed (forced airflow).
- FAN HIGH: ..... Pilot and copilot flapper valves are open and electrical fan operates at high speed (forced airflow).

2 PAX CREW rotary switch:

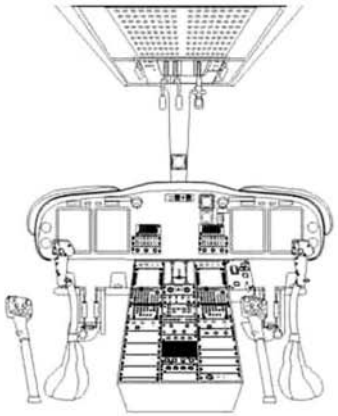
- OFF: ..... Cabin flapper valves are closed (no airflow).
- ON: ..... Cabin flapper valves are open (Ram airflow).
- FAN LOW: ..... Cabin flapper valves are open and electrical fan operates at low speed (forced airflow).
- FAN HIGH: ..... cabin flapper valves are open and electrical fan operates at high speed (forced airflow).

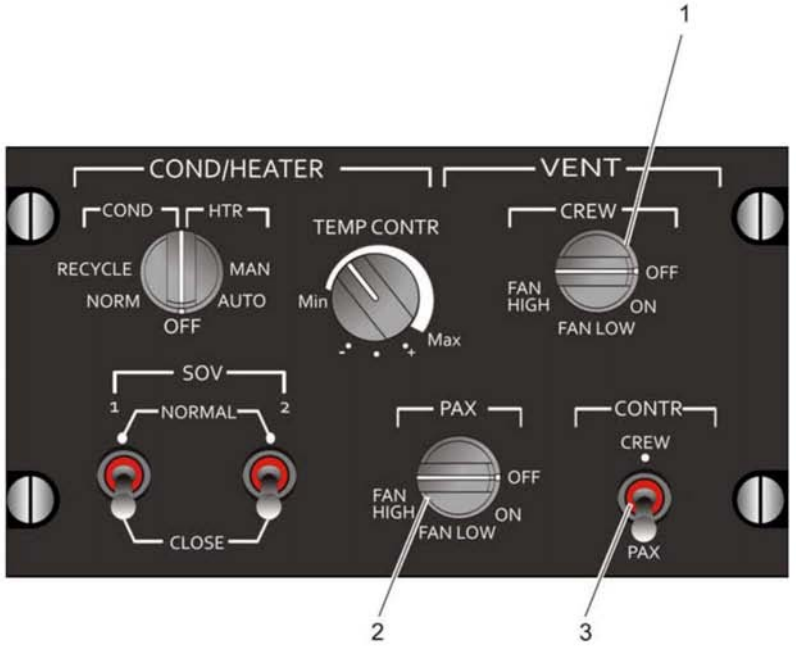
3 VENT CONTR switch:

- CREW ..... Enables the VENT PAX rotary switch (2).
- PAX ..... Enables the VENT rotary switch in the cabin (optional).

NOTE B

If the cabin controller is not installed, selecting the VENT CONTR switch to PAX causes the cabin flapper valves to open (Ram airflow) and disables the VENT PAX rotary switch.





ICN-39-A-212000-G-A0126-02885-A-01-1

VENTILATION SYSTEM CONTROLS

VENTILATION SYSTEM INDICATING

The caution provided in the CAS message window is as follows:

1 VENT FAIL:

- Failure of the crew ventilation fan.
- Failure of both nose avionic bay fans (long nose configuration only).

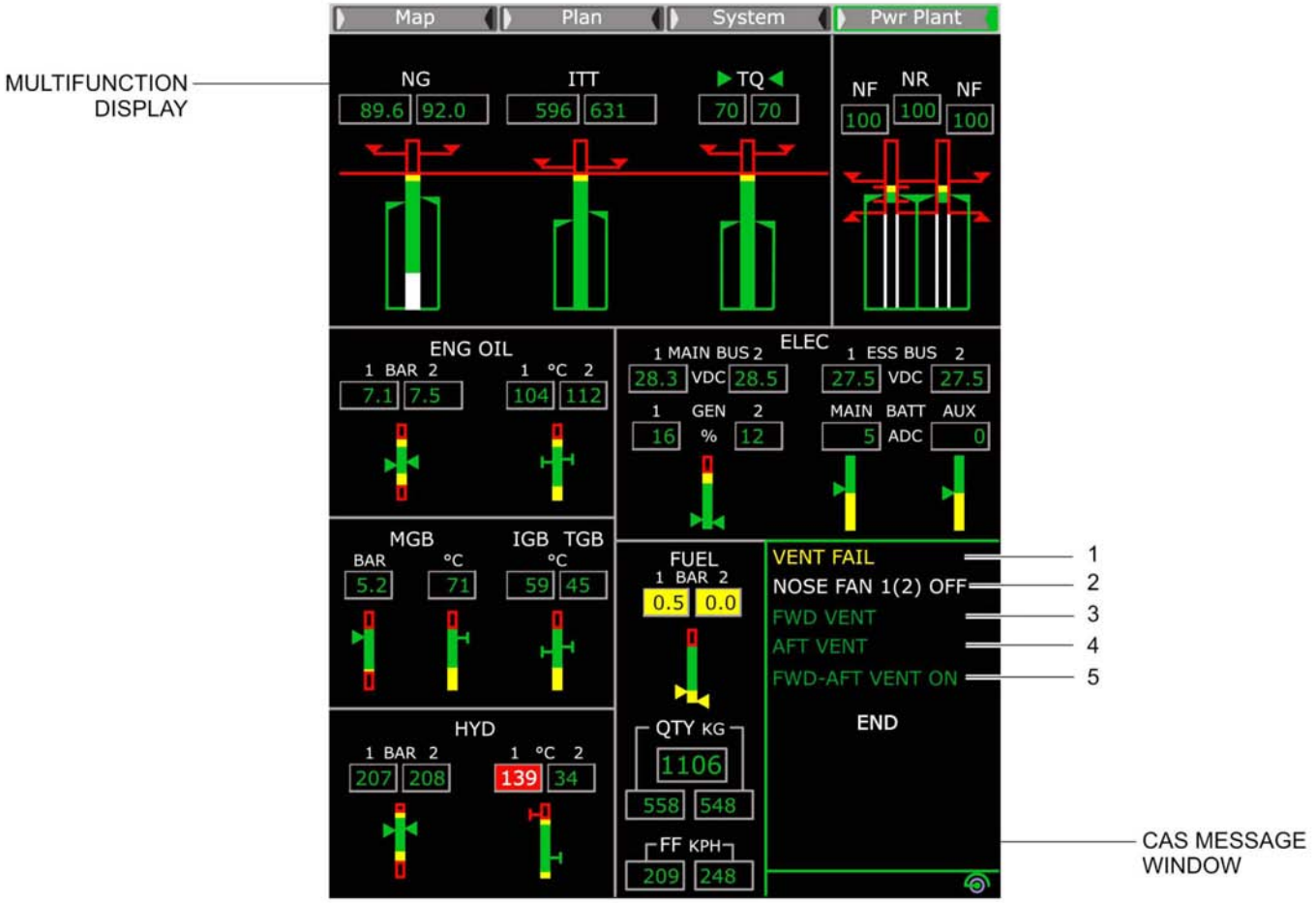
2 NOSE FAN 1(2) OFF ..... Caution triggered on ground only.

The advisories provided in the CAS message window are as follows:

3 FWD VENT ..... Forward (crew) ventilation ON.

4 AFT VENT ON ..... Cabin fan switched ON.

5 FWD-AFT ENT ON ..... Both forward (crew and cabin fans ON.

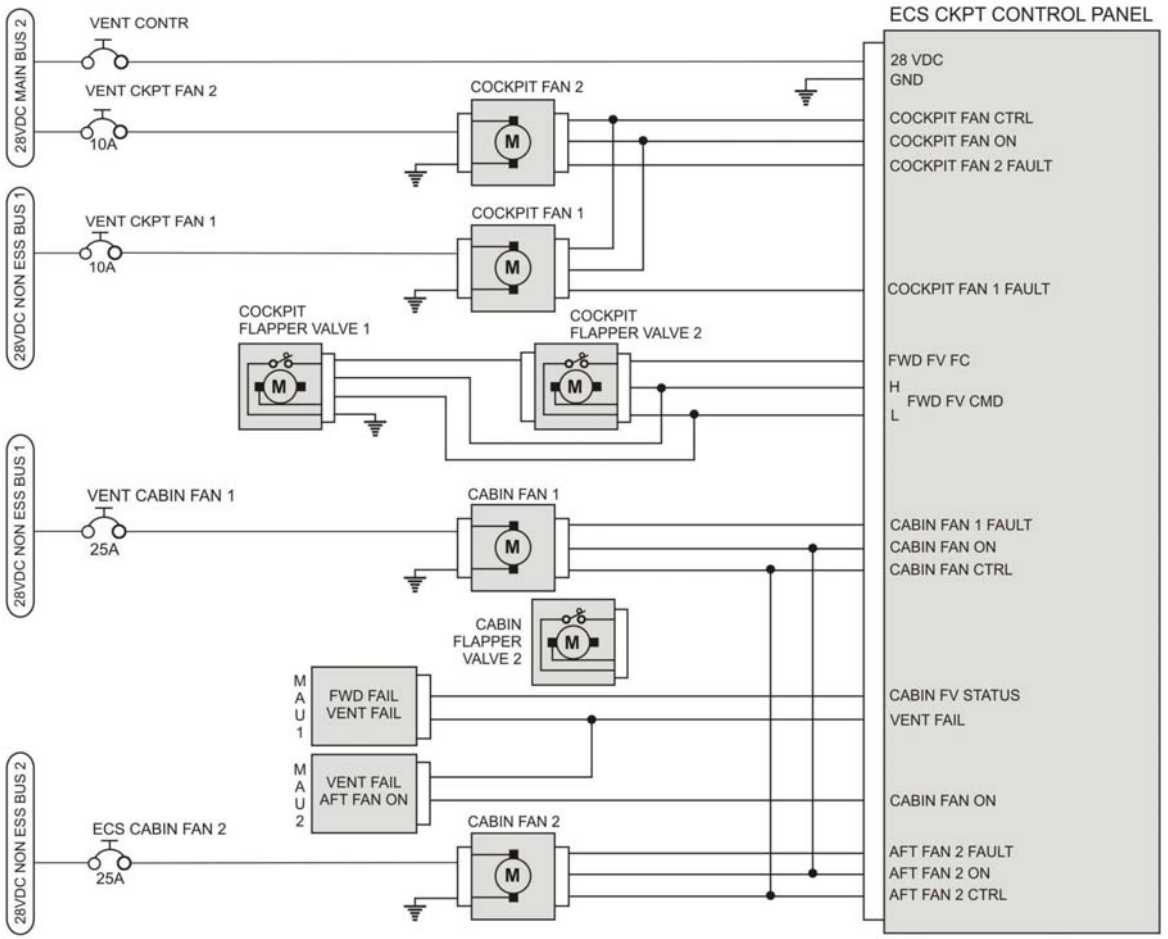


ICN-39-A-212000-G-A0126-00271-A-01-1

VENTILATION SYSTEM INDICATING

LEADING PARTICULARS

- PILOT FLAPPER VALVE:
  - Supply voltage..... 17V to 32V (28 VDC from ECS COCKPIT CONTROL PANEL)
  - Nominal current..... < 1A
  - Operating pressure..... 35 mbar g
  - Maximum opening / Closing time..... 5s
- COPILOT FLAPPER VALVE:
  - Supply voltage..... 17V to 32V (28 VDC from ECS COCKPIT CONTROL PANEL)
  - Nominal current..... < 1A
  - Operating pressure..... 35 mbar g
  - Maximum opening / Closing time..... 5s
- PILOT FAN:
  - Supply voltage..... 18V to 32V (28VDC MAIN BUS2)
  - Max current consumption..... 7A
  - Low speed..... 7350 rpm
  - High speed..... 14200 rpm
- COPILOT FAN:
  - Supply voltage..... 18V to 32V (28VDC MAIN BUS1)
  - Max current consumption..... 7A
  - Low speed..... 7350 rpm
  - High speed..... 14200 rpm
- CABIN FAN 1:
  - Supply voltage..... 18V to 32V (28VDC NON ESS BUS1)
  - Max current consumption..... 7A
  - Low speed..... 7500 rpm
  - High speed..... 13200 rpm
- CABIN FAN 2:
  - Supply voltage..... 18V to 32V (28VDC NON ESS BUS2)
  - Max current consumption..... 7A
  - Low speed..... 4900 rpm
  - High speed..... 9300 rpm
- ECS COCKPIT CONTROL PANEL:
  - Supply voltage..... 18V to 32V (28 VDC MAIN BUS2)



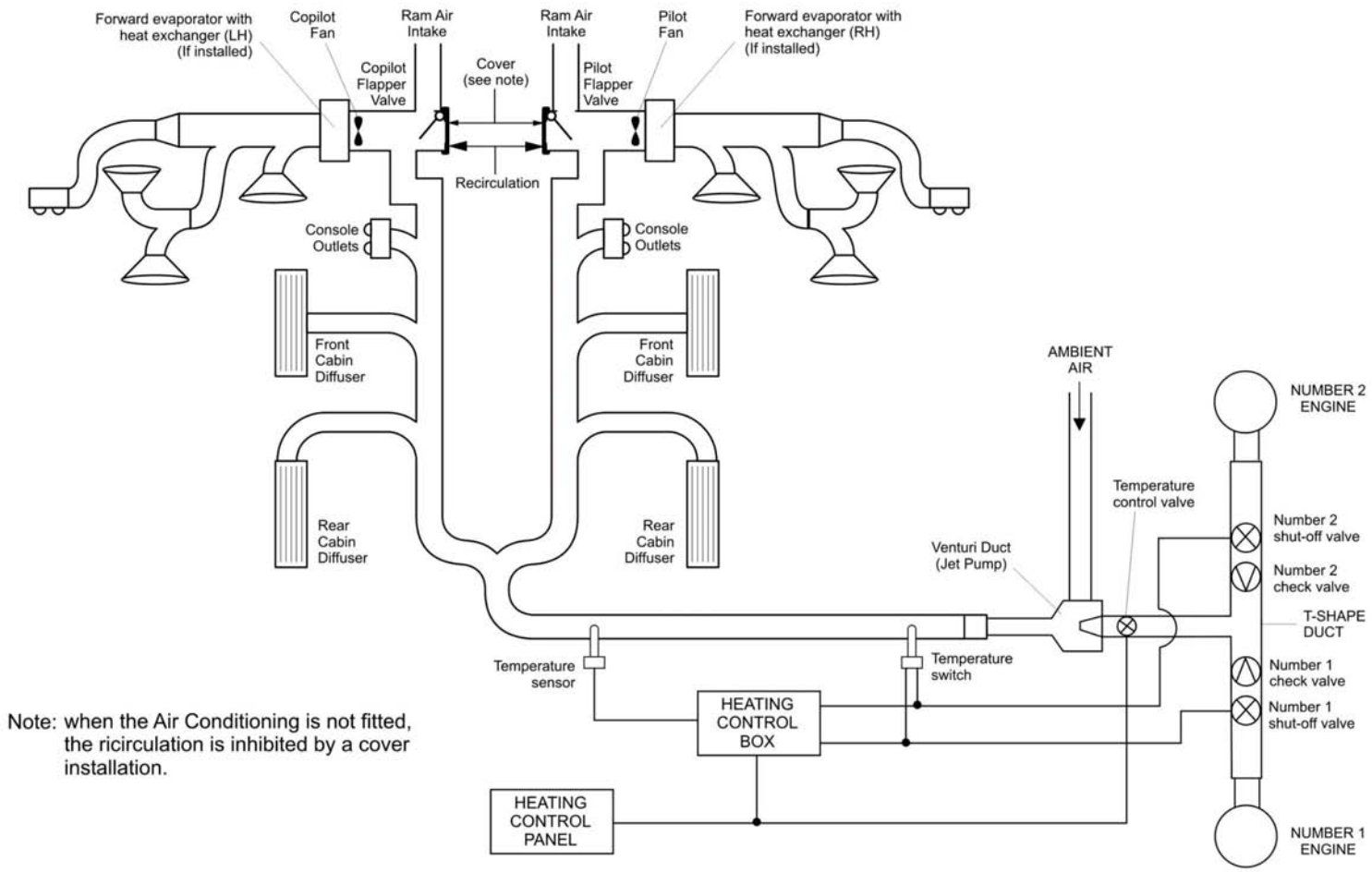
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VENTILATION SYSTEM ELECTRICAL SCHEMATIC

SECTION 40 - HEATING

DESCRIPTION

The role of heating system is to maintain a comfortable environmental in the cabin for the crew and passengers.
This system also provides defogging of the main windshields and the lower windows of the cockpit.
The basic heating system uses bleed air from the compressor discharge port of each engine.
The heating system supplies cockpit and cabin with a mix of hot pressurized air bled from the compressor discharge port (P3) of both engines and external air sucked in through an air inlet on the LH aft fuselage.
The engine hot pressurized air is routed to the Temperature Control Valve (TCV) which controls the quantity of hot air to be mixed with outside fresh air sucked in by a jet pump. The mixing occurs in the jet pump.
The mixed air enters the cabin and the cockpit via the air distribution ducts. Diffusers in the floor area outlet the heating air into the cabin (passenger area).
The distribution ducts connect the cockpit ventilation system for the pilots.
The airflow temperature is automatically controlled by the Heating Control Box (HCB) through the TEMP CONTR knob on the COND/HEATER control panel when the COND/HTR selector is set at AUTO.
In case of failure of the automatic temperature control, the pilot can manually control the position of the TCV by setting the COND/HTR selector to MAN and using the TEMP CONTR knob as a trim switch.



Note: when the Air Conditioning is not fitted, the ricirculation is inhibited by a cover installation.

ICN-39-A-214000-G-A0126-00252-A-01-1

HEATING GENERAL SCHEMATIC

AIR DISTRIBUTION

The air distribution distributes the mixed air from the jet pump flows to the cockpit and cabin through flexible and rigid ducts.
The airflow from the jet pump goes to the two lines installed on each side of the cabin under the floor. Each line gets half of the airflow. Then the air enters the cabin through four diffusers that are installed near the floor. They are the left and right-forward cabin-air diffusers and the left and right-aft cabin-air diffusers installed respectively at STA X3560 and STA X5700.
Downstream of the cabin air diffusers, the air flows into the cockpit through two air outlets that are installed on the left and right side of the pedestal, near the floor and adjacent the pilot and copilot pedal sets.
The heating control panel in the cockpit and the heating control box let the flight crew operate the heating system as necessary.
The heating control box also monitors the system and sends warnings to the flight crew when the air temperature becomes too high.
A duct temperature sensor and an overheat temperature switch give the data necessary for these functions. The switch is installed on the mixing unit of the venturi duct (jet pump).

INSULATED HOSE

The insulated hose is installed in the right side of the rear cabin and connected to the noise attenuator at one end. The opposite end of the insulated hose is connected to the venturi duct (jet pump). Its insulation keeps the temperature of the warm air that flows through, stable.

NOISE ATTENUATOR

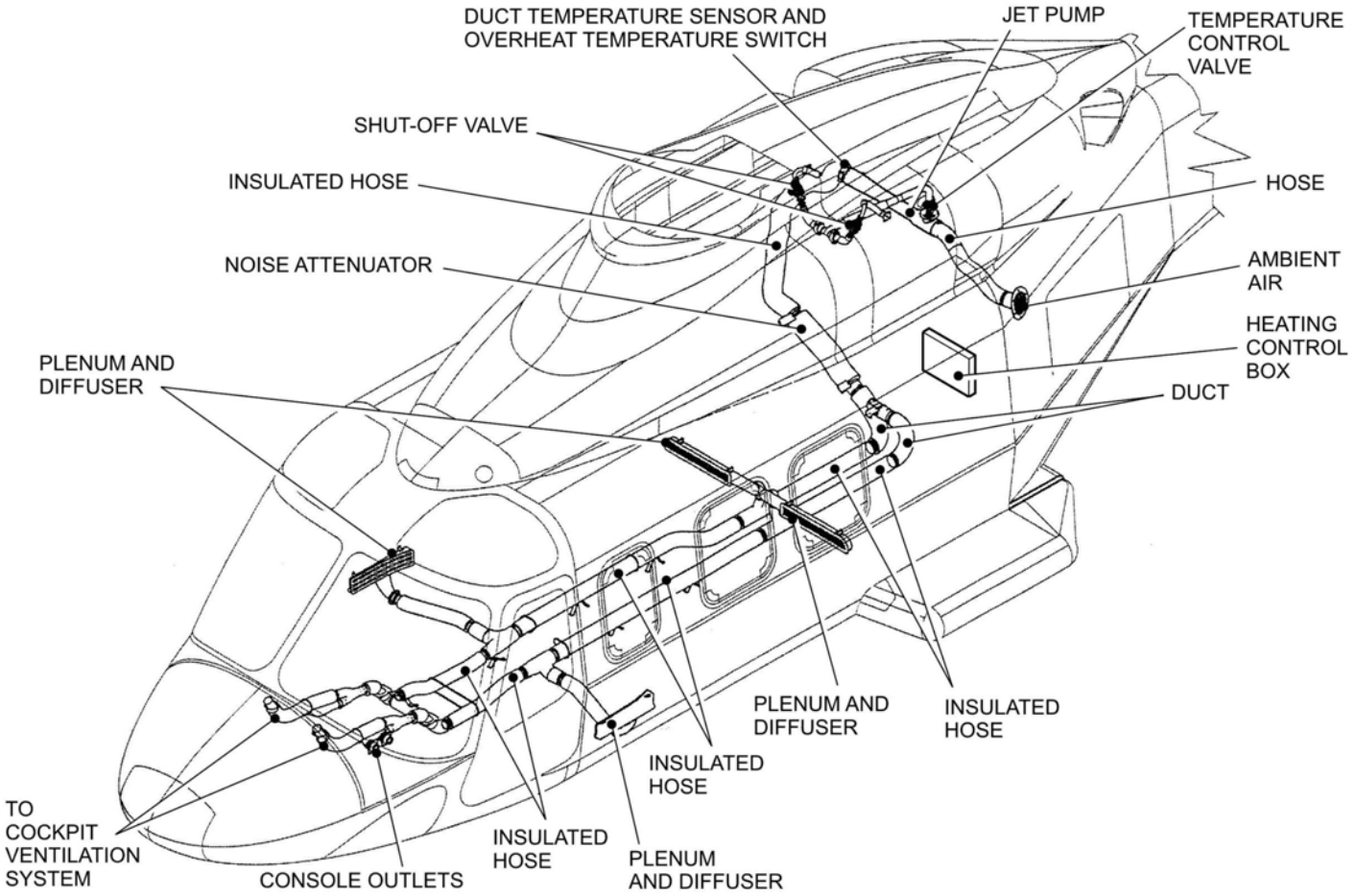
The noise attenuator is installed in the bottom right side of the aft cabin approximately at STA 5700. Two clamps attaches it to the Y-branch duct and to the insulated hose. It decrease the noise that came from the Number 1 and Number 2 engines bleed-air.

COCKPIT AND CABIN AIR OUTLETS AND DIFFUSERS

The air distribution has two air outlets at the bottom of the cockpit and four cabin air diffusers. The air outlets blow the warm air from a position near the floor and adjacent the pilot and copilot pedal sets. They are:

- The cockpit left air outlets
- The cockpit right air outlets
- The left forward cabin air diffuser
- The right forward cabin air diffuser
- The left aft cabin air diffuser
- The right aft cabin air diffuser.

The position of the two air outlets at the bottom of the cockpit is adjustable.



ICN-39-A-214000-G-A0126-00253-A-01-1

CABIN VENTILATION GENERAL SCHEMATIC

BLEED AIR

The air from the bleed port of each engine flows in the related stainless-steel pipe to one of the shut-off valves. There are two pipes in the helicopter, one for each bleed port, and their routing is through the engine compartments. Thus, these sections of the pipes go through the fire zone.
The bleed airflow is only 2 percent of the total engine airflow. The size of the opening on each of the flanges of the bleed pipes installed on the engines prevents higher bleed airflow.
Flexible sections in each pipe absorb the thermal expansion and the vibrations of the engines.
Downstream of the shut-off valves and out of the fire zone, each pipe goes to a check valve in the rear upper deck. The check valves prevent the flow of the bleed air from one engine to the other.
Downstream the check valves, the two pipes connect and become one stainless steel pipe. This pipe has a flexible section that absorbs the thermal expansion.
Through this pipe, the hot and high-pressure bleed air goes to a temperature control valve. This valve controls the quantity of bleed air that flows to the jet pump installed downstream of it. The hot bleed air and the external air from an inlet at the top of the cabin wall mix in the jet pump.
A secondary injector also supplies bleed air to the jet pump. The airflow from the secondary injector bypasses the temperature control valve when it is in failure.

HIGH PRESSURE DUCT

The two high-pressure ducts that connect the engines to the Number 1 and Number 2 shut-off valves are flexible. A section is made of a convoluted pipe from stainless steel that has a protection. This protection is a stainless steel braid.
The bleed airflow is only 2 percent of the total engine flow. This because the air goes through a hole on the engine connection flange, the dimensions of which do not let more air flow into the system. Downstream of the shut-off valves and below the engine firewall there are two rigid ducts. The ducts are made of stainless steel and have an insulation. They are connected to the check valves that are attached to a tee duct and an assembly that includes a flexible and a rigid duct. Also these ducts have an insulation. They are also connected to the temperature control valve.
The insulation of the ducts is made of fiberglass contained in a Viton casing, and attached with Velcro tape. The insulation keeps the temperature of the surface of the ducts at 200˚C, that is a temperature less than the temperature at which the fluids in the area can burn. The Viton casing does not let contamination go into the insulation.
The sections of the high pressure duct are connected through flanges with bolts, nut couplings and vee-band couplings.
Thus contamination cannot go into the heating system and the risk that there is a leakage of hot and high-pressure bleed air is minimum.

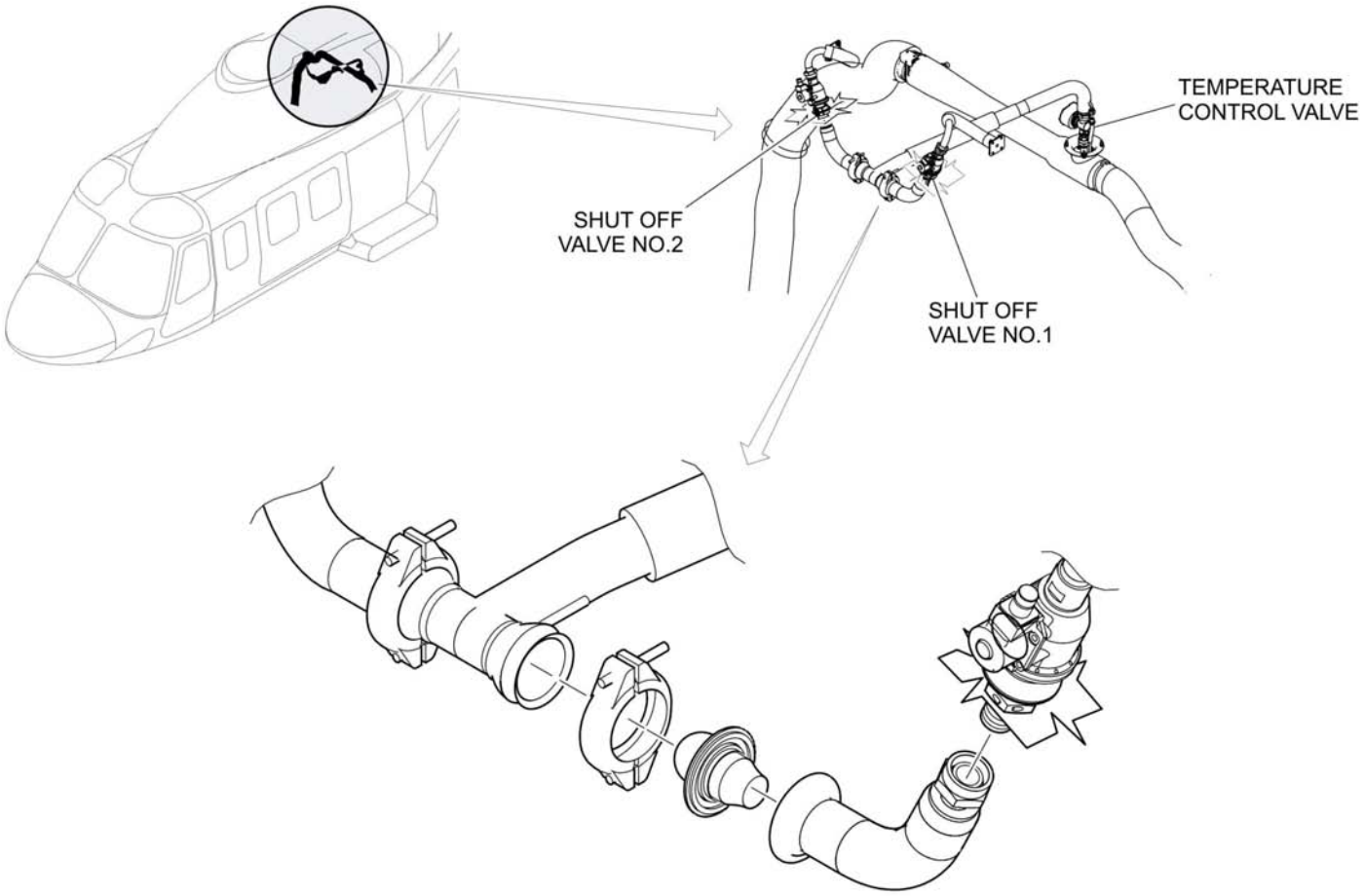
NUMBER 1 SHUT-OFF VALVE

The Number 1 shut-off valve (HR1) is attached to the firewall in the engine compartment. It stops the flow of bleed air from the Number 1 engine to the heating system when the temperature becomes too high.
The valve is usually closed when the solenoid of its actuator is not energized.
The valve body, the flanges, the piston and the internal components are made of stainless steel. The housing of the actuator is made of poly-ether ether-ketone and aluminium alloy that is anodized with sulphuric acid. The body of the electrical connector is made of stainless steel.
When the engines operate and the actuator solenoid is not energized (the SOV switches are at OFF), the valve vent port is closed.
In this condition, the bleed air pressurizes the chamber behind the piston. Thus the force on the piston inner side is higher than the force on the piston crown. The downstream pressure applies the force on the piston crown.
The difference in pressure holds the piston closed when the solenoid is not energized.
The piston rings prevent too much of a downstream leakage from the piston chamber. When the solenoid of the actuator becomes energized (SOV

switches ON), the solenoid shaft pushes the ball against the spring. The ball then moves away from the vent port, and the port stays open.
The shut-off valve is also of the fail-safe type. If power supply to the actuator stops, it closes automatically.

NUMBER 2 SHUT-OFF VALVE

The Number 2 shut-off valve (HR2) is attached to the firewall in the engine compartment. It stops the flow of bleed air from the Number 2 engine to the heating system when the temperature becomes too high.
The valve is of the same type of the Number 1 shut-off valve, and its function are also the same. The valve is in the symmetrical position related to the Number 1 shut-off valve.



ICN-39-A-214000-G-A0126-00254-A-01-1

HEATING SYSTEM SHUT-OFF VALVE 1 AND 2

VENTURI DUCT (JET PUMP)

The Venturi duct (jet pump) includes a venturi mixing unit and the two injectors of the engine bleed air. The mixing unit is made of composites, the injectors are made of stainless steel.
The diverging section of the venturi makes the mixing chamber. It has a protection from heat and is isolated from the injector assemblies that operate at high temperature.
The jet pump is installed on the ceiling of the cabin. It is connected to:

- The temperature control valve
- The intake duct of the external air
- The ducts of the low-pressure heating air.

The two functions of the jet pump are:

- To supply clean air to the cabin and cockpit
- To supply a flow of air at a controlled temperature.

The clean air is external air that flows into the helicopter through an external inlet because of the effect of suction.
The controlled temperature air is used to increase the internal temperature. Also it is used to remove the mist from the windshield and cockpit windows.
The temperature control valve controls the temperature and quantity of air that is supplied to the primary injector. The air that goes to the secondary injector, which is smaller than the primary one, bypasses the temperature control valve.
The hot and high-pressure air from the temperature control valve flows out of the primary injector at the throat of the jet-pump Venturi.
The speed of the air from the injector and the converging section of the Venturi causes a low pressure area. The suction of this area causes some air to flow to it from the higher pressure area behind the injector. This is external air because the area is connected to the external inlet at the top of the cabin wall.
The diverging section of the Venturi expands the bleed air and mixes it with the external air.
The duct temperature sensor installed at the outlet of the pump controls the temperature of the mixed air. To do so, it sends a feedback signal to the heating control box. Thus, the system supplies air at the correct pressure and speed and at the temperature that the crew sets.
A temperature switch is installed adjacent to the duct temperature sensor at the jet pump outlet.
When the temperature control valve is closed, the secondary injector supplies a small flow of bleed air to the Venturi throat. The suction causes again external air to go into the system, where it mixes with the bleed air. The flow of mixed air is sufficient to remove the mist from the helicopter windshield and cockpit windows.

AIR INTAKE DUCT

The air intake duct is installed in top left side of the cabin at STA 6700. Six screws attaches it to the fuselage structure. A mesh is the protection of the inlet and prevent ingestion of unvwanted objects.

NUMBER 1 CHECK VALVE

The Number 1 check valve is of the double-flapper type and has the following components:

- Vee-band couplings that attach the check valve to the Number 1 bleed-air ducts
- The base
- The two flappers
- The hinge pin
- The collar
- The spring
- The two spacers.

When the check valve is fully open, the airflow flows freely, with minimum resistance.

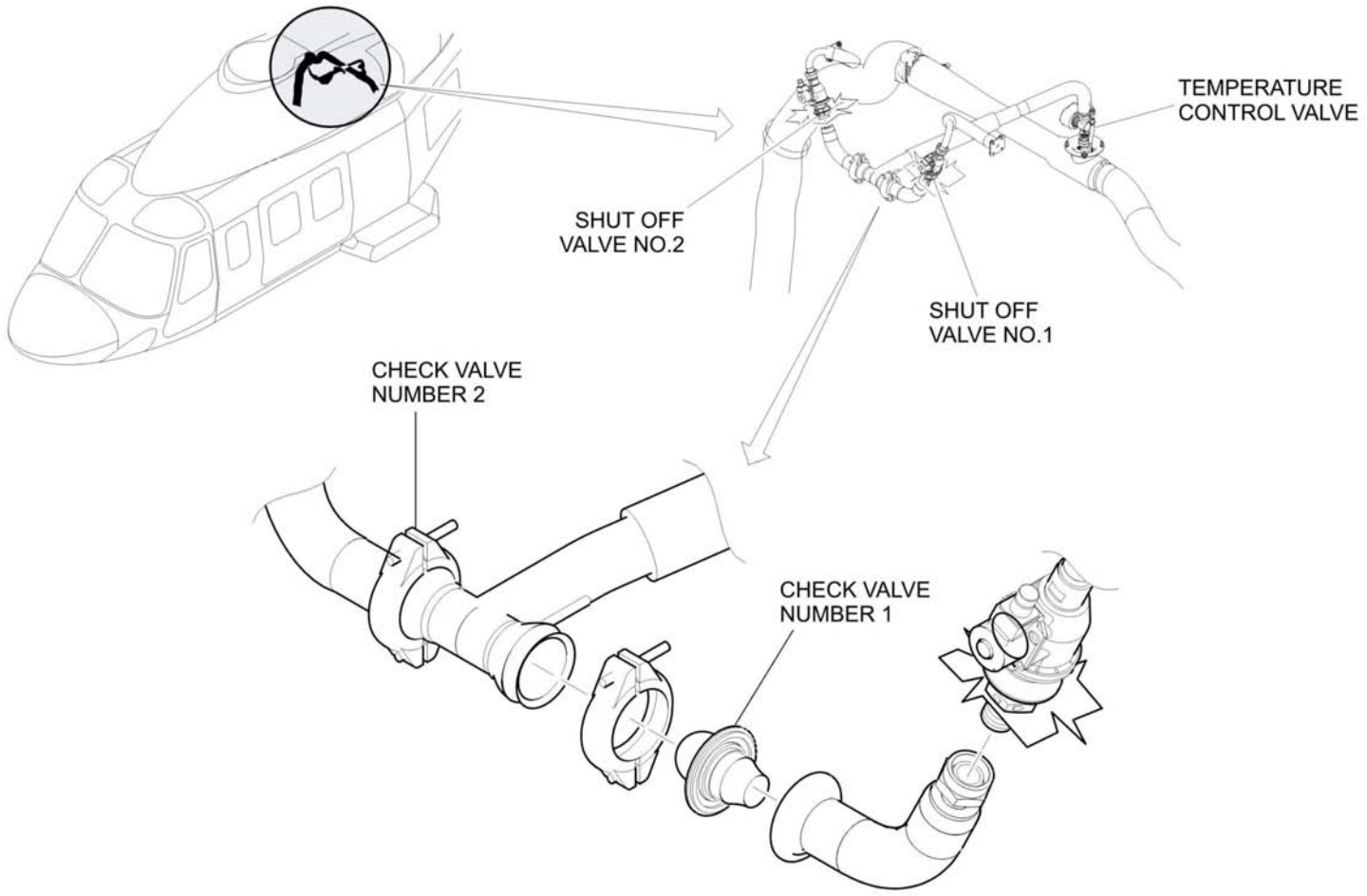
The check valve prevent an opposite flow of bleed air to the other engine. This when the pressure of the bleed air from the two engines is very different.

NUMBER 2 CHECK VALVE

The Number 2 check valve is of the double-flapper type and has the same components and function as the Number 1 check valve. Its position is at the opposite symmetrical side of the high-pressure T-shape duct.

TEMPERATURE CONTROL VALVE

The temperature control valve is installed on the top of the jet pump, and below the engine firewall.
The valve is cylindrical, and includes a carbon plug and a stainless steel body.
A DC motor with brushes is connected to the valve plug through a tangential motor and a shaft.
The valve actuator assembly is installed in a casing made of poly-ether ether-ketone and aluminum alloy. It is installed on a base that has an insulation from heat.
The plug can turn in the body to any position from fully open to fully closed.
Microswitches sense and control the fully open and fully closed positions.
A spring washer with a cap holds the valve against its seal.
The temperature control valve controls the temperature of the bleed air that flows through the high pressure duct to the jet pump injector.



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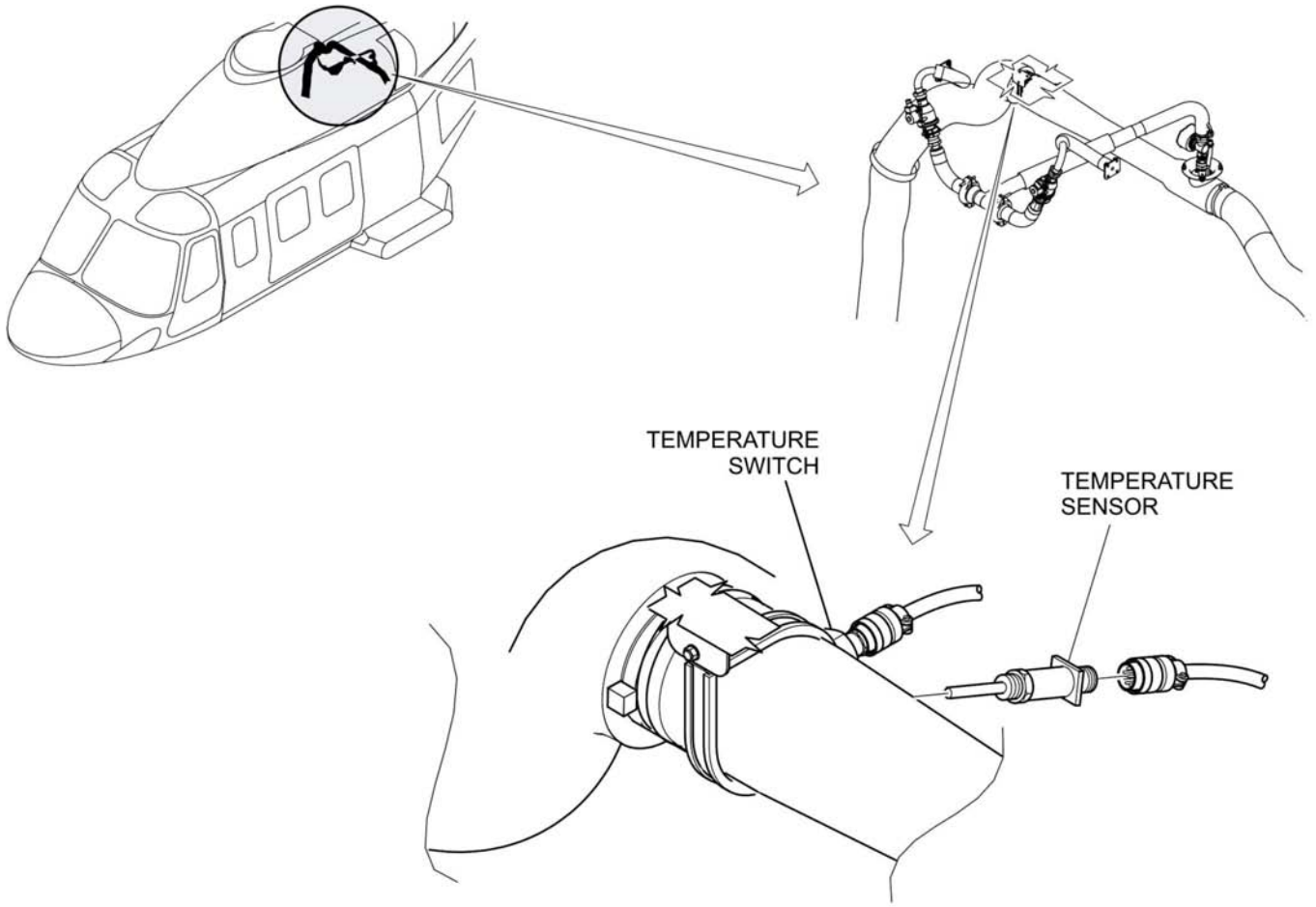
HEATING SYSTEM CHECK VALVE 1 AND 2

TEMPERATURE SWITCH

The temperature switch is attached to a boss with a thread at the outlet of the mixing unit of the jet pump. It includes a housing, a thermal switch and an electrical connector.
The housing is made of stainless steel. The thermal switch is of the two-metal type. The electrical connector is welded to the housing.
When the temperature in the duct becomes higher than the set temperature, the switch opens. This prevents damage to the ducts and too much increase in cabin and cockpit temperature. The temperature switch opens when the duct temperature is 85˚ ± 5˚C. It closes when the temperature is 70˚ ±5˚C.
A signal is sent to the heating control box and the shut-off valves to give a DOTS open or closed indication.
When the switch is at ON because the temperature in the duct is too high, an HEATER FAIL caution message is shown in the CAS window of the Multifunction Display. Also, the two shut-off valves automatically stop the heating system.

TEMPERATURE SENSOR

The temperature sensor is attached to a boss with a thread at the outlet of the mixing unit of the jet pump. It includes one element that senses the temperature, and a stainless steel body.
Its electrical connection is through a 6-pin connector contained in an aluminium body.
The air flows directly around the sensor housing, thus the sensor measures the temperature accurately and immediately.
The temperature sensor measures the temperature of the mixed air downstream of the mixing unit of the jet pump.
The result is transmitted to the heating control box. The heating control box sends an automatic feedback signal that is used to control the temperature control valve.



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TEMPERATURE SWITCH AND TEMPERATURE SENSOR

HEATING SYSTEM CONTROLS

The heating controls are housed on the ECS Cockpit Control Panel that is located on the console and operated by pilots.
The operative function and the malfunction are displayed in the CAS message window of the MFD.
The panel provides the following functions:

1 HTR:

- OFF: ..... the heating/air conditioning system is not operative.
- AUTO: ..... the heating system is in automatic mode.
- MAN: ..... the heating system is in manual mode.

2 TEMP CONTR: ..... It controls the temperature in the cockpit and cabin.

\+ ..... opens the temperature control valve when MAN is set.

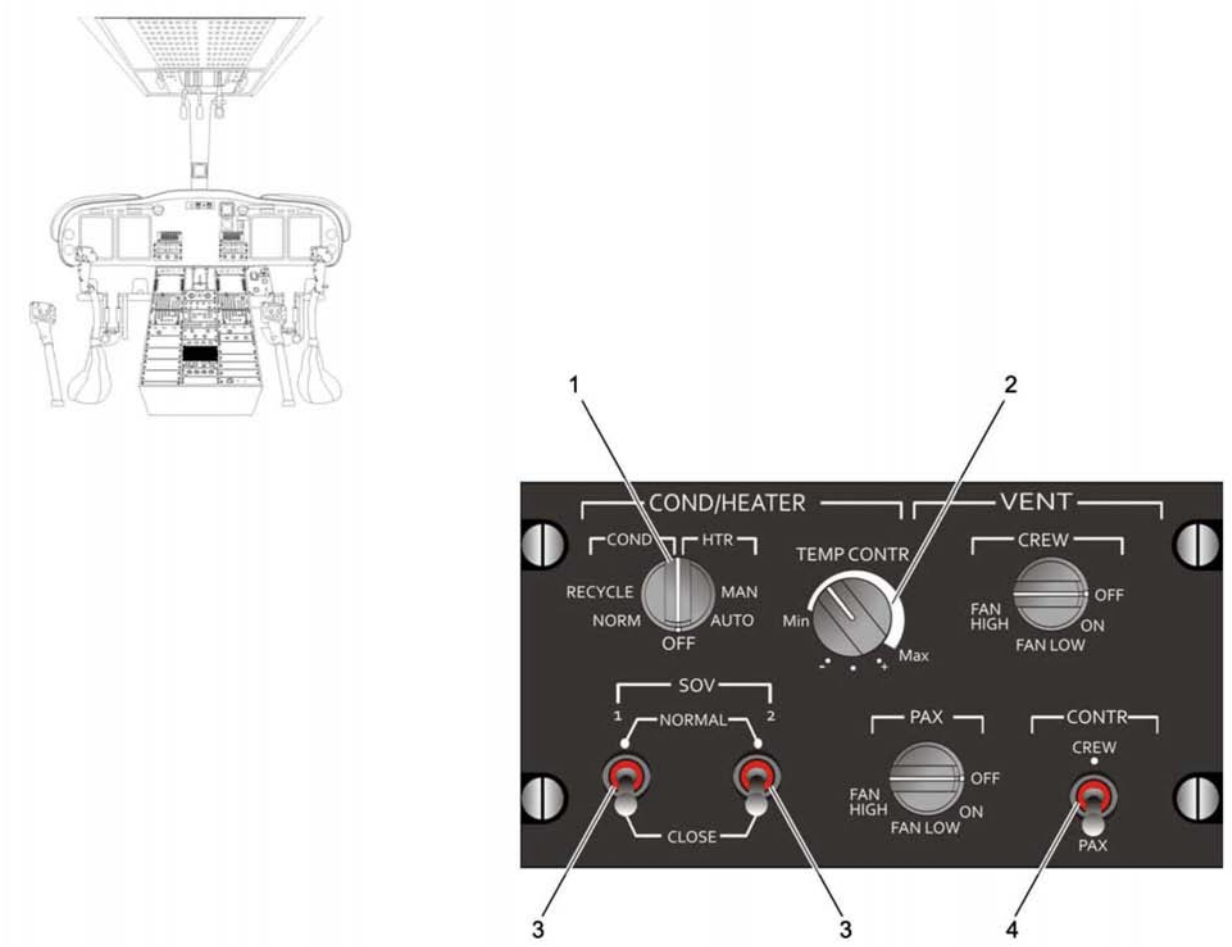
\- ..... closes the temperature control valve when MAN is set.

3 1(2) SOV:

- CLOSE: ..... the shut-off valve is forced to close.
- NORMAL: ..... the shut-off valve is open or closed.

4 CONTR:

- CREW: ..... the crew retains authority over the temperature control in the cockpit and cabin, when AUTO/ MANUAL mode is set, through the cockpit control panel.
- PAX: ..... this selection is not operative when heating system is set both AUTO and MAN mode.



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HEATING SYSTEM CONTROLS

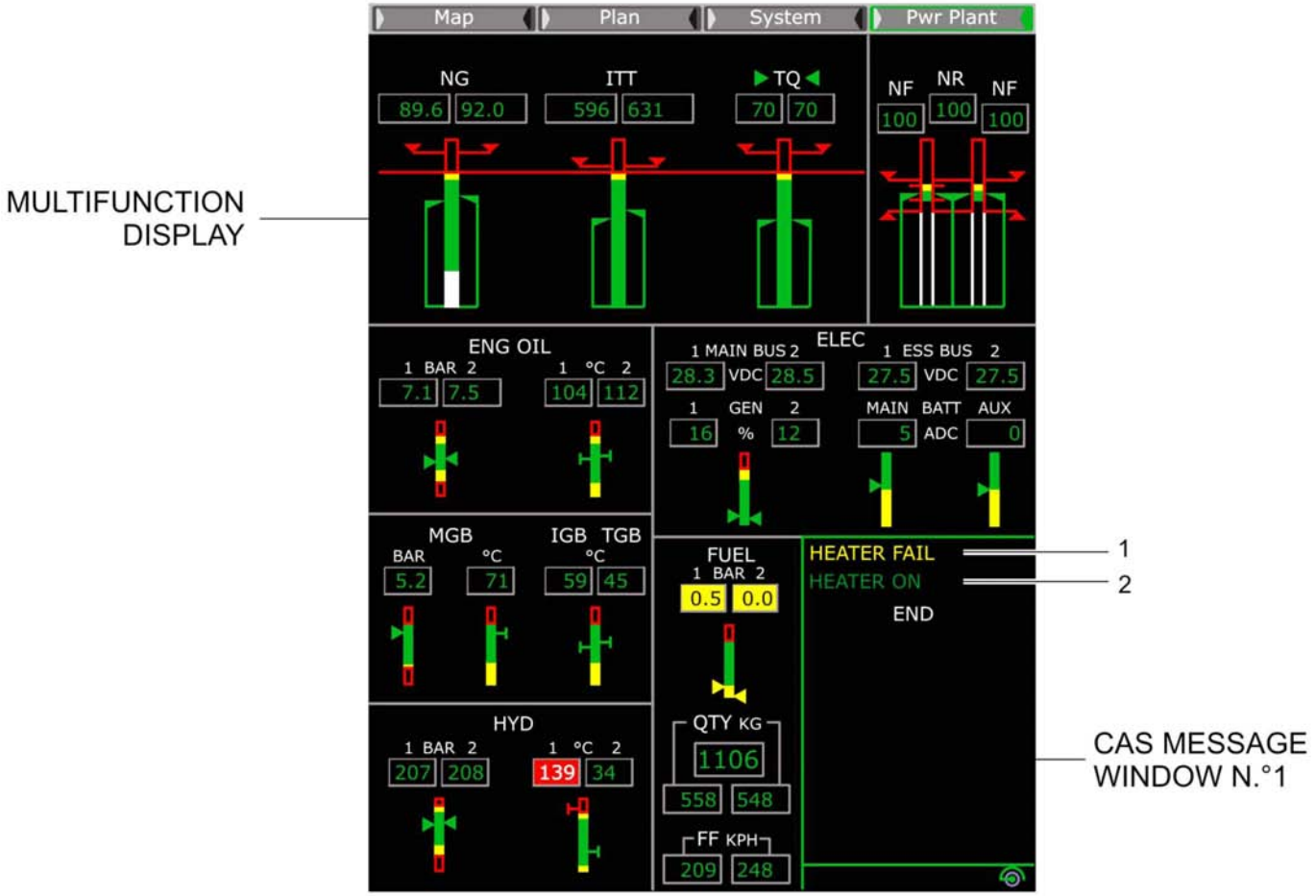
HEATING SYSTEM INDICATING

The Primary Flight Display (PFD) and the Multifunction Display (MFD) show the related indications for the flight crew. The MFD shows, on the CAS window, the caution and advisory messages that follow:

1 HEATER FAIL ..... The heating system is unserviceable.

2 HEATER VENT ..... The Heating system ON.

The indication shown on the PFD and the MFD are sent from the Number 1 and Number 2 Modular Avionics Units (MAU1 and MAU2). This because they have an interface with the heating system.



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HEATING SYSTEM INDICATING

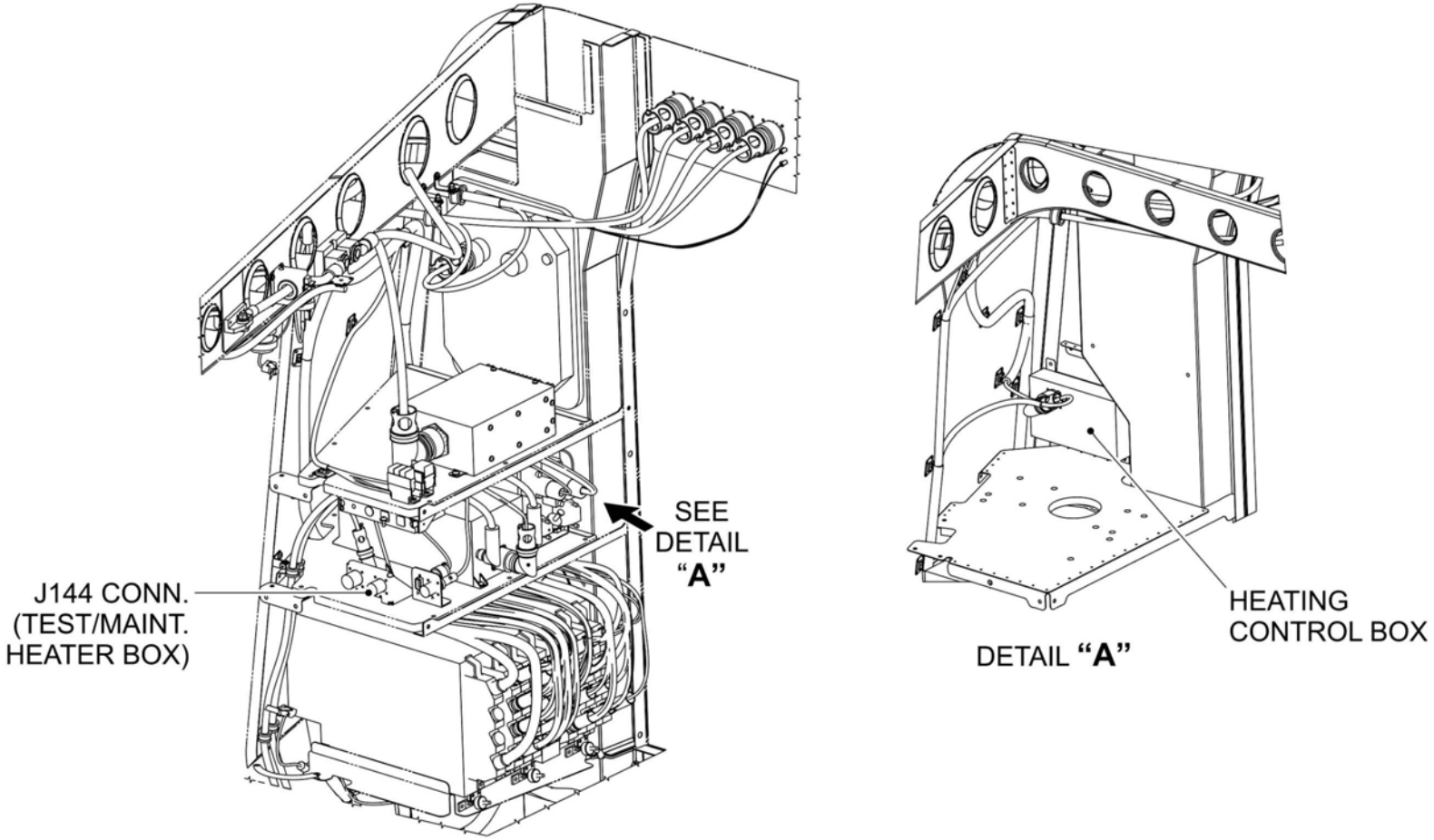
HEATING CONTROL BOX

The Heating Control Box (HCB) located on the LH avionic bay and the Control Panel provide the proper function of the heating system and monitor it to alarm the crew and to ensure the overheat protection.
The electronic card is made of epoxy resin. It is painted with a layer of acrylic varnish that gives protection from water.
It includes an analog and logic-array card that is installed in a metal housing.
The Number 2 non-essential bus supplies the 28 V DC power to the heating control box through the HTR circuit breaker. This circuit breaker is installed on the circuit breaker panel on the overhead console.
The heating control box operates in the AUTO heater mode when:

- The ENG 1 and ENG 2 switches on the engine control panel are not in the ARM position
- The HTR selector is at AUTO position
- The heating control box operates in a closed loop.

The loop controls the temperature of the air at the outlet of the jet pump. To do so, it compares the reference temperature (the crew sets it) with the temperature in the duct (the temperature sensor measures it). The difference between the two values is supplied to a corrector, and is then used to operate the control of the temperature control valve. This prevents too high temperature in the ducts. The corrector is of the proportional type, its gain does not change.
Two modes are basically proposed for the system:

- In the automatic mode the temperature of the mixed airflow is controlled by the heating control box by comparison between the mixed air temperature at the Air Duct Temperature Sensor (ADTS) and the temperature demand set by the rotary potentiometer labelled TEMP CONTR selected on the control panel, up to a temperature of 75°C.
  The correct temperature is achieved by opening or closing the Temperature Control Valve (TCV) to minimize the error between the crew selection and the temperature measured at the exit of the jet pump. In case of overheat not detected and controlled by the duct temperature sensor, the Duct Overheat Temperature Switch (DOTS) will trip, to close directly the shut-off valves.
- The Ext. Air Temp Sensor gives the information about air temperature that feeds the jet pump. With this temperature value, the HCB is able to compute the position that the TCV should have in order to obtain the right temperature (required by pilot) at the outlet of the jet pump.
- In the manual mode the heating control box is by-passed and authority over the temperature control valve is assigned to the 2 position toggle function of the rotary potentiometer labelled TEMP CONTR. Left position (-) for closing and right position (+) for opening.
  This enables the pilot to manage the position of the temperature control valve and therefore control directly the air temperature.
  The heating system also utilises the cockpit ventilation distribution system when the fans are operated, to drive the air to the transparency diffusers and upper gaspers.
  In case of fire, when the extinguishing system is armed, the heater control box closes automatically the two shut-off valves.



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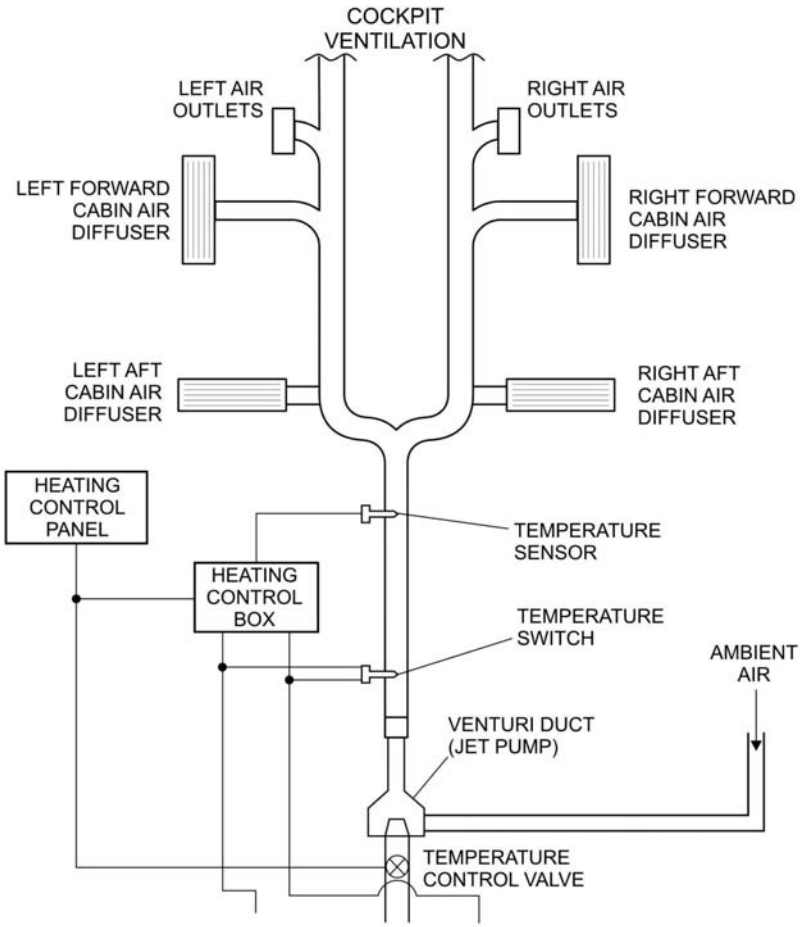
HEATING CONTROL BOX LOCATION

HEATING SYSTEM OPERATION

The Heating Control Box (HCB) located on the LH avionic bay and the Control Panel provide the proper function of the heating system and monitor it to alarm the crew and to ensure the overheat protection.
The heating system operates in those modes: the automatic mode and the manual mode.
In the automatic mode the heating control box controls the temperature of the mixed airflow. To do this, it compares the temperature of the mixed air supplied by the temperature sensor with the temperature set on the heating control panel.
The temperature control valve opens or closes to give the correct temperature. This temperature is the temperature at which the difference between:

- The temperature the flight crew has set on the cockpit control panel and
- The temperature measured at the duct temperature sensor at the exit of the venturi duct is minimum.

When the AUTO mode is set (HTR selector set on AUTO), the flapper valves of the cockpit and cabin ventilation sub-systems are closed. But the flight crew can use the related selector to manually operate them.
If the temperature sensor senses a too high temperature in the duct, the temperature switch operates and closes the Number 1 and Number 2 shut-off valves.
The heater control panel sends the HEATER AUTO MODE command to the ENG FIRE EXTING control panel before it sends it to the heater control box. This disconnects the heating system if, on the ENGINE FIRE EXTING control-panel, the ENG 1 and ENG 2 are at set at ARM.
In the manual mode (HTR selector set on MAN), the heating control box is overridden. Then the TEMP CONTR selector, on the heating control panel, controls the temperature control valve.
This lets the crew change the position of the temperature control valve as necessary to have a direct control of the temperature of the air that goes in the cockpit and in the cabin. This is to remove the mist from the windshield and cockpit windows if necessary.

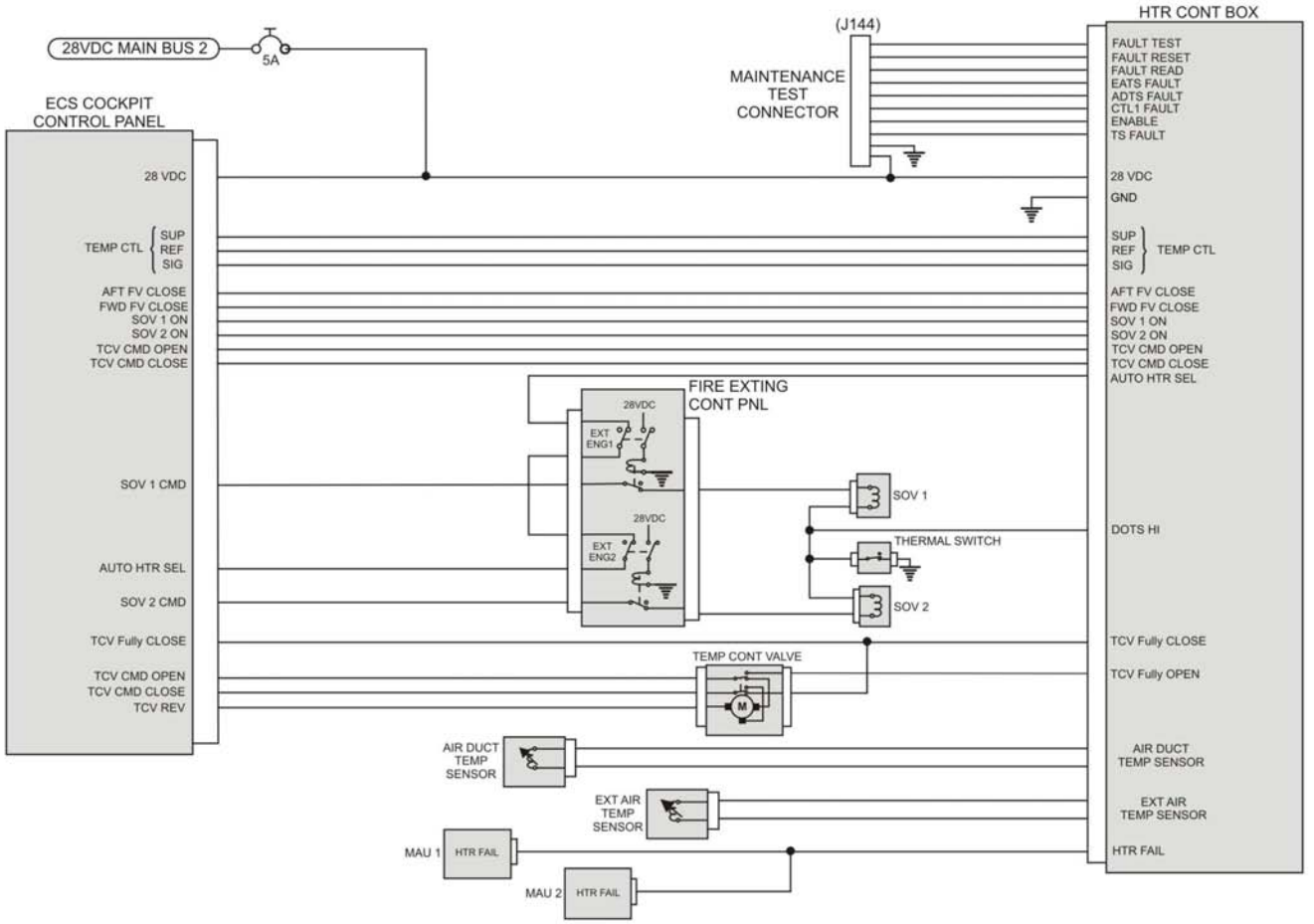


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HEATING SYSTEM CONTROLS

LEADING PARTICULARS

- SHUT-OFF VALVE NO.1 (2)
  - Nominal supply voltage..... 28VDC (from ECS COCKPIT CONTROL PANEL)
  - Coil current consumption..... ≤0.4 A
  - Maximum operating pressure..... 12.8 bar
  - Opening time..... ≤3 s
  - Closing time..... ≤4 s
- TEMPERATURE CONTROL VALVE
  - Supply voltage..... 17V to 32V (from ECS COCKPIT CONTROL PANEL)
  - Nominal current..... 0.6 A
  - Maximum servicing pressure..... 12.8 bar
  - Maximum opening and closing time..... 4 s
- HEATING CONTROL BOX:
  - Nominal supply voltage..... 28VDC (MAIN BUS 2)
- AIR DUCT TEMPERATURE SENSOR:
  - Operating ambient temperature range..... -40°C to +125°C
  - Ambient storage temperature range..... -55°C to 125°C
  - Accuracy..... ±1°C (0°C to +85°C)
  - Response time..... 5 m/s air speed < 15 sec 50 m/s air speed < 6 sec
- DUCT OVERHEAT TEMPERATURE SWITCH:
  - Single contact switch..... Normally closed
  - Switch OFF..... at 85°C ±2°C
  - Switch ON..... at 70 °C ± 5 °C
  - Operating range (Limited to +85°C on AB139)..... -40 °C to +120 °C



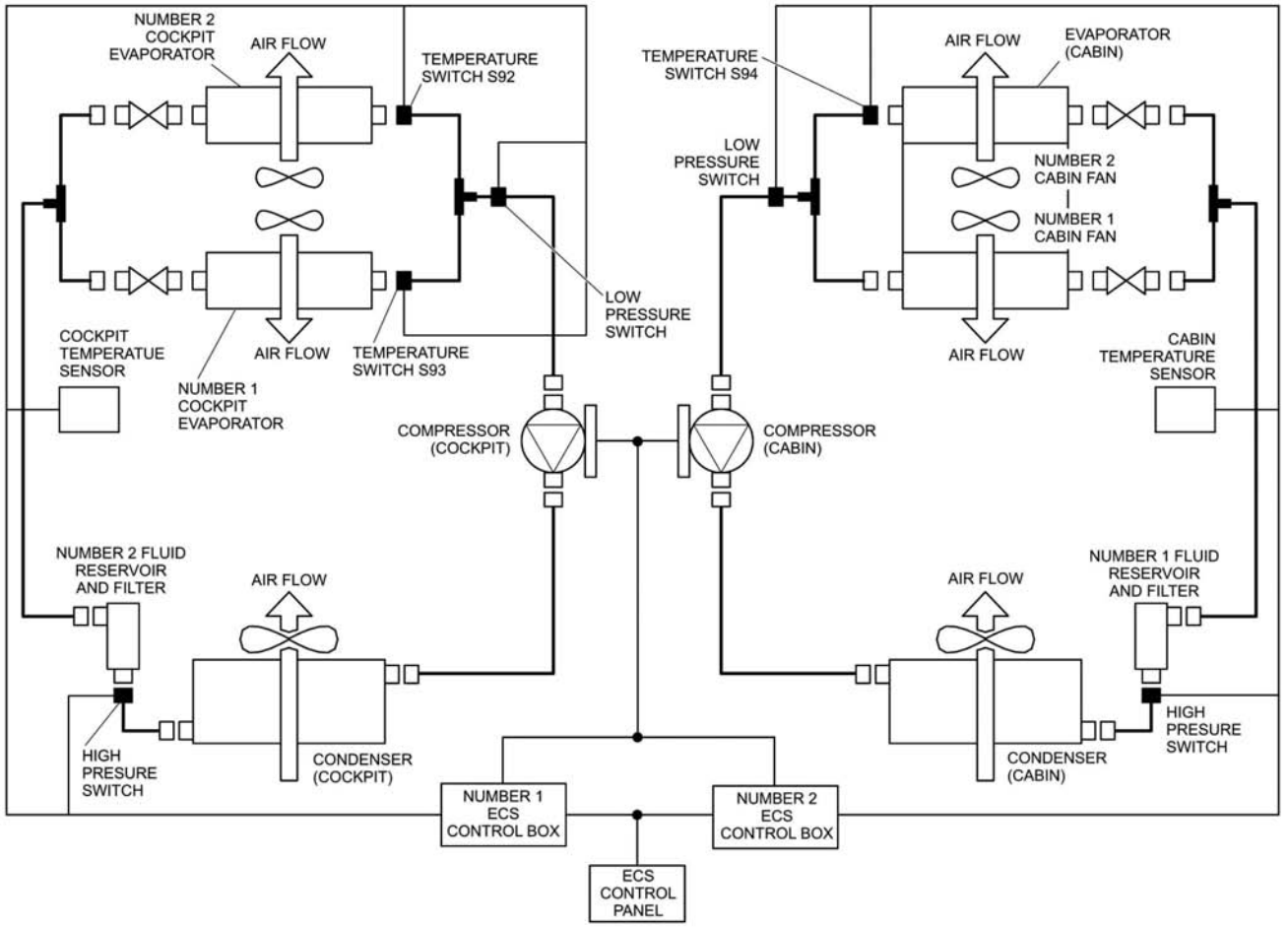
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HEATING SYSTEM ELECTRICALSCHEMATIC

SECTION 90 - INTEGRATED ENVIRONMENTAL CONTROL SYSTEM (ECS)

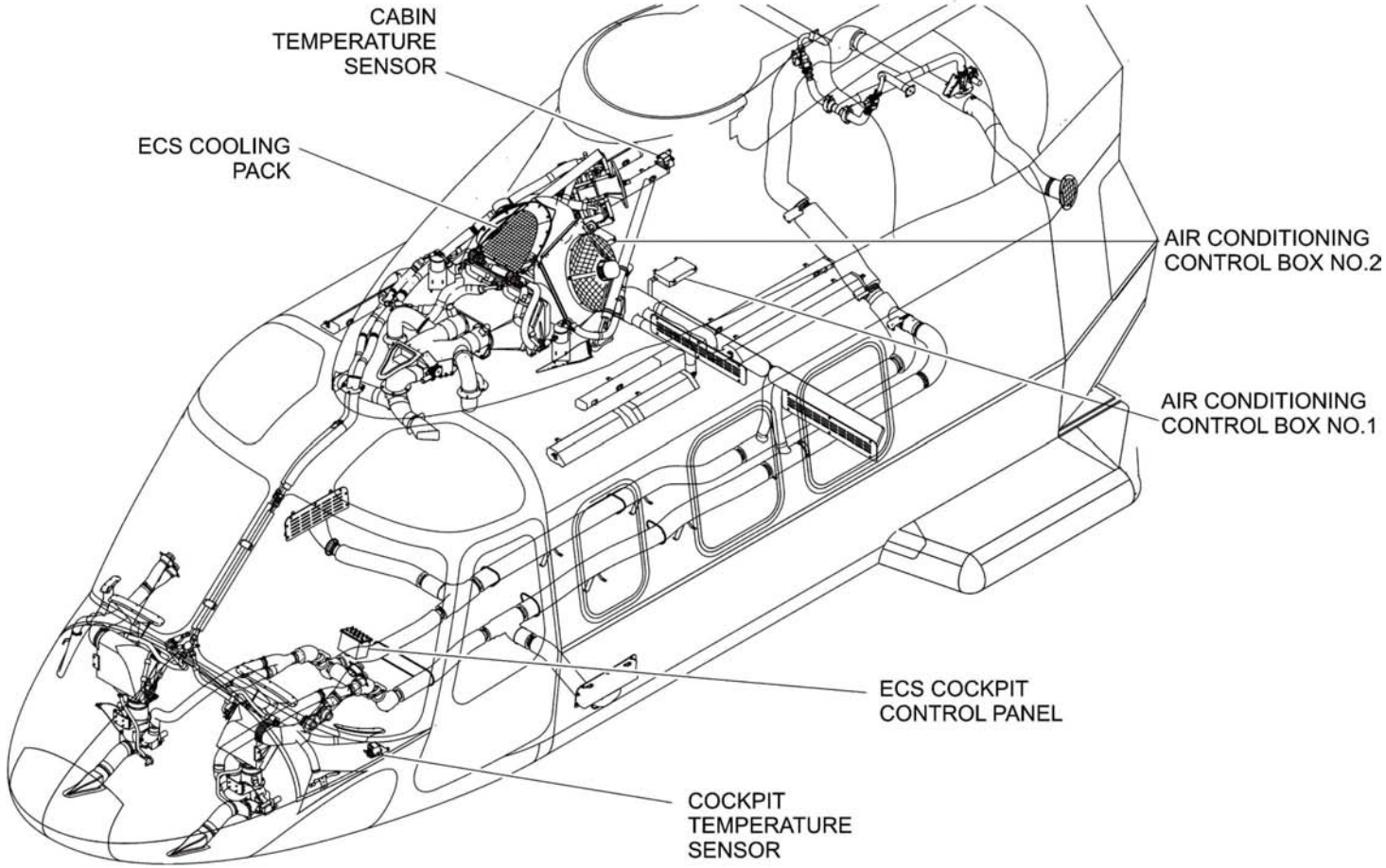
DESCRIPTION

The Air conditioning System is provided by two vapour cycle systems, dedicated respectively to the cockpit and the cabin zones.
The two vapour cycle systems operate with the HFC 134a refrigerant, chemical name tetrafluorethane and also known as Freon.
Freon is gaining in customer acceptance worldwide as a substitute for CFC fluids.
This fluid is non – toxic, non – flammable and non – aggressive to the ozone layer.
It is commonly used in domestic fridges and in automotive applications.
Each vapour cycle loop can be divided into three different sub–assemblies: compressor, condenser and evaporator.
The two compressors are mechanically driven by the main transmission gearbox through two dedicated pulleys and clutches.
The condenser consists of two identical sub – assemblies on each side of the pylon fairing.
The evaporator consists of different sub – assemblies for the cockpit and the cabin.
It allows the cooling of re-circulated air before being distributed in the cabin and in the cockpit.
The cockpit and the cabin temperature is selected by the crew on the control panel.



ICN-39-A-219000-G-A0126-00235-A-01-1

INTEGRATED ENVIRONMENTAL CONTROL SYSTEM-SCHEMATIC



ICN-39-A-219000-G-A0126-00236-A-01-1

CABIN VENTILATION GENERAL SCHEMATIC

OPERATION

When the main transmission rotates, also the two pulleys rotate but the compressors start only if the relative clutches are engaged.
The two clutches engage the compressors when the air conditioning system is set.
The thermodynamic process of the vapour cycle has the following steps:

- One compressor is fitted for each vapour cycle to compress the gaseous refrigerant and drive it from the evaporator to condenser (BC line).
  The condenser heat exchanger induces a refrigerant constant pressure phase change from gas to liquid.
  Condensation of the gaseous refrigerant is achieved by a heat transfer from the refrigerant to the heat sink, in this case external ambient air, passing through the heat exchanger. A dynamic flow of ambient air is driven through the heat exchanger by ram air action and by the axial flow condenser fan. The liquid refrigerant is then recovered inside the receiver (point D).
- The receiver filter drives has the following functions:
  - Storing a reserve of refrigerant to counteract volumetric variation of liquid refrigerant caused by variation in condensing pressure around the operational envelope; this ensures liquid feeding of the expansion valves.
  - Filtering and drying of the refrigerant preventing foreign body movement during normal operation and water contamination of the refrigerant during charging.
  - Filling the loop with additional refrigerant if needed.
  - The thermostatic expansion valve (TEV) expands the liquid refrigerant at a regulated flow rate into a mixed liquid/vapour phase before entering the evaporator (point A).
  - The evaporator with heat exchanger induces a refrigerant constant pressure phase change from liquid to gas, entirely evaporating the low – pressure liquid refrigerant expanded by TEV.
    The necessary heat for vaporization is transferred from the cockpit or cabin change air passing through the air side of the matrix.
    Change air to be cooled is drawn from outside ambient or recirculated from the cabin or cockpit environment and blown through the evaporator heat exchanger by the axial flow evaporator fan. In this way, air supplied to the cockpit or cabin distribution system is cooled according to demand (point B).

There are two control boxes. Each control box drives vapour cycle. It monitors the sensors and warns the crew when a fault occurs.
The control principle is based on the comparison of the real temperature measured by the cabin temperature sensor (or by the cockpit temperature sensor) and the desired one fixed by the crew on the control panel.
From this comparison, the controller acts directly on the compressor to adapt by ON/OFF cycles the cooling production to the real need.

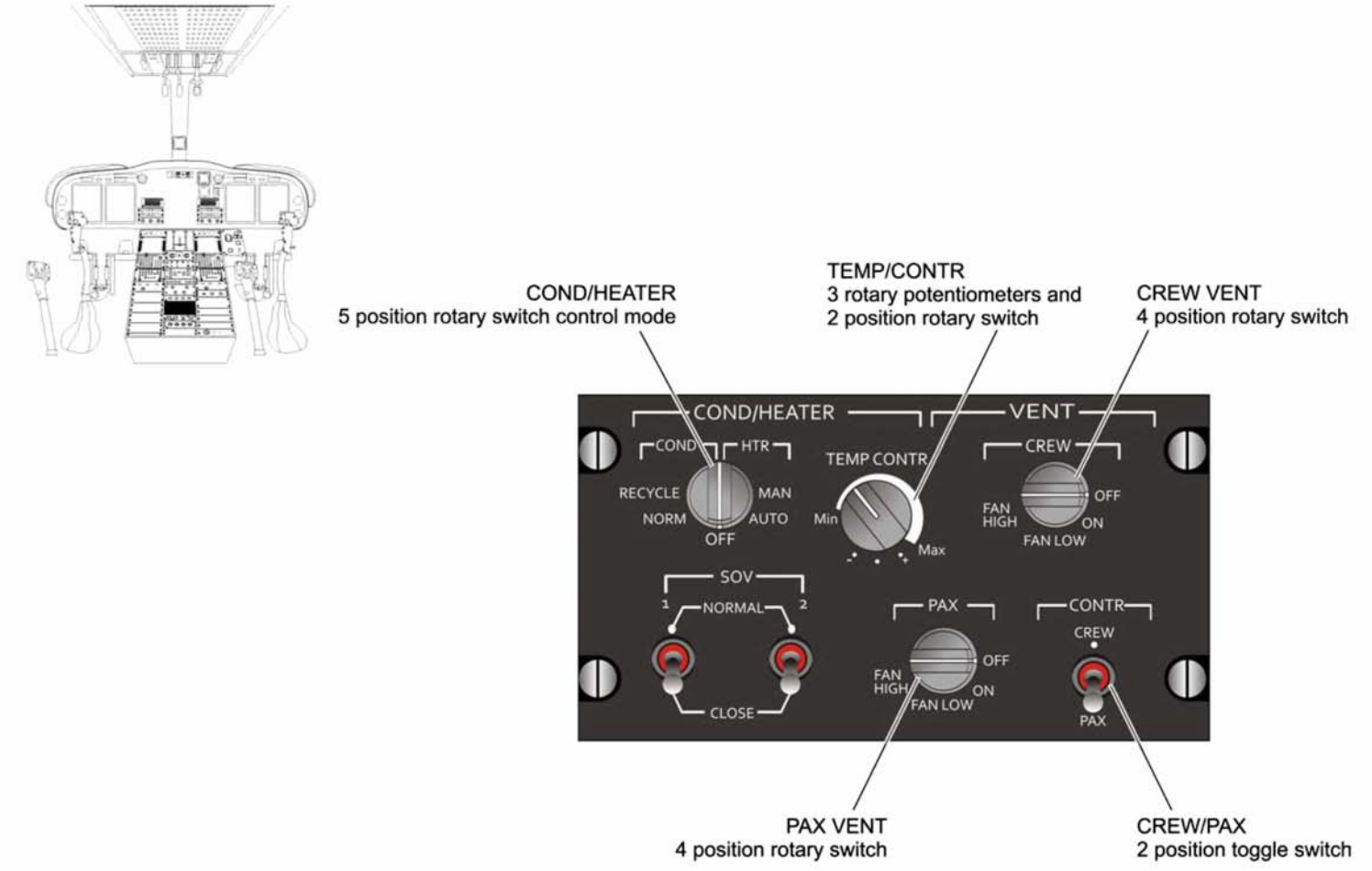
- The Thermal Switch senses the temperature of the air leaving the evaporator and sends output to the Air Condition Control Box (ACCB).
  There is a thermal switch installed in the outlet manifold of the cabin evaporator and one in the outlet manifold of both cockpit evaporator.
  There is a functional requirement to shut down the system before the evaporator freezes.
  The thermal switch consists of a K4 type thermocouple, sensing the temperature of the air at the outlet to the evaporator.
  It is manufactured from stainless steel and silicon resin. The resistance varies proportionally with the temperature sensed in the airflow.
  When the resistance reaches the level corresponding to the trip level, the switch opens and initiates the system failure condition.
- The Low Pressure switch protects against damage which may occur on the low pressure port of the loop (between TEV outlet and compressor inlet) if the evaporator pressure falls below the minimum acceptable limit.
  System protection is achieved by switching open an electrical circuit when the sensed pressure falls to the pre – set level.
  The low pressure switch is mounted at a circuit junction block downstream of the evaporators.

- The High Pressure switch is part of the drier filter assembly and prevents any damage which may occur on the high pressure part of the loop (between compressor outlet and TEV inlet) if the condensing pressure exceeds the maximum acceptable limit.
  System protection is achieved by switching open an electrical circuit when the sensed pressure rises to the pre-set level. The high pressure switch is mounted at inlet to the receiver filter drier unit.

CONTROLS AND INDICATORS

The air conditioning controls are housed on the ECS Cockpit Control Panel that is located on the console and operated by pilots.
The operative functions and the malfunction are displayed in the CAS message window of the MFD.
The ECS Cockpit Control Panel provides the following functions:

- COND:
  - OFF: the heating / air conditioning system is not operative.
  - NORM: the air conditioning system is operative.
  - RECYCLE: the air conditioning system is operative with recirculating air activated automatically.
- TEMP CONTR: it controls the cockpit and cabin temperature in the air conditioning.
- (-) or (+): inoperative when in the air conditioning mode
- CONTR:
  - CREW: the crew retains authority and the TEMP CONTR potentiometer controls the cockpit and the cabin temperature either the NORM or the RECICLE position set.
  - PAX: the TEMP CONTR potentiometer controls the cockpit temperature while the cabin control panel potentiometer on the cabin control panel controls cabin temperature either the NORM or the RECYCLE position set.



ICN-39-A-219000-G-A0126-02887-A-01-1

AIR CONDITIONING SYSTEM CONTROL

CAPTIONS

The caution provided in the CAS message window is as follows:

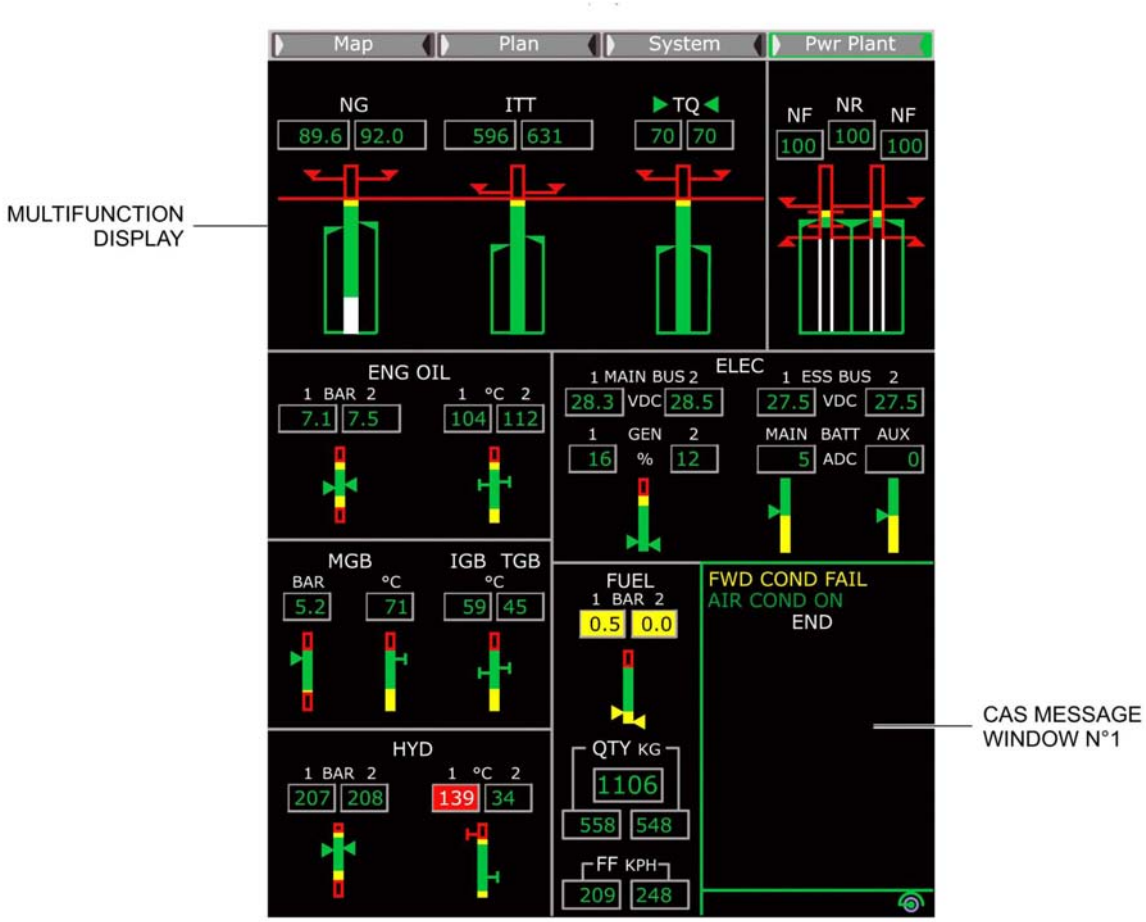
- FWD COND FAIL: when the cockpit air conditioning system fails.
- AFT COND FAIL: when the cabin air conditioning system fails.
- FWD – AFT COND FAILS: when the cockpit – cabin air conditioning system fails.

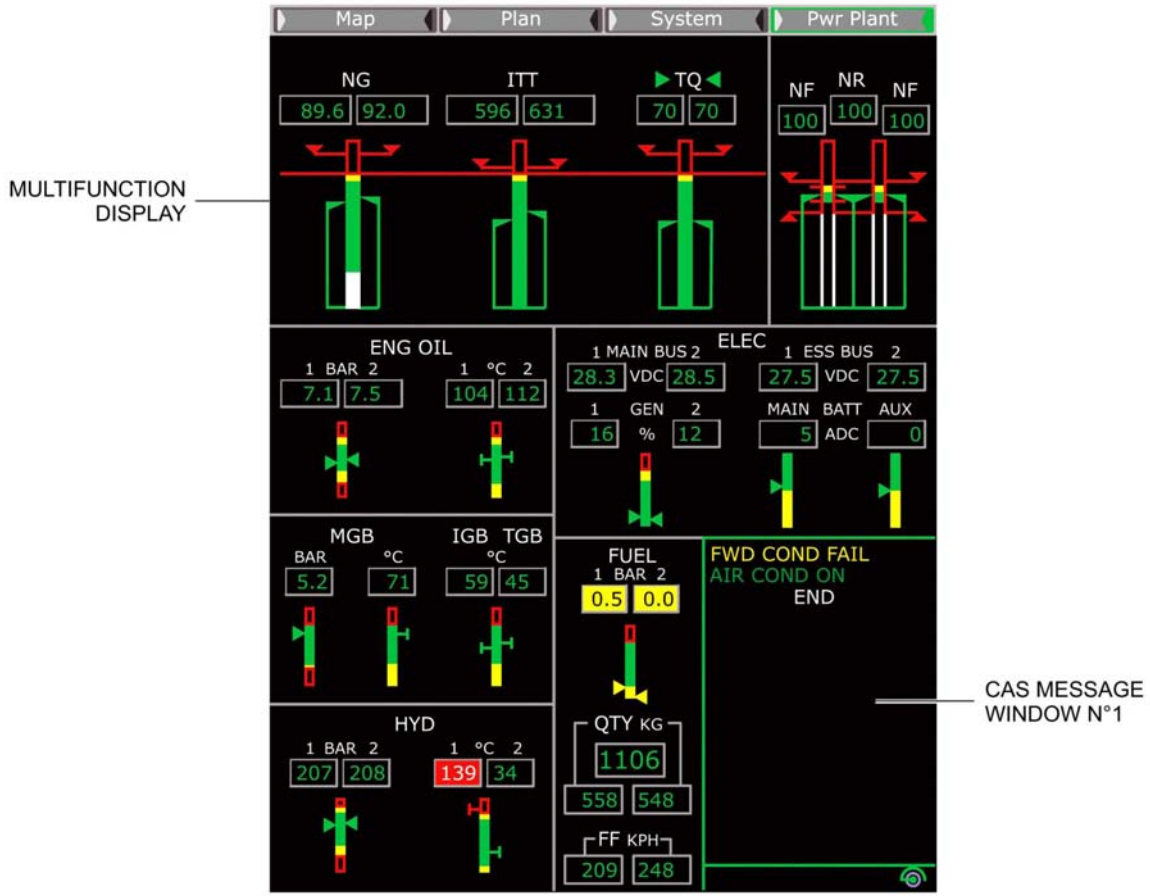
The advisory provided is as follows:

- AIR COND ON: when the air conditioning system is operative.

LEADING PARTICULARS

- COCKPIT FLAPPER VALVES..... see Section 21-20
- COCKPIT FANS (2 each)..... see Section 21-20
- CABIN FLAPPER VALVE..... see Section 21-20
- CABIN FAN 1..... see Section 21-20
- CABIN FAN 2..... see Section 21-20 except for 28 VDC NON ESS BUS 2
- ECS COCKPIT CONTROL PANEL..... see Section 21-20
- AIR CONDITIONING CONTROL BOX 1:
  - Nominal supply voltage..... 28 VDC (NON ESS BUS1)
- AIR CONDITIONING CONTROL BOX 2:
  - Nominal supply voltage..... 28 VDC (NON ESS BUS2)
- CABIN TEMPERATURE SENSOR:
  - Power supply..... 28 VDC (NON ESS BUS 2)
  - Current consumption..... 0.1 A max
  - Type probe PT100 Class B..... 1000Ω at 0°C
  - Temperature range..... - 40°C to +100°C
  - Response time..... ≤ 15s
- COCKPIT TEMPERATURE SENSOR ..... as above except for 28 VDC (NON ESS BUS 1)
- FORWARD CONDENSER FAN:
  - Nominal supply voltage..... 28 VDC (NON ESS BUS 1)
  - Operating temperature..... - 40°C to +85 °C
  - Current consumption..... 32 A max
  - Nominal rotation speed..... 3500 rpm
- AFT CONDENSER FAN:
  - Nominal supply voltage.....
  - Operating temperature.....
  - Current consumption.....
  - Nominal rotational speed..... 3500 rpm


Map
Plan
System
Pwr Plant
NG
89.6
92.0
ITT
596
631
TQ
70
70
NF
100
NR
100
NF
100
MULTIFUNCTION DISPLAY
ENG OIL
1 BAR 2
7.1
7.5
1 °C 2
104
112
ELEC
1 MAIN BUS 2
28.3
VDC
28.5
1 ESS BUS 2
27.5
VDC
27.5
1 GEN 2
16
%
12
MAIN BATT AUX
5
ADC
0
MGB
BAR
5.2
°C
71
IGB TGB
°C
59
45
FUEL
1 BAR 2
0.5
0.0
FWD COND FAIL
AIR COND ON
END
HYD
1 BAR 2
207
208
1 °C 2
139
34
QTY KG
1106
558
548
FF KPH
209
248
CAS MESSAGE WINDOW N°1

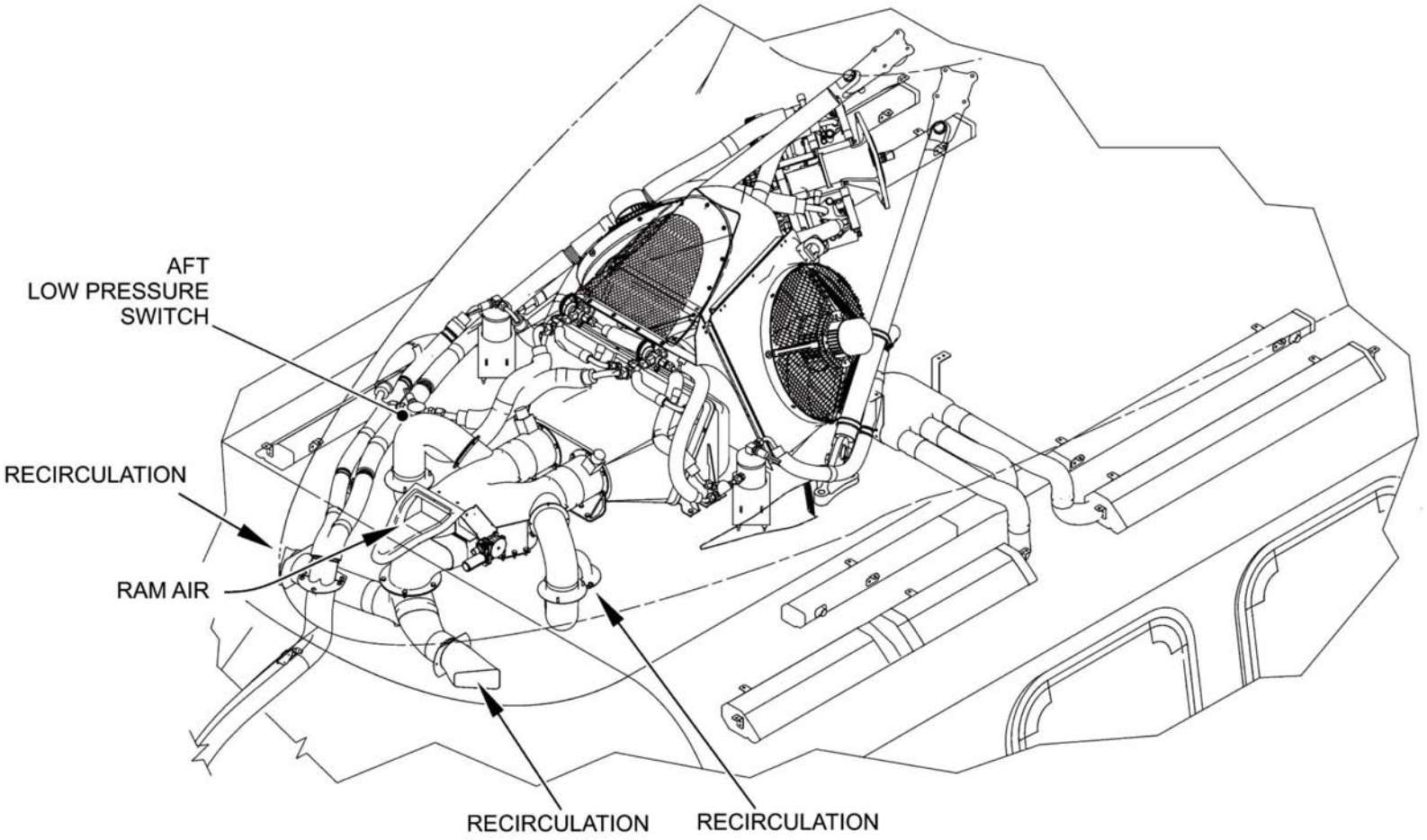


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AIR CONDITIONING SYSTEM INDICATING

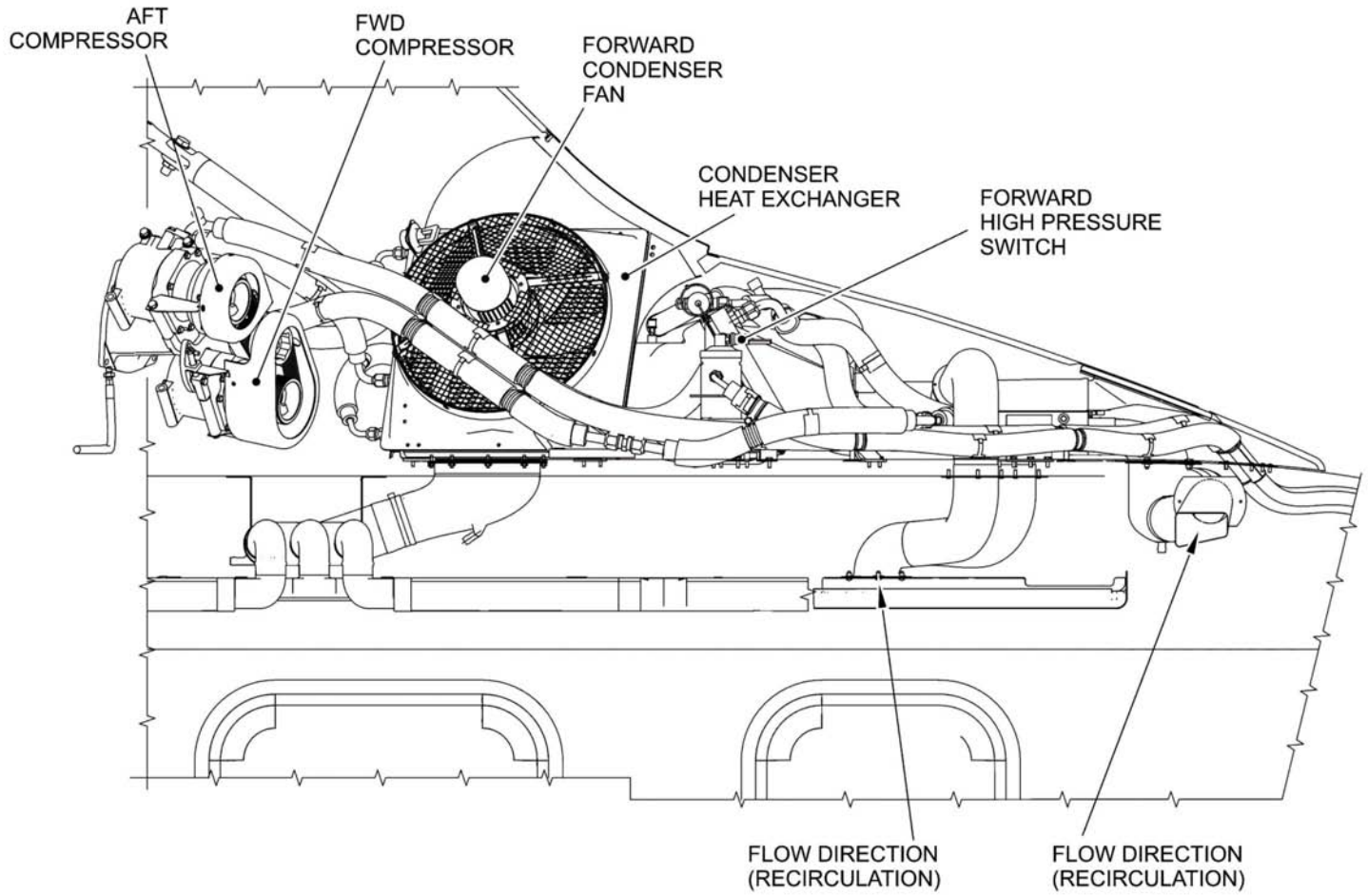
- FORWARD HIGH PRESSURE SWITCH:
  - Nominal supply voltage..... 28 VDC (NON ESS BUS 1)
  - Single contact switch..... normally closed
  - Switch OFF..... at 29 bar g (± 0.7)
  - Switch ON..... at 24 bar g (± 0.7)
- AFT HIGH PRESSURE SWITCH:
  - Nominal supply voltage..... 28 VDC (NON ESS BUS 2)
  - Single contact switch..... normally closed
  - Switch OFF..... at 29 bar g (± 0.7)
  - Switch ON..... at 24 bar g (± 0.7)
- FORWARD LOW PRESSURE SWITCH:
  - Nominal supply voltage..... 28 VDC (NON ESS BUS 1)
  - Single contact switch..... normally closed
  - Switch OFF..... at 0.5 bar g (± 0.2)
  - Switch ON..... at 0.9 bar g (± 0.2)
- AFT LOW PRESSURE SWITCH:
  - Nominal supply voltage..... 28 VDC (NON ESS BUS 2)
  - Single contact switch..... normally closed
  - Switch OFF..... at 0.5 bar g (± 0.2)
  - Switch ON..... at 0.9 bar g (± 0.2)
- FORWARD THERMAL SWITCH (2 EACH):
  - Nominal supply voltage..... 28 VDC (NON ESS BUS 1)
  - Temperature range..... - 65°C to 175 °C
  - Precision..... ± 2°C
  - Single contact switch..... normally closed
  - Switch open setting..... 1°C
  - Switch close setting..... 11°C
- AFT THERMAL SWITCH:
  - Nominal supply voltage..... 28 VDC (NON ESS BUS 2)
  - Temperature range..... - 65°C to 175 °C
  - Precision..... ± 2°C
  - Single contact switch..... normally closed
  - Switch open setting..... 1°C
  - Switch close setting..... 11°C
- AFT COMPRESSOR:
  - Refrigerant..... R134a
  - Ambient temperature..... - 40°C to + 85°C
  - Maximum refrigerant outlet pressure..... 29 bar
  - Safety thermostat:.....
    - CUT OUT… 125°C ± 3° C
    - CUT IN..... 110°C ± 6° C
- FORWARD COMPRESSOR:
  - Refrigerant..... R134a
  - Ambient temperature..... - 40°C to + 85°C
  - Maximum refrigerant outlet pressure..... 29 bar
  - Safety thermostat:.....
    - CUT OUT… 125°C ± 3° C
    - CUT IN..... 110°C ± 6° C
  - Lubricating oil..... 150 cc (see Refrigerant Oil)

- AFT COMPRESSOR CLUTCH:
  - Magnetic clutch electrical requirements..... 28 VDC (NON ESS BUS 2)
  - Current consumption..... 3 A max
  - Electrical power consumption..... 84 W max
  - Minimum engagement voltage..... 10.5 V
  - Nominal operating speed..... 3900 rpm
- FORWARD COMPRESSOR CLUTCH:
  - Magnetic clutch electrical requirements..... 28 VDC (NON ESS BUS 1)
  - Current consumption..... 3 A max
  - Electrical power consumption..... 84 W max
  - Minimum engagement voltage..... 10.5 V
  - Nominal operating speed..... 3900 rpm



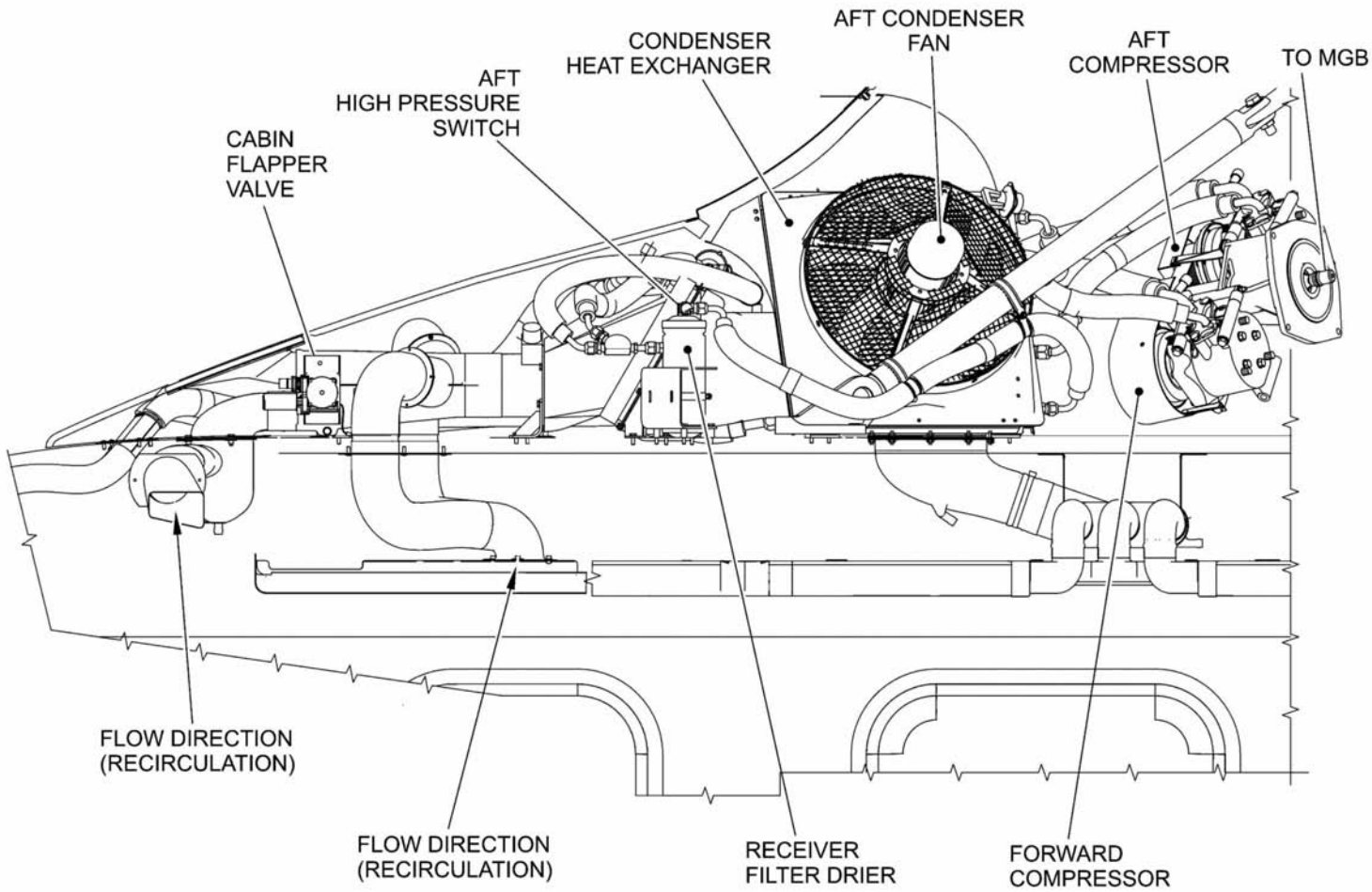
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INTEGRATED ENVIRONMENTAL CONTROL COMPONENT LOCATION (SHEET 1 OF 6)



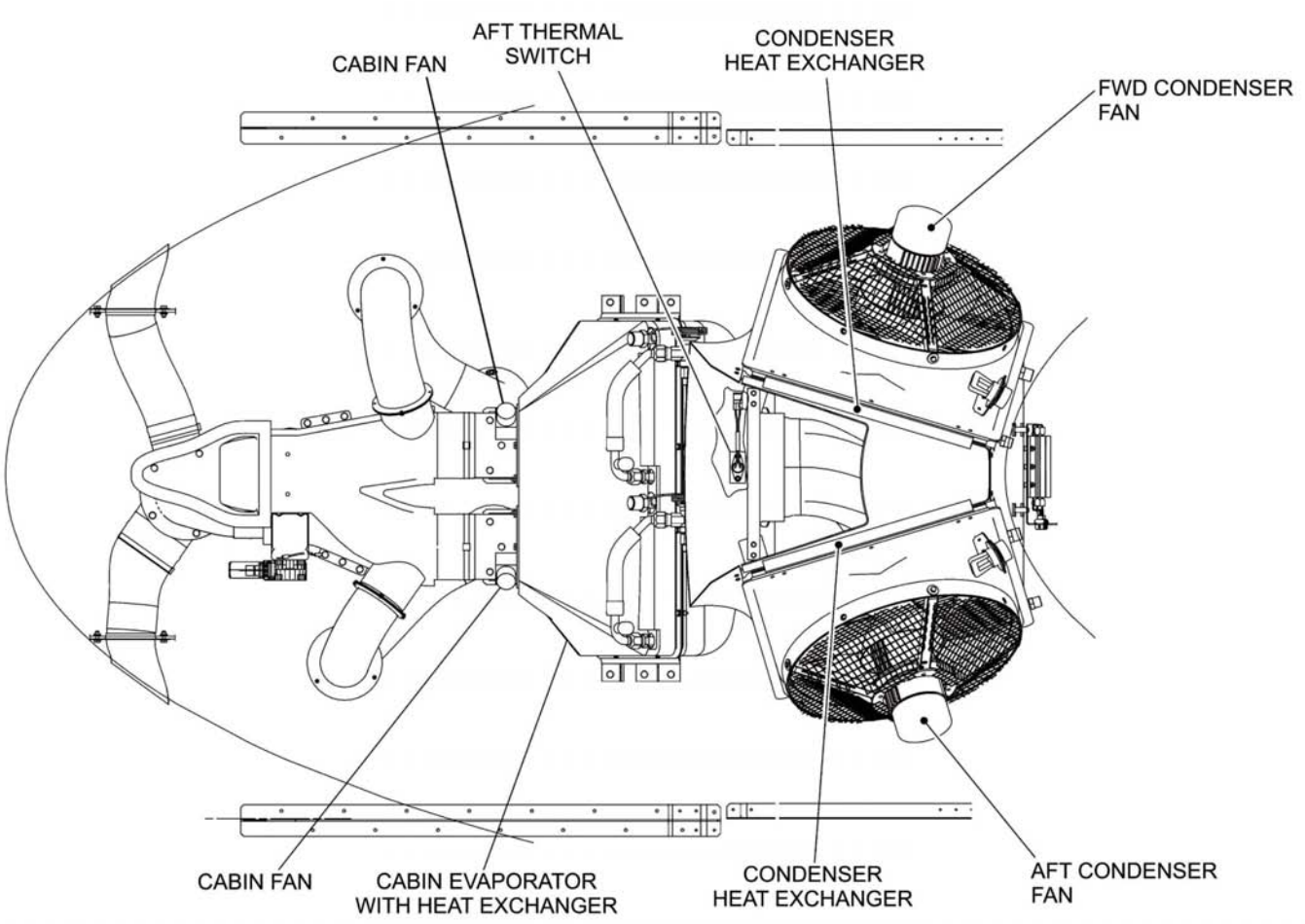
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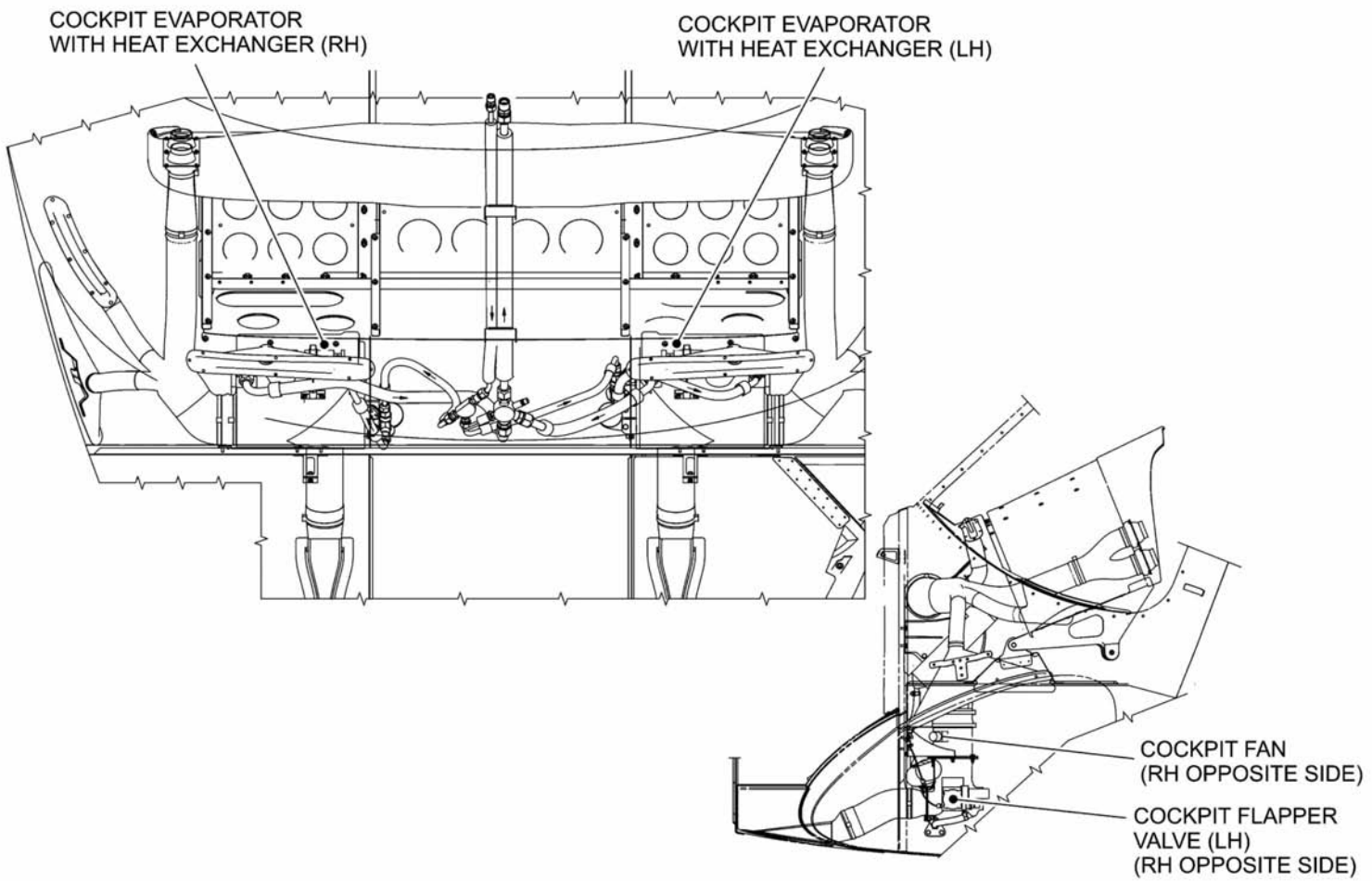
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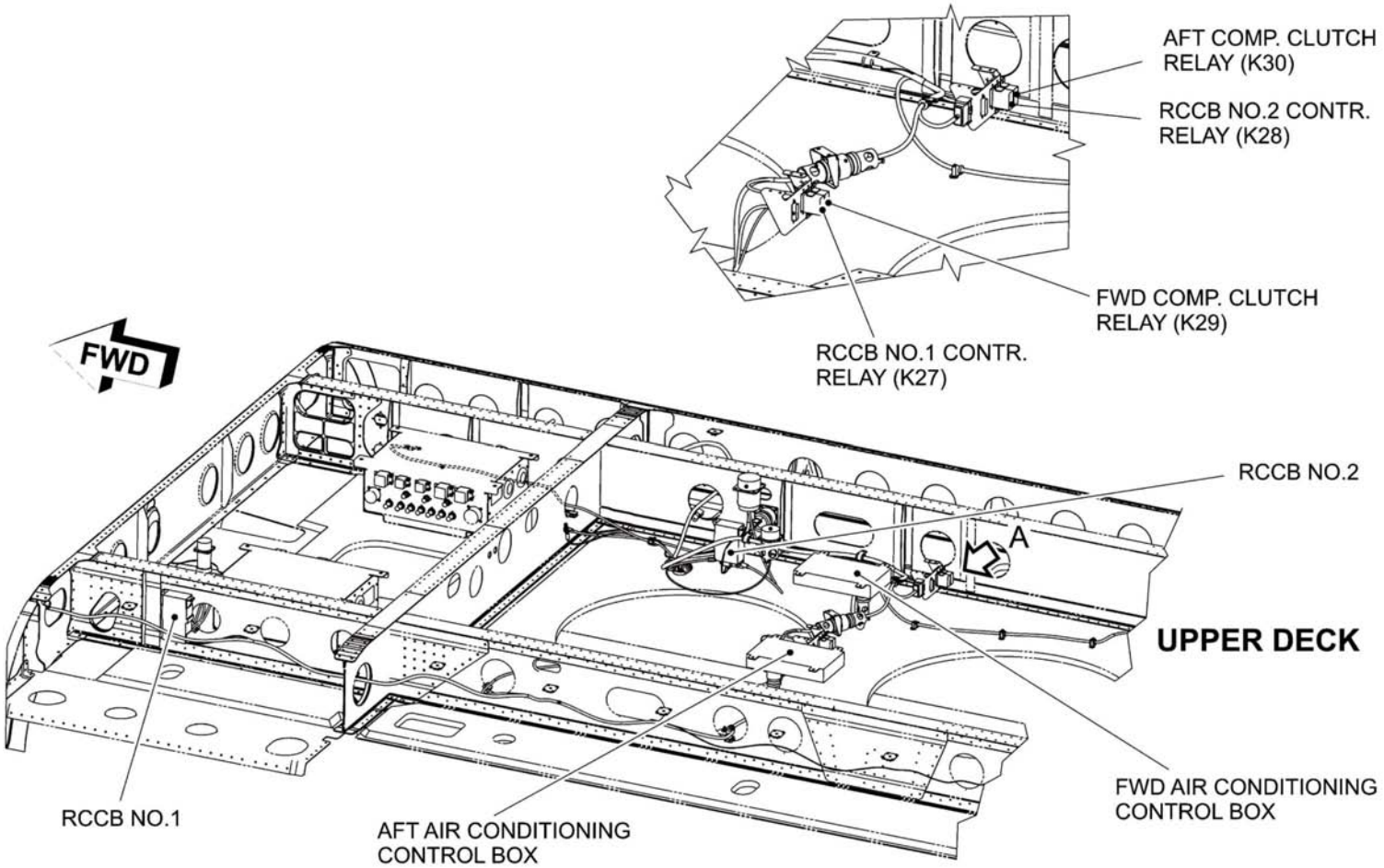
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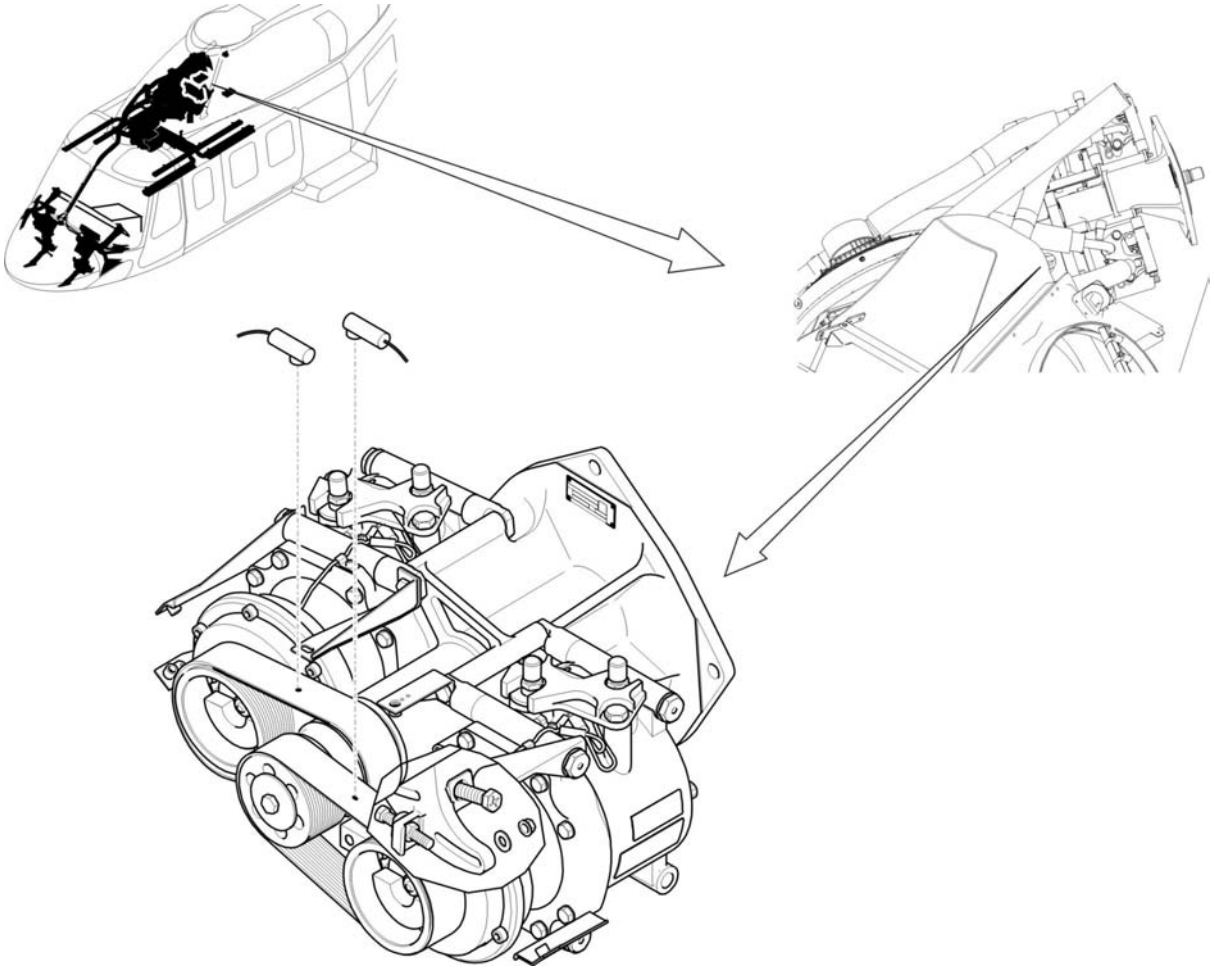
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INTEGRATED ENVIRONMENTAL CONTROL COMPONENT LOCATION (SHEET 5 OF 6)



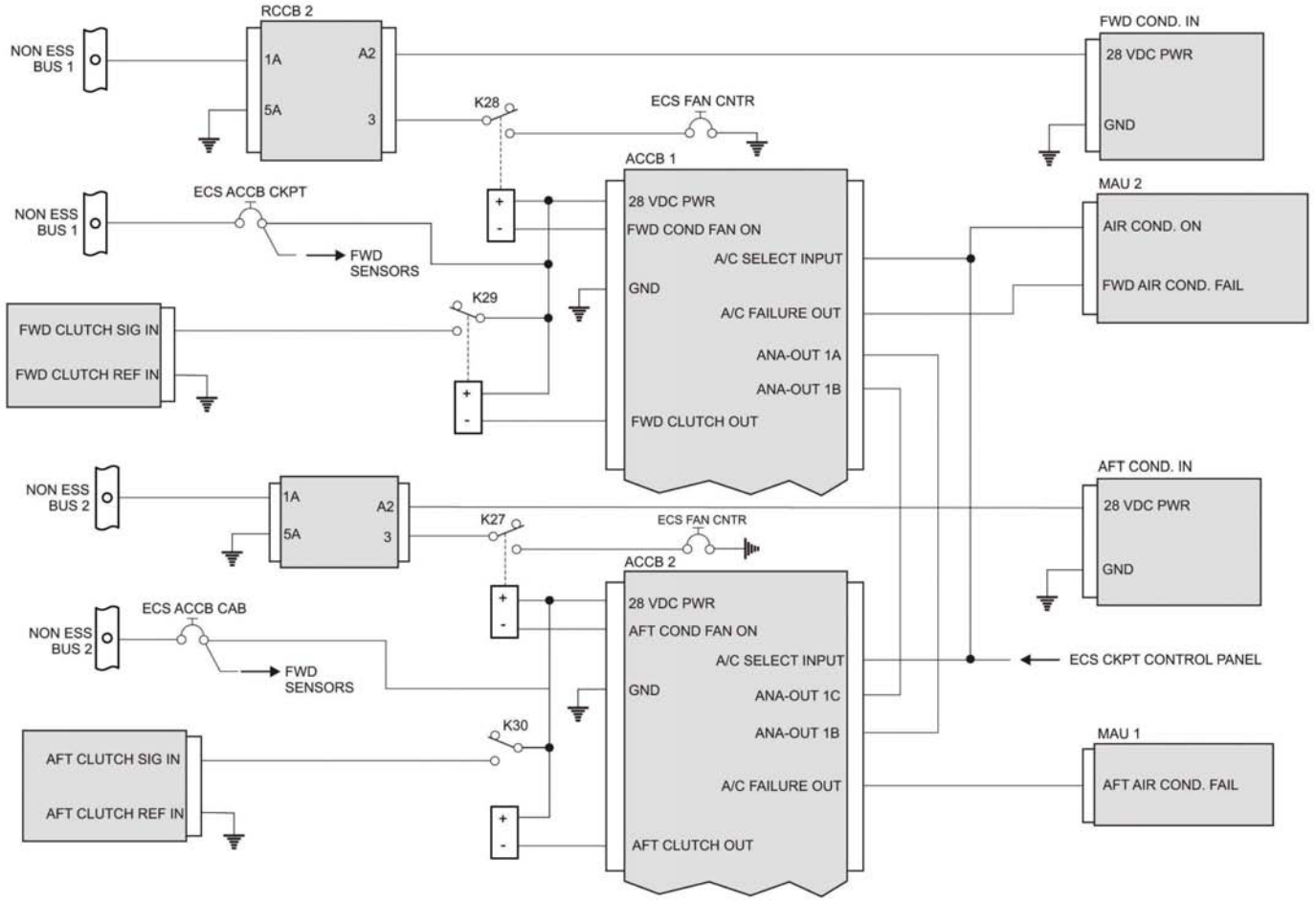
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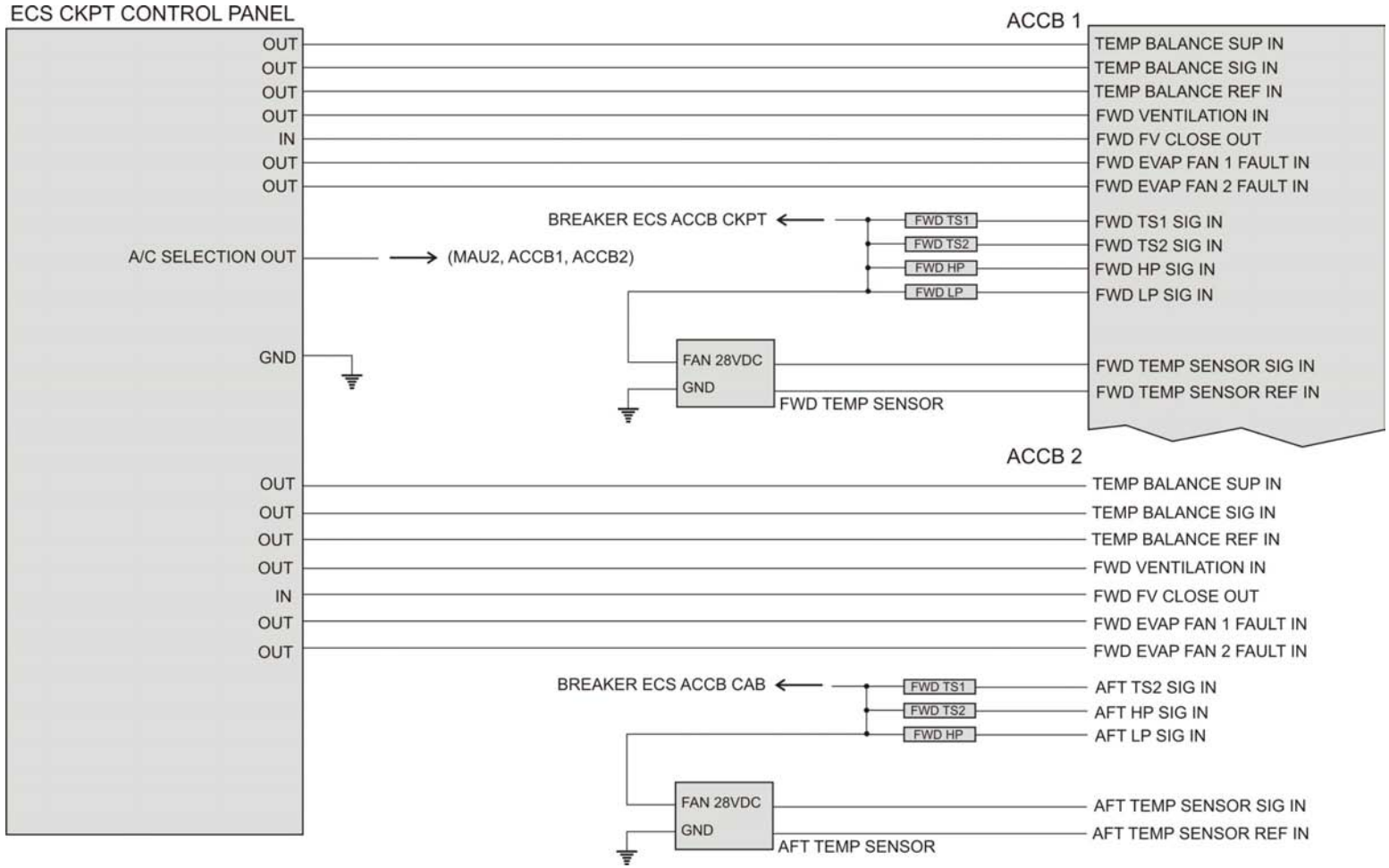
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COMPRESSOR DRIVE BELT – ADJUST



ICN-39-A-219000-G-A0126-00246-A-01-1

INTEGRATED ENVIRONMENTAL ELECTRICAL SCHEMATIC (SHEET 1 OF 2)



ICN-39-A-219000-G-A0126-00251-A-01-1

INTEGRATED ENVIRONMENTAL ELECTRICAL SCHEMATIC (SHEET 2 OF 2)

CHAPTER
22
AUTO FLIGHT

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SECTION 00 - AUTO FLIGHT

AUTOMATIC FLIGHT CONTROL SYSTEM (AFCS) – GENERAL

The AW139 Automatic Flight Control System (AFCS) is part of the PRIMUS EPIC® Integrated Avionics System (see chapter 46-00).
The AFCS is available in four configurations with increasing capabilities:

- BASIC 3 AXIS AUTOPILOT SYSTEM. The configuration provides dual pitch, roll, and yaw stabilization and attitude retention (Autopilot — AP) with Force Trim on pitch, roll, yaw and collective. No Flight Director (FD) is included (VFR only).
- 4 AXIS SYSTEM WITH BASIC 3 CUE FLIGHT DIRECTOR. It consists of a 4-axis Autopilot (Basic 3 axis Autopilot with collective Trim) and a 3-cue (pitch, roll and collective) Flight Director (no Auto Hover mode).
- 4 AXIS SYSTEM WITH ENHANCED 3 CUE FLIGHT DIRECTOR. It consists of a 4-axis Autopilot (Basic 3 axis Autopilot with collective Trim) and a 3-cue (pitch, roll and collective) Flight Director including Hover/ Velocity Hold mode.

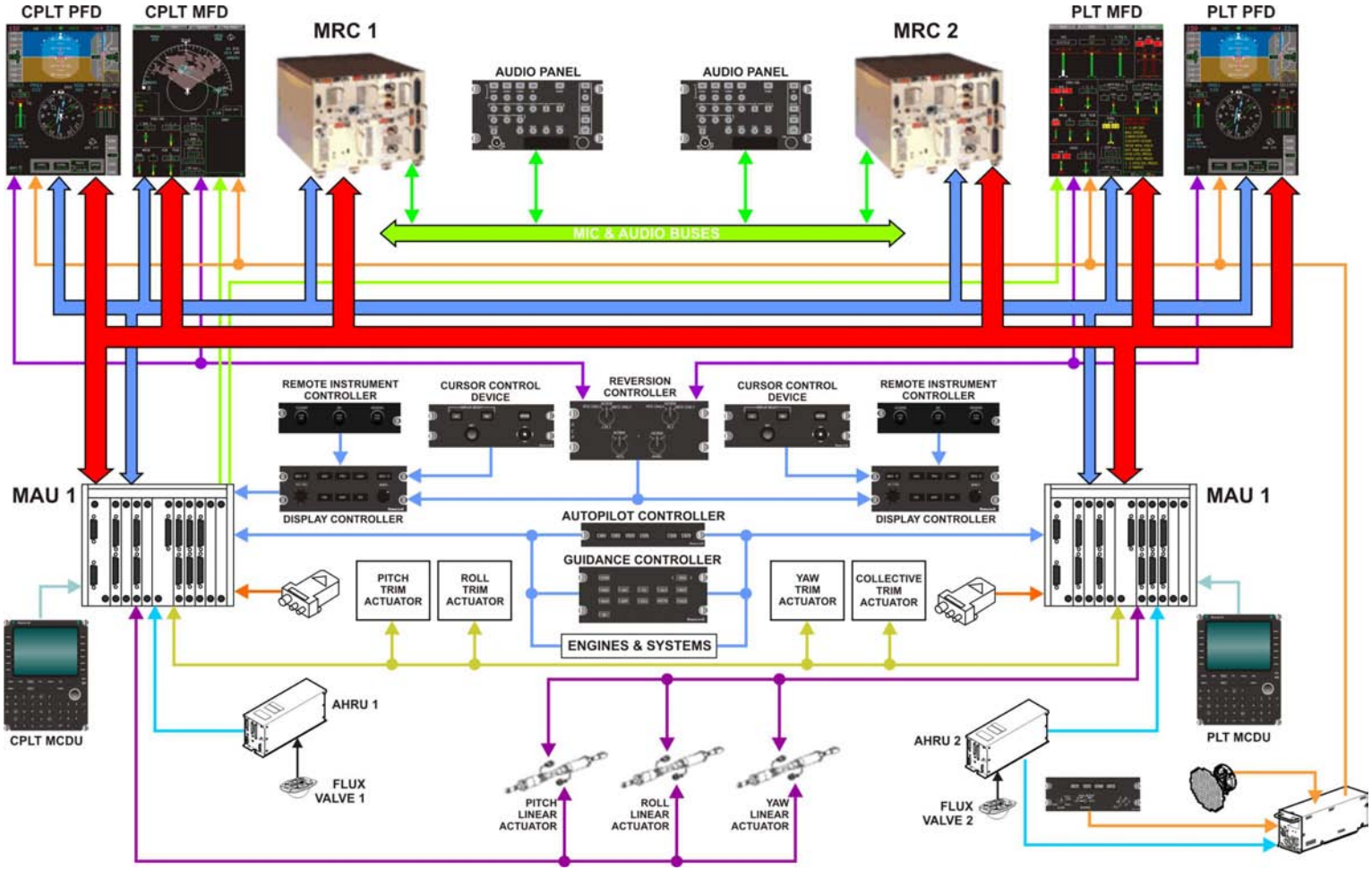
The AFCS is a system fully duplicated for redundancy and consists of:

- Autopilot No.1 (AP1)
- Autopilot No.2 (AP2)
- Flight Director No.1 (FD1)
- Flight Director No.2 (FD2)

The AFCS is an active-active system because both Autopilots operate on the flight controls at the same time.
The No.1 and No.2 AP and FD systems normally operate together. Each AP and FD is capable of providing full functionally also in case of failure of the paired system.

The AFCS is designed to be fail-safe because autopilot malfunctions are demonstrated to be safely recoverable by the pilot flying the helicopter manually.
The pilot can override the AFCS at any time by manually operating the flight controls. The system gives the pilot full authority over flight controls regardless of whether the autopilot is engaged or disengaged.
The dual AFCS uses:

- two AFCS computers (part of MAU1 and MAU2)
- two independent sets of sensors (AHRS, ADS; RAD ALT; VOR/ILS; FMS);
- two sets of linear (series) actuators (Pitch, Roll, Yaw);
- one set of single trim (parallel) actuators (Pitch, Roll, Yaw and Collective)



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AFCS – GENERAL ARRANGEMENT

AFCS MAIN COMPONENTS

The AFCS consists of the following main components:

- four AFCS modules located inside the MAUs (two modules in each MAU) named Actuator Input/Output with Processor (AIOP) modules.
- one Autopilot Controller
- one Guidance Controller
- three sets of dual Linear Actuators
- four Trim Actuators
- eight Trim Relays
- control switches on both cyclic sticks, both collective levers, both yaw pedals and on the central console.

The Control Input/Output (CIO) and the Custom Input/Output (CSIO) modules in each MAU interface the Autopilot Controller, the Guidance Controller and the external control switches to the AIOP modules.
AFCS indications are provided on the Display Units.
The following systems provide the data necessary for AFCS operation:

- both ADS1 and ADS2
- both AHRS1 and AHRS2
- the Standby Instrument
- both Radar Altimeter 1 and Radar Altimeter 2
- both VHF NAV1 and VHF NAV2
- both FMS1 and FMS2

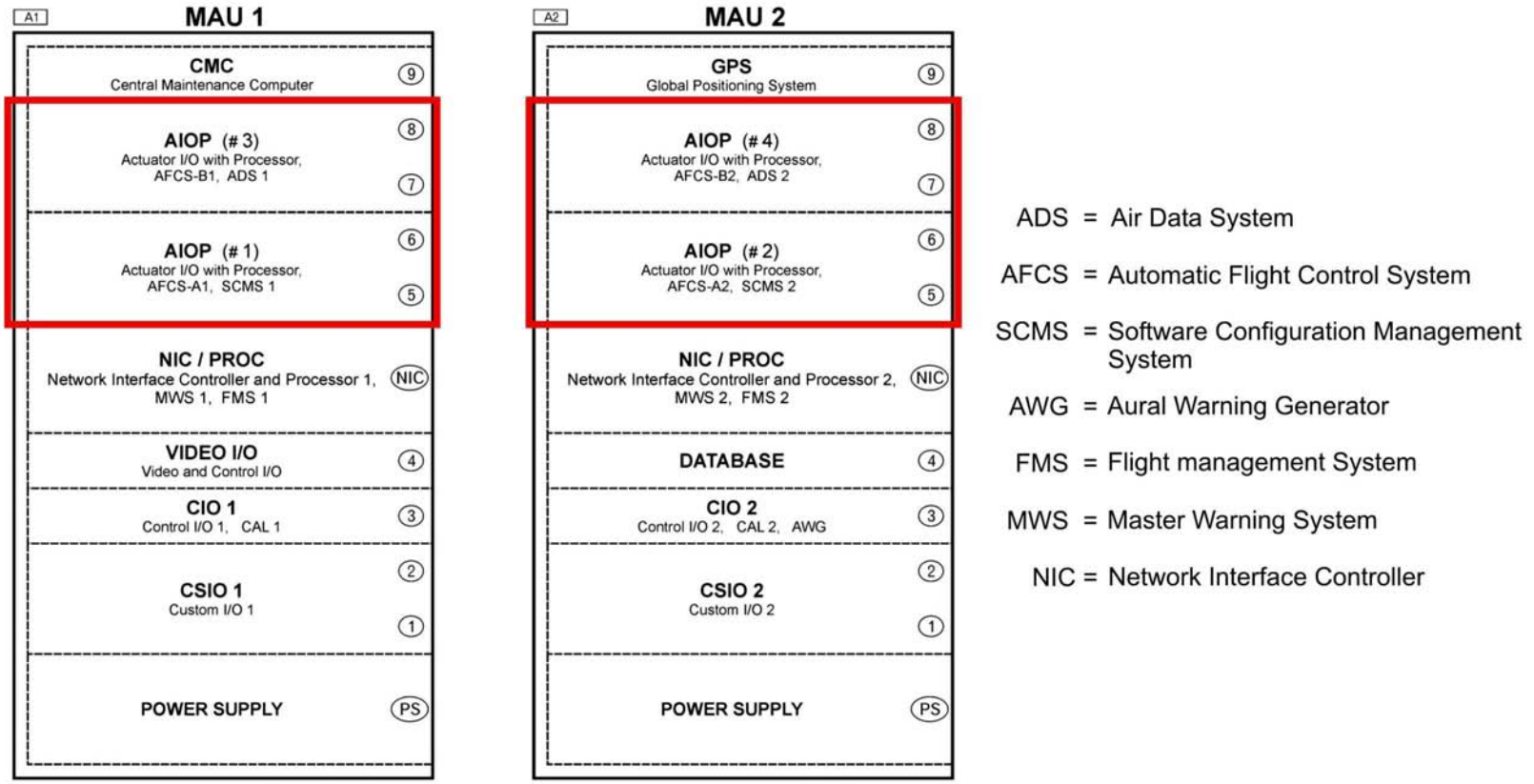
MODULAR AVIONICS UNIT (MAU)

The CIO module in each MAU supplies interface to the system Display Controller, the Cursor Control Device, the Remote Instrument Controller, the Autopilot Controller, the Guidance Controller and the Multifunction Control Display Unit (MCDU).The CIO module in MAU2 also contains the aural warning circuitry that generates alerts in the cockpit.
The CSIO modules receive data from external sensors and then transmit the data to the processor part of the AI/OP modules (AIOP-A and AIOP-B) for the actual aircraft control and monitoring calculations.
Two AFCS modules are installed in each Modular Avionic Unit (MAU) to perform AFCS computations, output commands and indication data and perform system monitoring.
The two AFCS modules —named Actuator Input/Output with Processor (AIOP) modules— in MAU1 are part of AFCS1 (AP1, FD1); the two AIOP modules in MAU2 are part of AFCS2 (AP2, FD2).
That makes the AFCS a dual-redundant computer system (AFCS1 and AFCS2) and each AFCS dual-redundant inside (channel A and channel B).
The two modules of an AFCS share the tasks and continuously monitor each other performances to positively identify an internal failure and disengage automatically.
The AIOP receives AHRS and Standby Instrument data through the ARINC 429 digital data buses.
The AIOP modules supply commands/monitors to the pitch, roll and yaw linear actuators through the Control Area Network (CAN) digital bi-directional bus. These modules also interface with the trim actuators.
Each AIOP module is a dual slot, single-lane module that contains two electrically connected circuit cards (Processor card and Actuator I/O [AIO] card). While both cards are connected to the MAU backplane, the AIO card only uses the connection for power. All data communication goes through the Processor card.
Each AIOP Channel A module (AIOP-A1 and AIOP-A2) provide:

- command function for the Trim Actuators and
- monitoring function for the associated Linear Actuators.

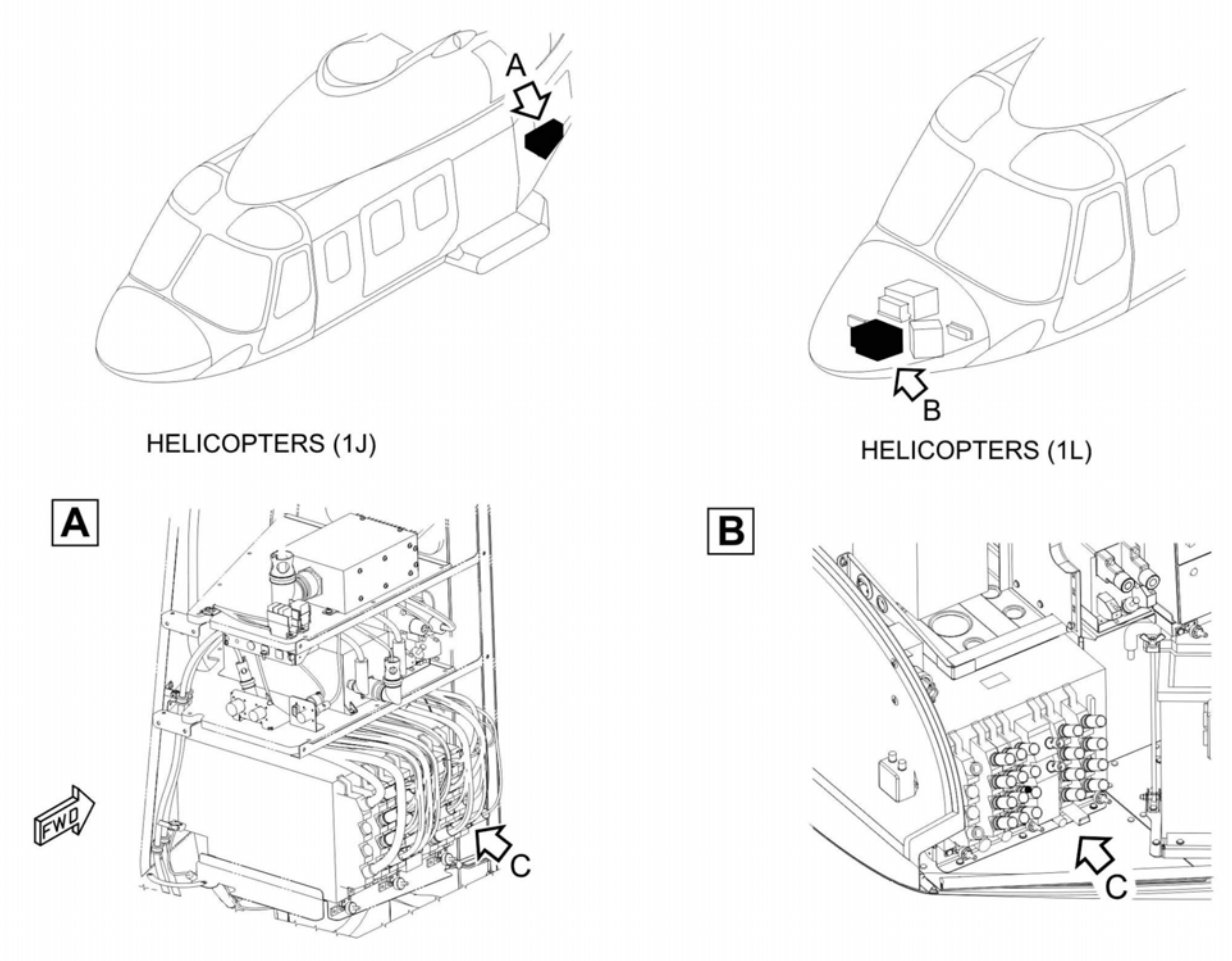
Each AIOP Channel B module (AIOP-B1 and AIOP-B2) provide:

- command function for the associated Linear Actuators
- monitoring function for the Trim Actuators.



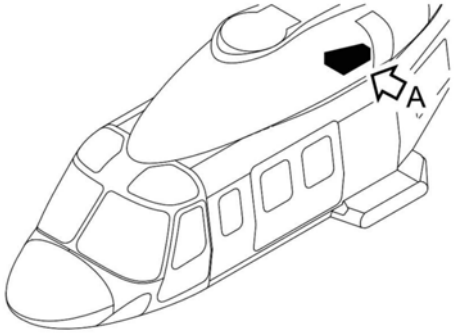
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MAU – AFCS MODULES

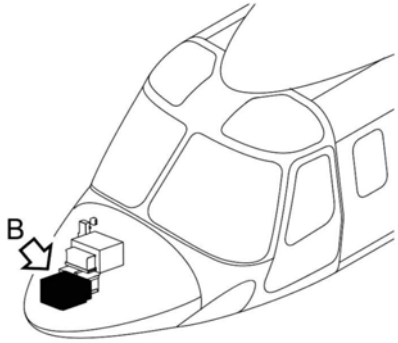


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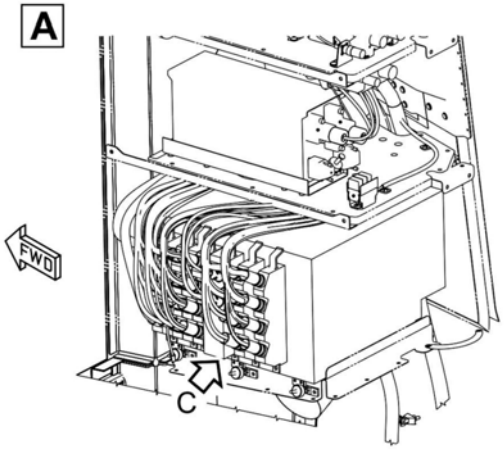
MAU 1 – LOCATION

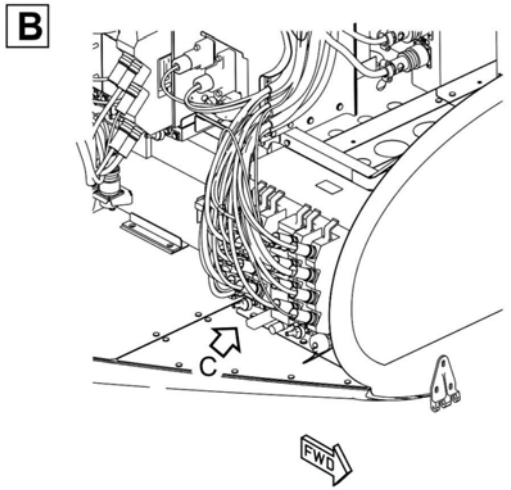


HELICOPTERS (1J)



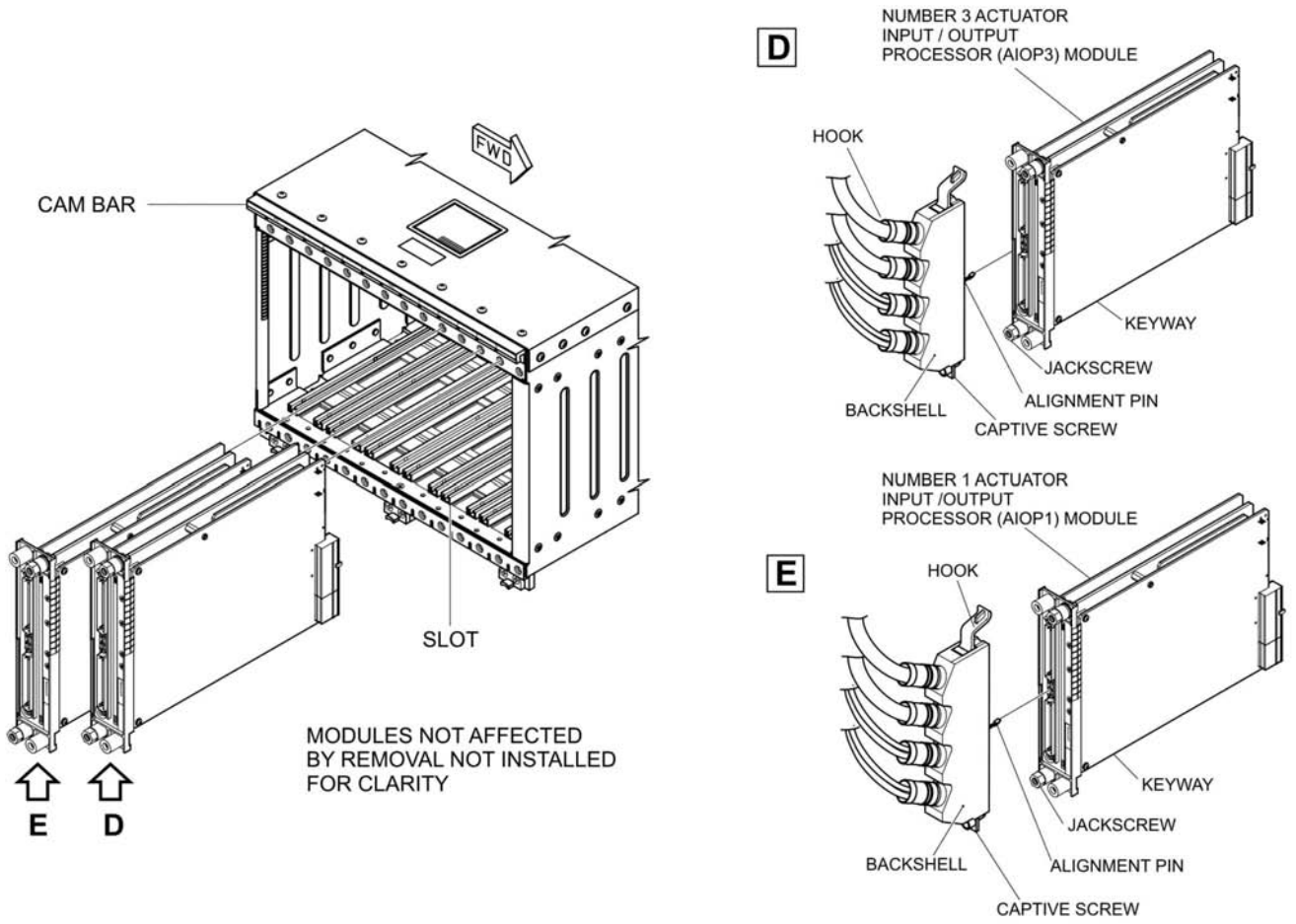
HELICOPTERS (1L)





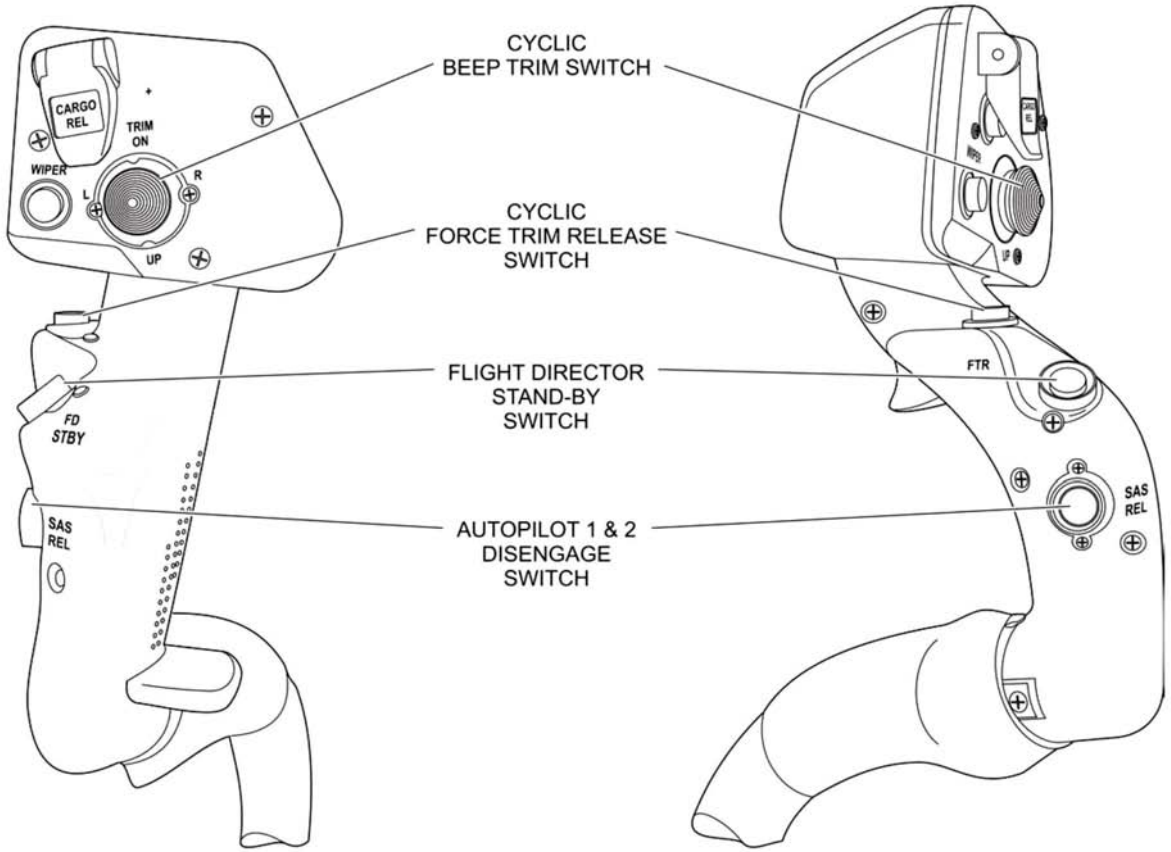
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MAU 2 – LOCATION



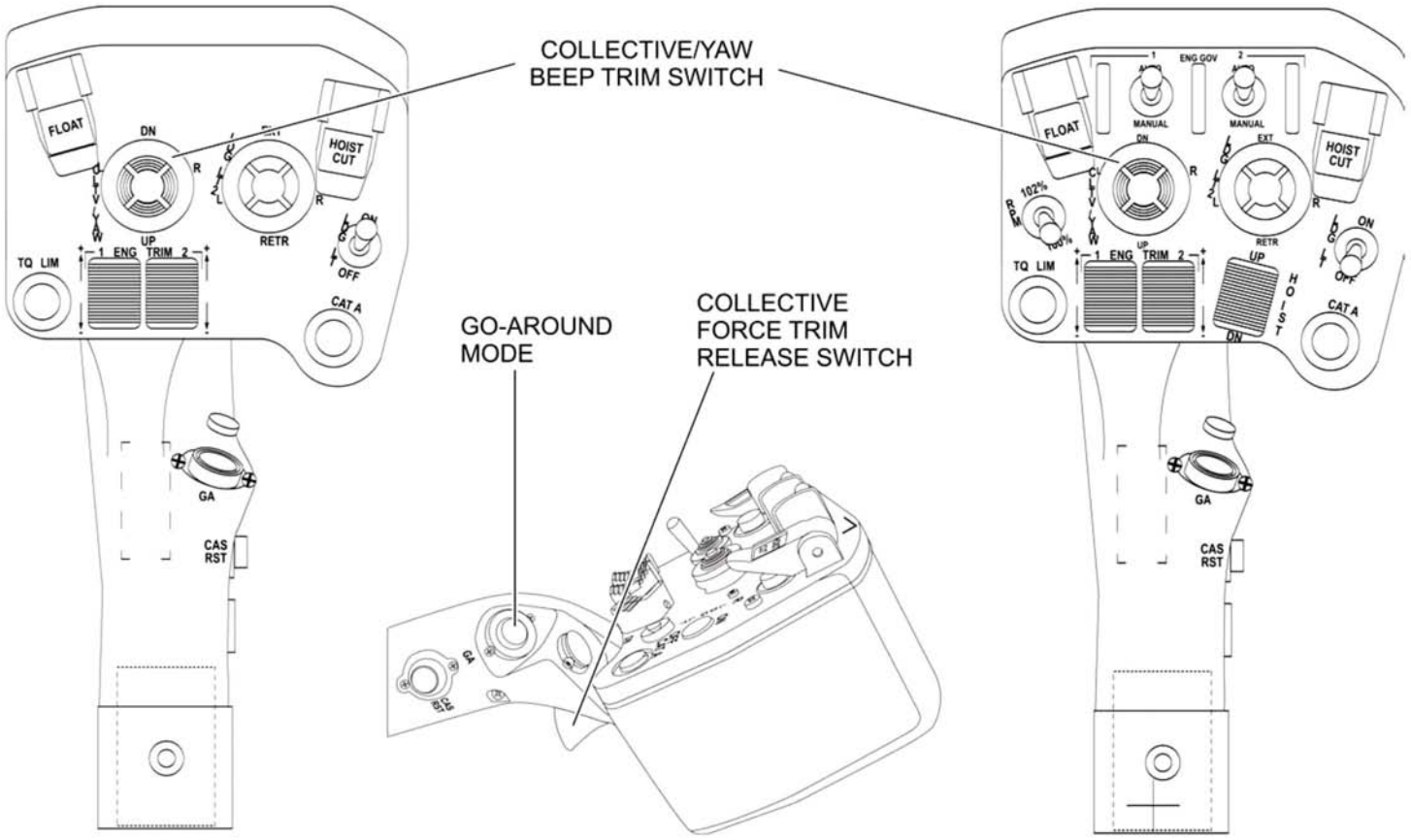
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MAU – AIOP MODULES



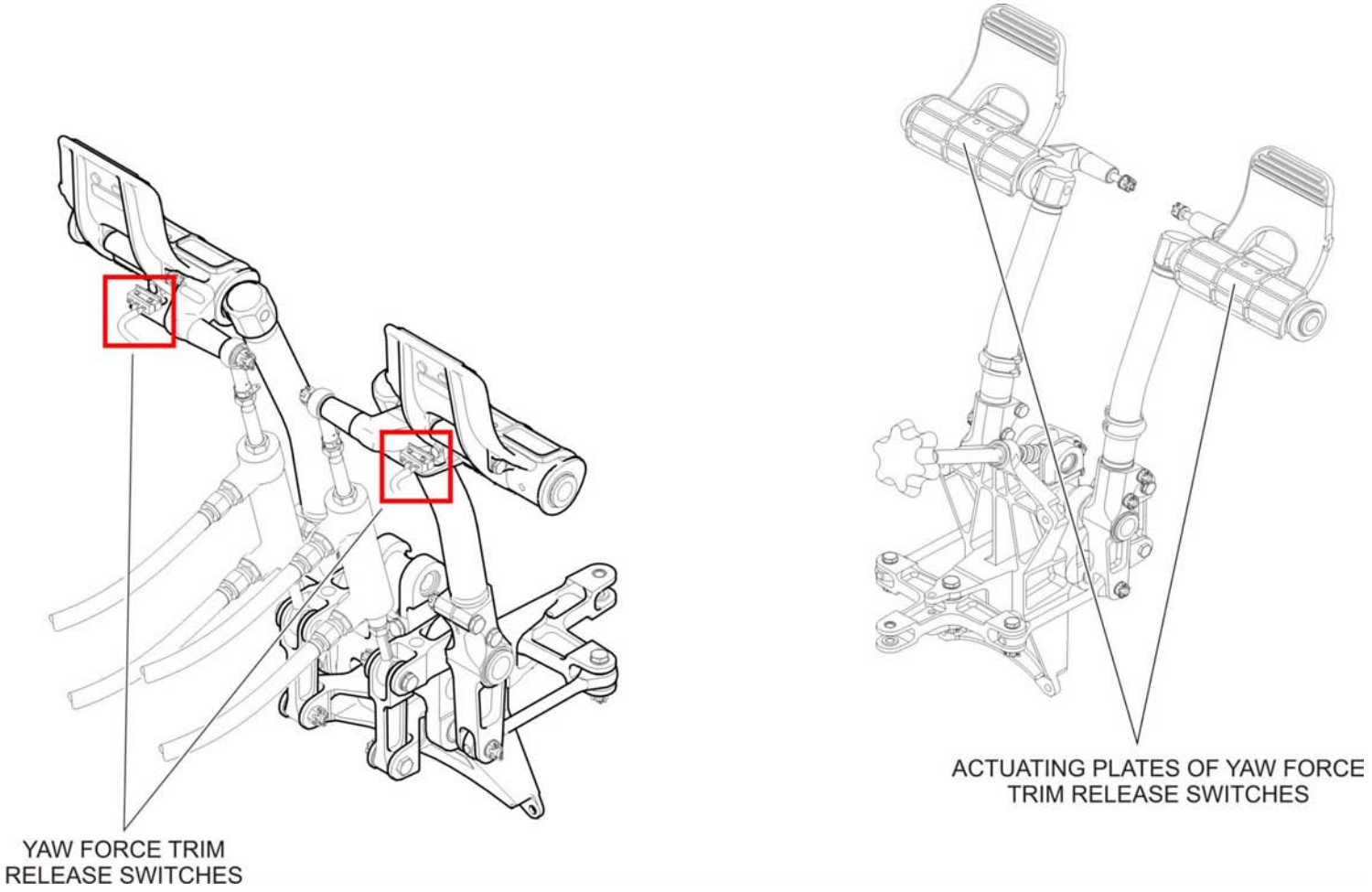
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AFCS CONTROLS – CYCLIC STICK



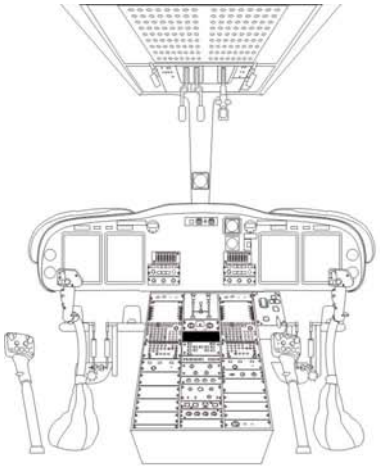
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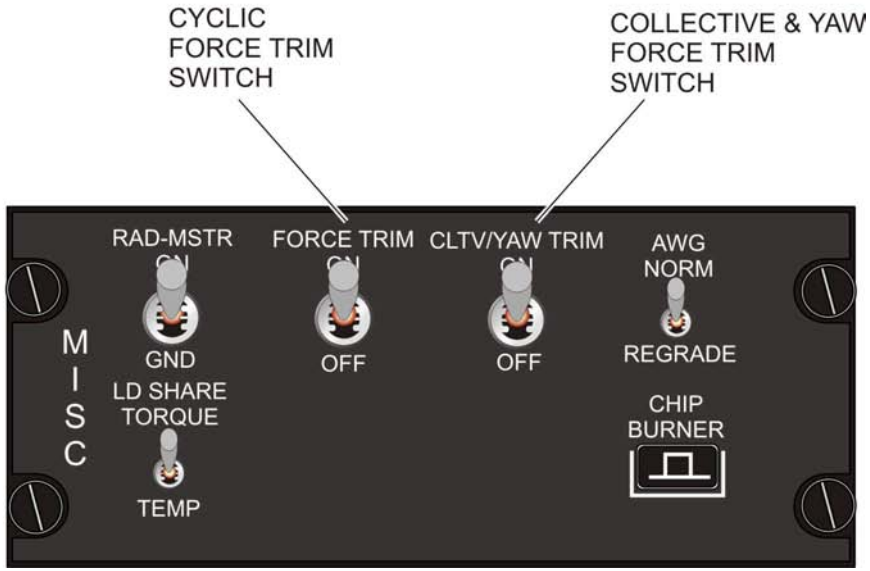
AFCS CONTROLS – COLLECTIVE LEVER



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AFCS CONTROLS – PEDALS





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AFCS CONTROLS – MISCELLANEUS PANEL

AUTOPILOT CONTROLLER

The Autopilot Controller located on the control pedestal of the cockpit provides pushbuttons for engaging and disengaging the following systems and functions:

- Autopilot system #1(AP1)
- Autopilot system #2 (AP2)
- Autopilot Pre-flight Test function (TEST)
- Autopilot/Flight Director coupling and de-coupling (CPL)
- Stability Augmentation System (SAS) mode
- Attitude retention mode (ATT)

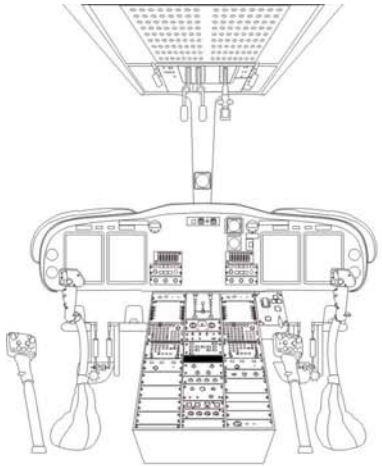
Each pushbutton on the Autopilot Controller has an associated integral green indicator light (led type) that comes on when the associated function is activate. All push-buttons on the Autopilot Controller have backlighting for Night Vision Goggle (NVG) operation.
Power for the indicators on the Autopilot Controller is 28 Vdc for daytime operation and 14 Vdc for night-time (dim) operation. A dimming circuit remotely controls the dimming of the status lights.

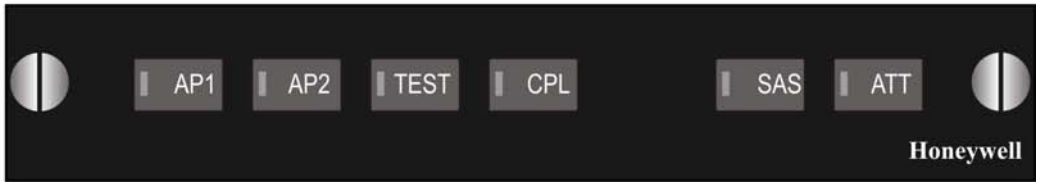
NOTE A

When either AP is engaged, a small magenta HDG bug is automatically displayed at actual aircraft heading on HSI (PFDs).

The Pre-flight Test function is used to verify autopilot control and monitor functions to reduce the possibility of having dormant failures in critical system components.
As part of the test sequence, the AFCS moves the linear/trim actuators. The collective actuator is also moved. For the pitch, roll and yaw axes the linear and trim actuators are simultaneously driven in opposite directions. This limits the amount of actual flight control movement that occurs during the pre-flight test. The synoptic messages (MFD) show the status and results of the test to the pilot.

The CPL function is activated when the APs are engaged in ATT mode and a FD mode is engaged. The CPL push-button can toggle between coupled and uncoupled.



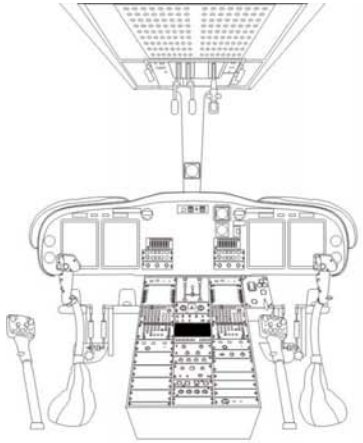


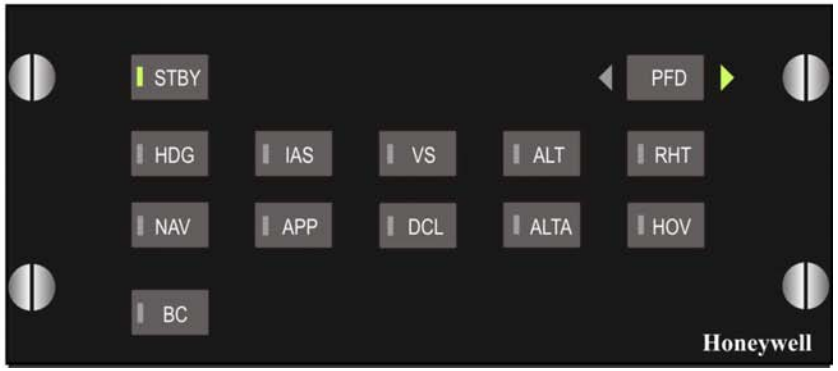
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AFCS CONTROLS – AUTOPILOT CONTROLLER

GUIDANCE CONTROLLER

The GC located on the control pedestal of the cockpit is used to control the AFCS flight director functions and PFD coupling. Each push-button on the Guidance Controller has an associated integral green indicator light (led type) that comes on when the associated functions is activate. All push-buttons on the GC have backlighting for Night Vision Goggle (NVG) operation.
Power for the indicators on the GC is 28 Vdc for daytime operation and 14 Vdc for night-time (dim) operation. A dimming circuit remotely controls the dimming of the status lights.
Each button contains two independent channels that supply communication to an MAU each.
The Guidance Controller engages and disengages the mode(s). The mode(s) is activated by pressing the relative push-button and with the illumination of the green annunciator light (led): a visual annunciator on the PFDs indicates that particular FD mode(s) is engaged (armed or captured). The PFD push-button is used to select which PFD (left or right) source information is used as a data source for both FDs. The PFD source selection is indicated by green left/right arrow annunciators located on each side of the PFD push-button on the GC and repeated on the PFDs.





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AFCS CONTROLS – GUIDANCE CONTROLLER

LINEAR ACTUATOR

Three sets of dual Linear Actuators provide limited control inputs to pitch, roll and yaw axis flight control lines in series with pilot input (see chapter 67 00).

Each set is connected to the relevant axis flight control line through a dual-action bellcrank that permits summing of actuator inputs to pilot inputs as well as preventing a Linear Actuator mechanical failure from losing pilot manual control on that axis.

Each dual Linear Actuator set incorporates two identical and independent electrical motors, one controlled by AP 1 and one controlled by AP 2 via dedicated digital buses (CAN Bus).

They are also called “smart” linear actuators since they include a microprocessor to internally close the servo loop on the position command from the on-side MAU.

Each brushless motor drives a ball-screw which displaces the flight control line

The linear actuator is a high frequency, limited authority control of the pitch, roll and yaw axis. The linear actuator consists of a digital bus command interface, a brushless motor, a balls crew, a dc engaged brake and an independent Linear Variable Differential Transformer (LVDT) position sensor. This is used as a feed back for the linear actuator movement.

The 3 Vac, 2048 Hz reference is an external excitation from the AI/OP-A1 and A2.

The speed of travel depends on the single or dual actuator operation.

The actuator contains a microprocessor that closes a servo loop on a position command from the MAUs. The position command to the linear actuator is via the CAN bus.

Each linear actuator includes an integrity centering function to protect against runaway failure modes.

During normal operation with both Autopilots engaged, each AP outputs 50% of the computed input for an axis.

In case of single AP operation, 100% of the computed input is provided to the on-side Linear Actuator: in this case the total authority of the control is reduced to a half, resulting in a degradation of the system performance

An easy-on/easy-off function helps to prevent undemanded transients in the controls when the AP is engaged or disengaged. This function causes the pitch, roll and yaw linear actuator to move carefully back to the centre. After the AP is disengaged and the actuator is at the centre, the 28Vdc brake signal is removed, which causes the brake to lock the actuator at the centre.

The Linear Actuators are identified and powered as follows:

- Pitch Actuator #1 (HP2) and Roll Actuator #1 (HP1) powered by MAIN BUS1 via AFCS ACT CPLT circuit breaker (CB55) and fuse F3.
- Pitch Actuator #2 (HP15) and Roll Actuator #2 (HP14) powered by ESS BUS2 via AFCS PLT circuit breaker (CB54) and fuse F4.
- Yaw Actuator #1 (HP3) powered by MAIN BUS1 via AFCS ACT CPLT circuit breaker (CB55) and fuse F5.
- Yaw Actuator #2 (HP16) powered by ESS BUS2 via AFCS ACT PLT circuit breaker (CB54) and fuse F6.

The linear actuator stroke and authority are assigned as follows:

• Pitch axis	±5.5 mm	±5% each act.	±10% both act.
• Roll axis	±5.0 mm	±5% each act.	±10% both act.

- Yaw axis ±8.0 mm ±5% each act. ±10% both act.

The linear actuator threshold for the Trim Activation is as follows:

- Pitch axis:
  - Single channel: 10% actuator deviation must exist constantly for longer than 0.025sec.
  - Dual channel: 10% actuator deviation must exist constantly for longer than 1.0 sec.
- Roll axis:
  - Single channel: 18% actuator deviation must exist constantly for longer than 0.4 sec.
  - Dual channel: 18% actuator deviation must exist constantly for longer than 0.025 sec.
- Yaw axis:
  - Single channel: 18% actuator deviation must exist constantly for longer than 0.4 sec.
  - Dual channel: 18% actuator deviation must exist constantly for longer than 0.025 sec.

Four rotary Trim Actuators provide full control of pitch, roll, yaw and collective axis for automatic flight.

The Trim Actuators are connected in parallel with pilot flight control input (cyclic stick, collective lever and pedals) and have the full authority on the flight control lines (see chapter 67 00).

The purpose of the rotary actuators is to maintain roll, pitch, and yaw linear actuators at the centre of travel position to ensure full actuator authority in response to aircraft control commands.

Each Trim Actuator incorporates:

- A motor with non-reversible gears (Trim Motor)
- A magnetic clutch (Force Trim)
- An artificial feel device with detent microswitch
- A dual position sensor

TRIM MOTOR

The four Trim Motors are controlled by one Autopilot system at a time: the elected AP is the first that is engaged and is named “TRIM MASTER”.
The “Autotrim” function of the Trim Master AP directly controls the Trim Motors using a position servo loop that is closed within the MAU. Position feedback is given to the MAU by a position sensor that is integrated with the rotary actuator.
The trim motors are controlled by PWM commands. This signal operates adjustable speed dc motors with PWM between 28Vdc/ground condition. The two wire open/ground connection changes polarity for bi-directional control.
Trim actuators are powered by MAIN BUS 2 via the AFCS AUTO TRIM circuit breaker (CB61) and the AIOP Channel A module of the Trim Master AP.

FORCE TRIM

The Force Trim clutch permits locking of the associated flight control line for hands-off flying.
The Force Trim is normally engaged, ie. with no electrical power the clutch is closed.
The Force Trim can be released (open clutch for hands-on flying) by either:

- Setting the FORCE TRIM switch (pitch and roll) or the CLTV/YAW TRIM switch (collective and yaw) on the MISC panel to OFF, or
- Pressing the FTR switch on the cyclic stick (pitch and roll), on the collective lever (collective) or on the pedals (yaw)

The Force Trim release power is supplied by ESS BUS1 via the FORCE TRIM circuit breaker (CB64).

ARTIFICIAL FEEL DEVICE

A dual-action spring permits force-feel hands-on flying while the Force Trim is engaged: pilot moving the flight controls without releasing the Force Trim results in full hands-on controllability with control force feedback to the pilot.
As pilot releases the control to resume hands-off flying, the spring returns the relevant flight control line to its neutral position against the Force Trim.
Whenever the spring is out of its neutral position, a detent microswitch forces both Autopilots to suspend ATT mode operations: SAS function remains active.

DUAL POSITION SENSOR

The dual position sensor provides both AP computers with a feedback for closed-loop control.
Pitch, roll and yaw position sensors interface with AIOP-B1 and AIOP-B2 modules; collective trim position sensor interfaces with AIOP-A1 and AIOP-A2 modules.
Each sensor consists of two resolvers that receive a 3 Vac 2048 Hz excitation from the relevant AIOP module.

TRIM RELAYS

Eight relays are used to isolate the Autopilot not selected as Trim Master from the Trim Actuators.

- Relays K65 and K66 are connected to the Pitch Trim Actuator.
- Relays K67 and K68 are connected to the Roll Trim Actuator.
- Relays K69 and K70 are connected to the Yaw Trim Actuator.
- Relays K71 and K72 are connected to the Collective Actuator.

All relays are installed under the baggage compartment floor (helicopter 1Z) or under copilot seat floor (helicopter 2A).

FORCE TRIM OPERATION

When the cyclic FORCE TRIM switch on the MISC panel is set to OFF, the ATT mode reverts automatically to SAS.
Returning the FORCE TRIM switch to ON, the ATT mode is automatically re-engaged.
In a hands-off flight, as long as the pilot holds an FTR switch pressed, on the relevant axis:

- The Force Trim clutch is disengaged
- Pilot is temporarily flying hands-on SAS
- The Linear Actuators are centered
- If a FD mode is engaged, the AP ignores the FD commands

When pilot releases the FTR switch, on the relevant axis:

- The Force Trim clutch is re-engaged
- The ATT mode reference value is reset to the present attitude
- If a FD mode is engaged, the reference target parameter is reset to the present value (eg. IAS, VS, Radio Height, etc)
- Pilot returns to hands-off flight

When the Force Trim clutch is disengaged then:

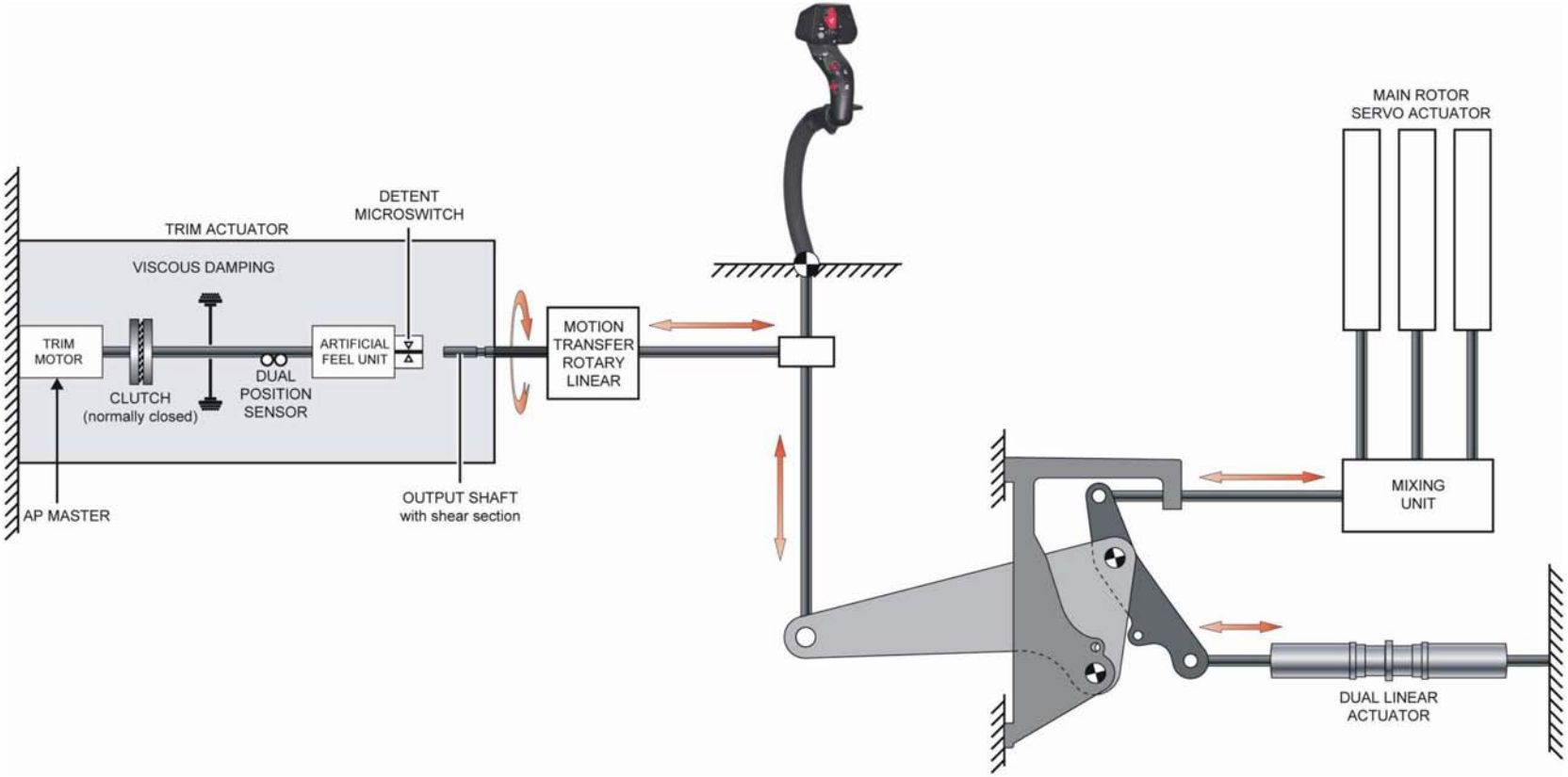
- The force feel system is disengaged
- The Trim Actuator drive is disengaged
- The Autotrim is disabled

Upon detection of the detent switch activation (Pilot moving controls without disengaging the Force Trim), the AFCS disables commands to the respective trim motors.



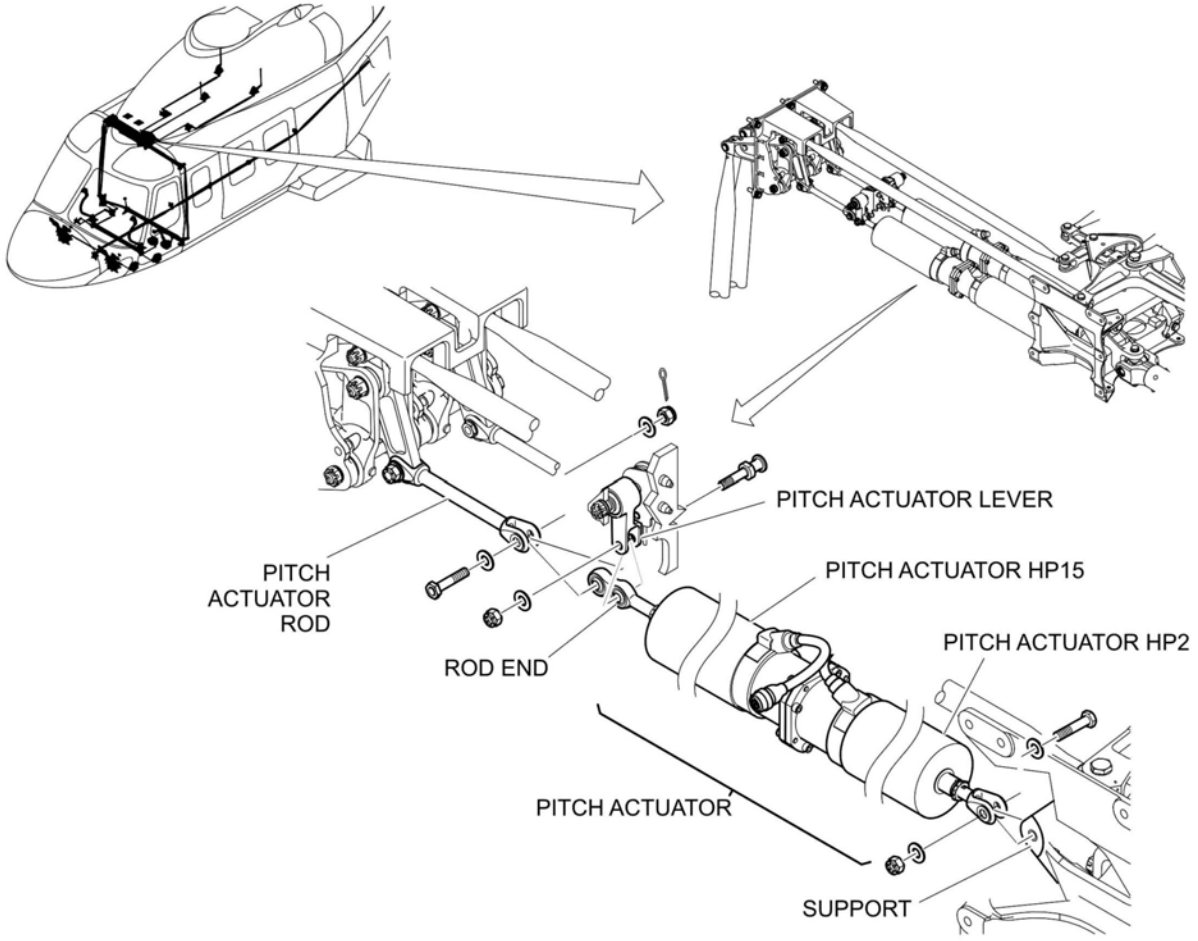
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AFCS – MAIN COMPONENTS



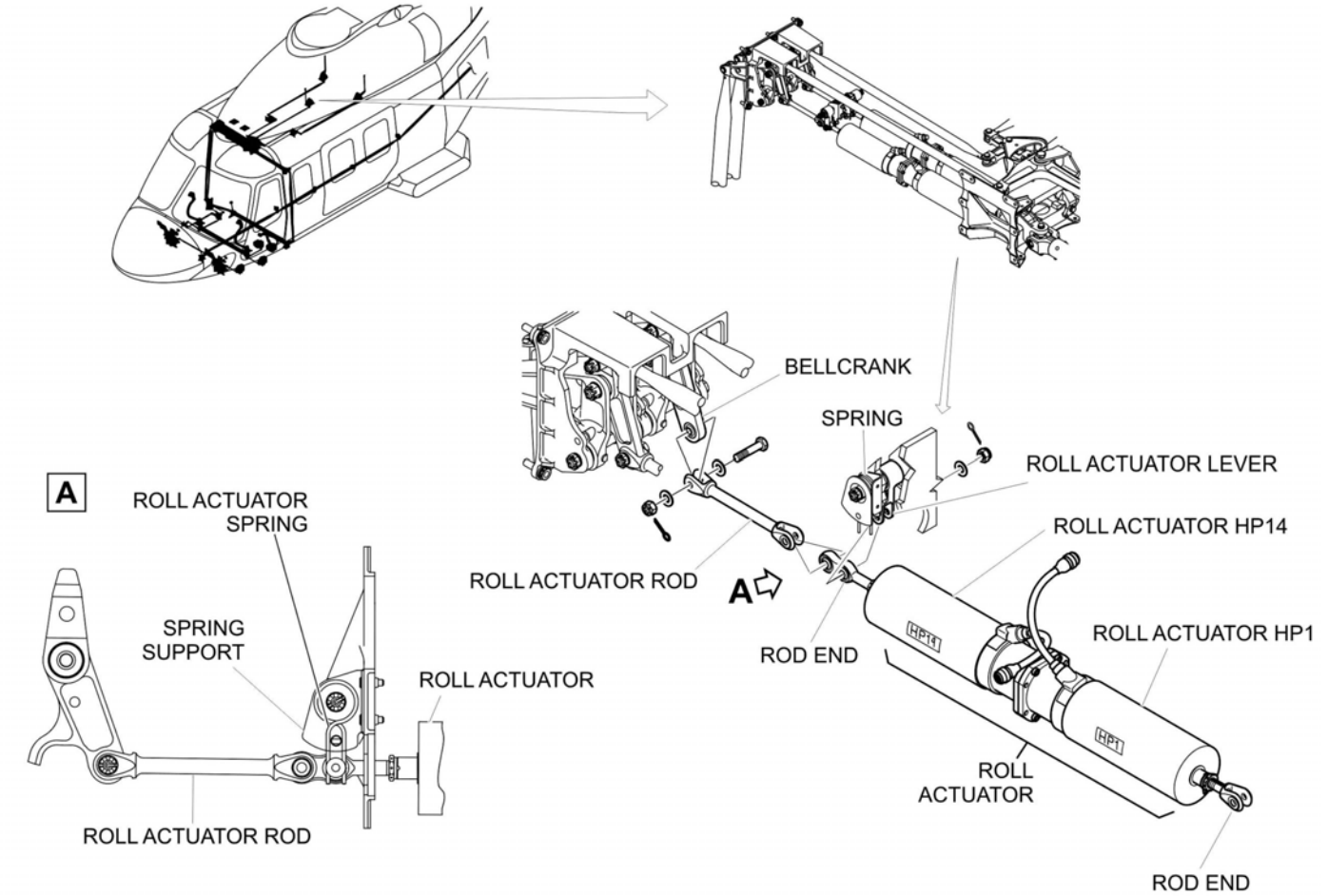
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TYPICAL FLIGHT CONTROL AXIS SCHEMATIC (PITCH OR ROLL OR YAW)



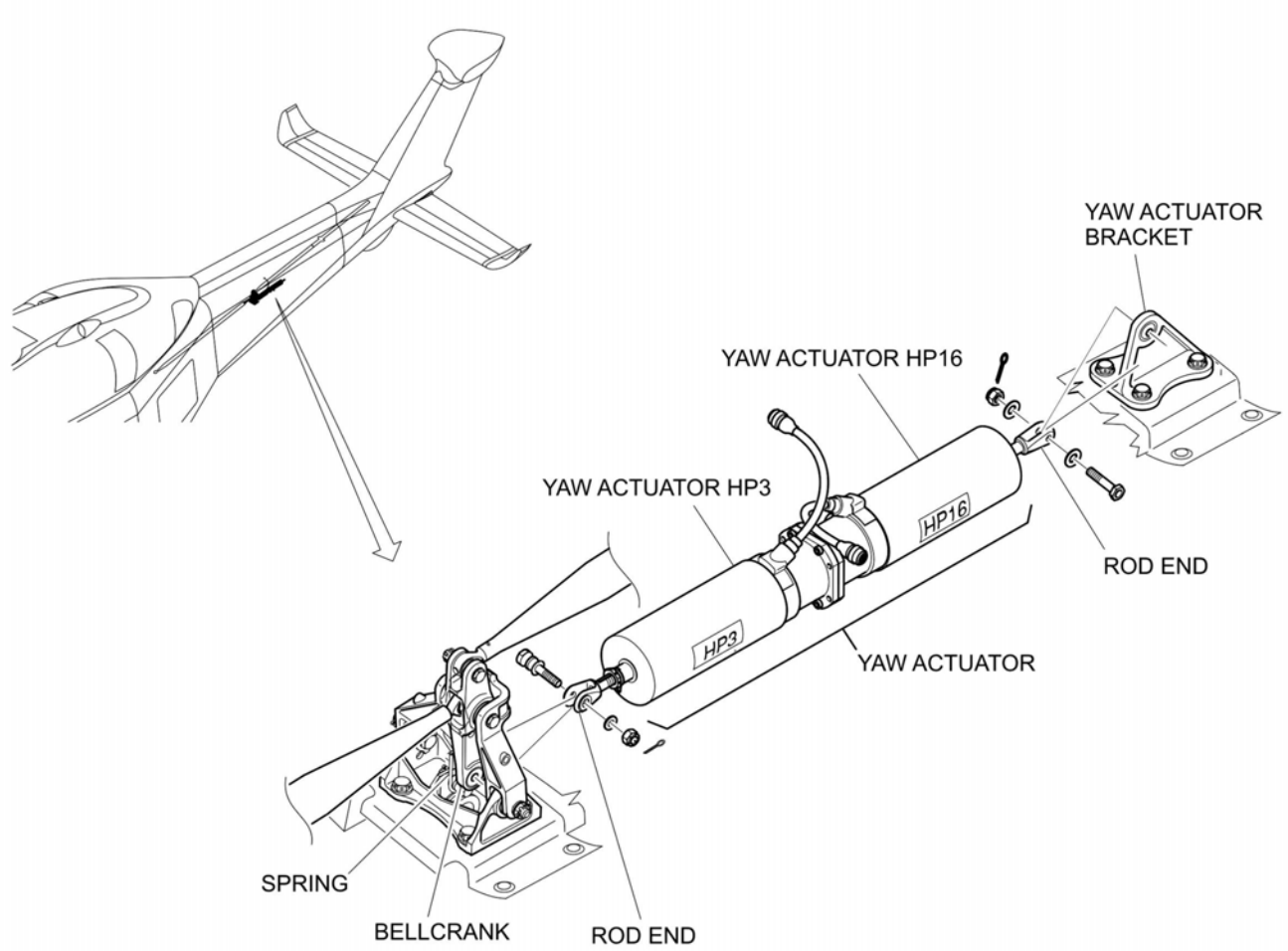
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PITCH LINEAR ACTUATOR



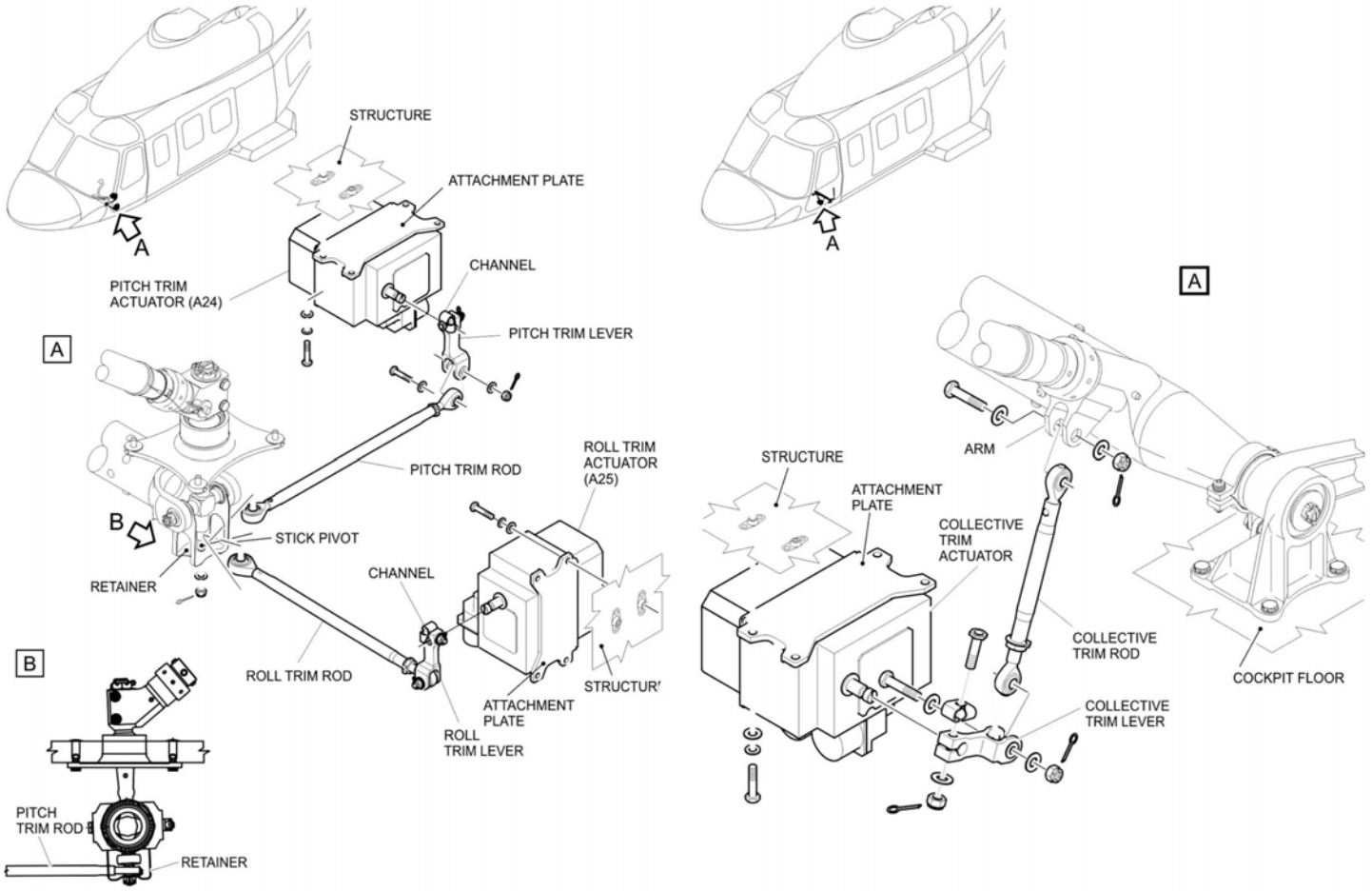
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ROLL LINEAR ACTUATOR



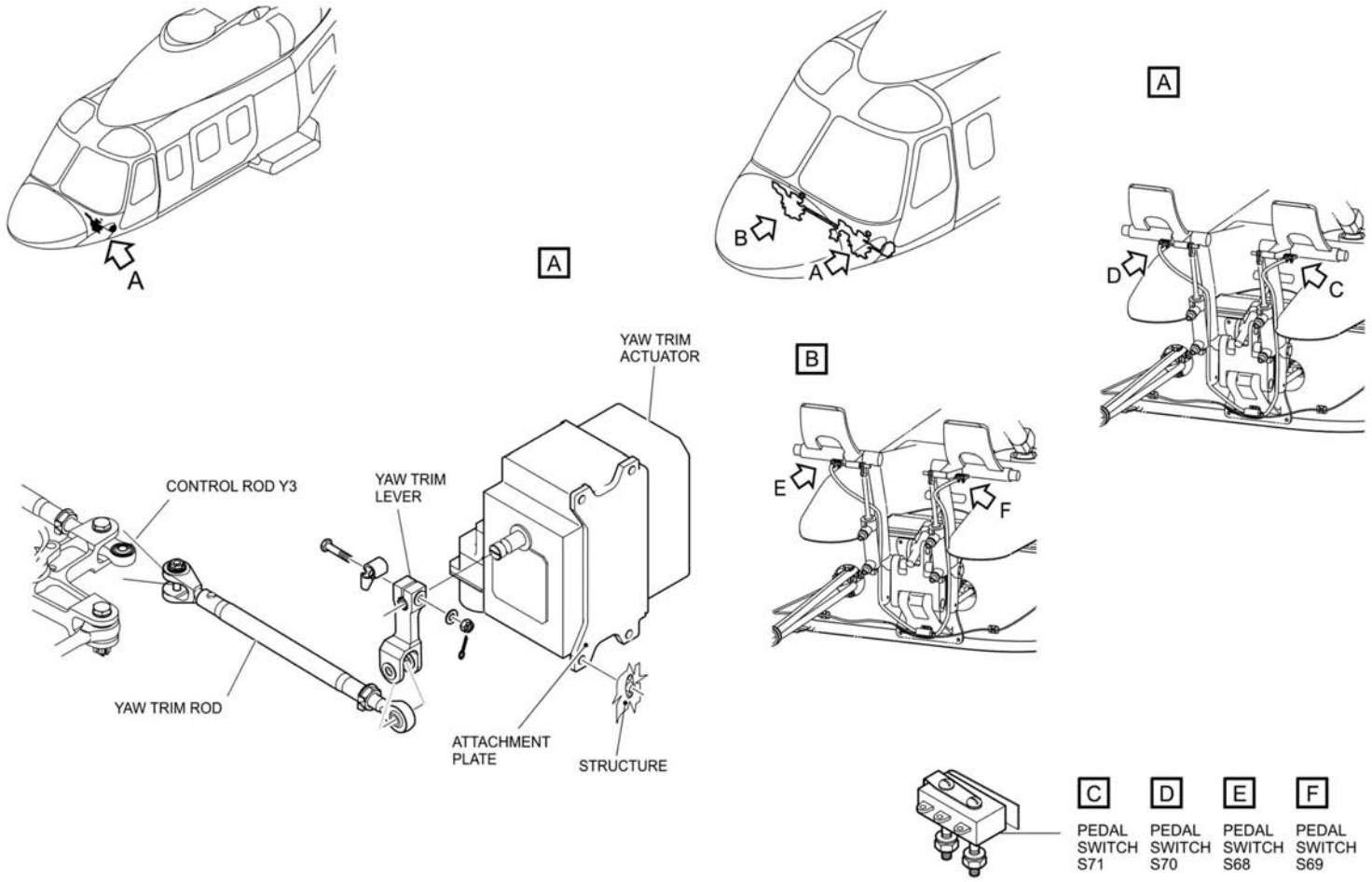
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YAW LINEAR ACTUATOR



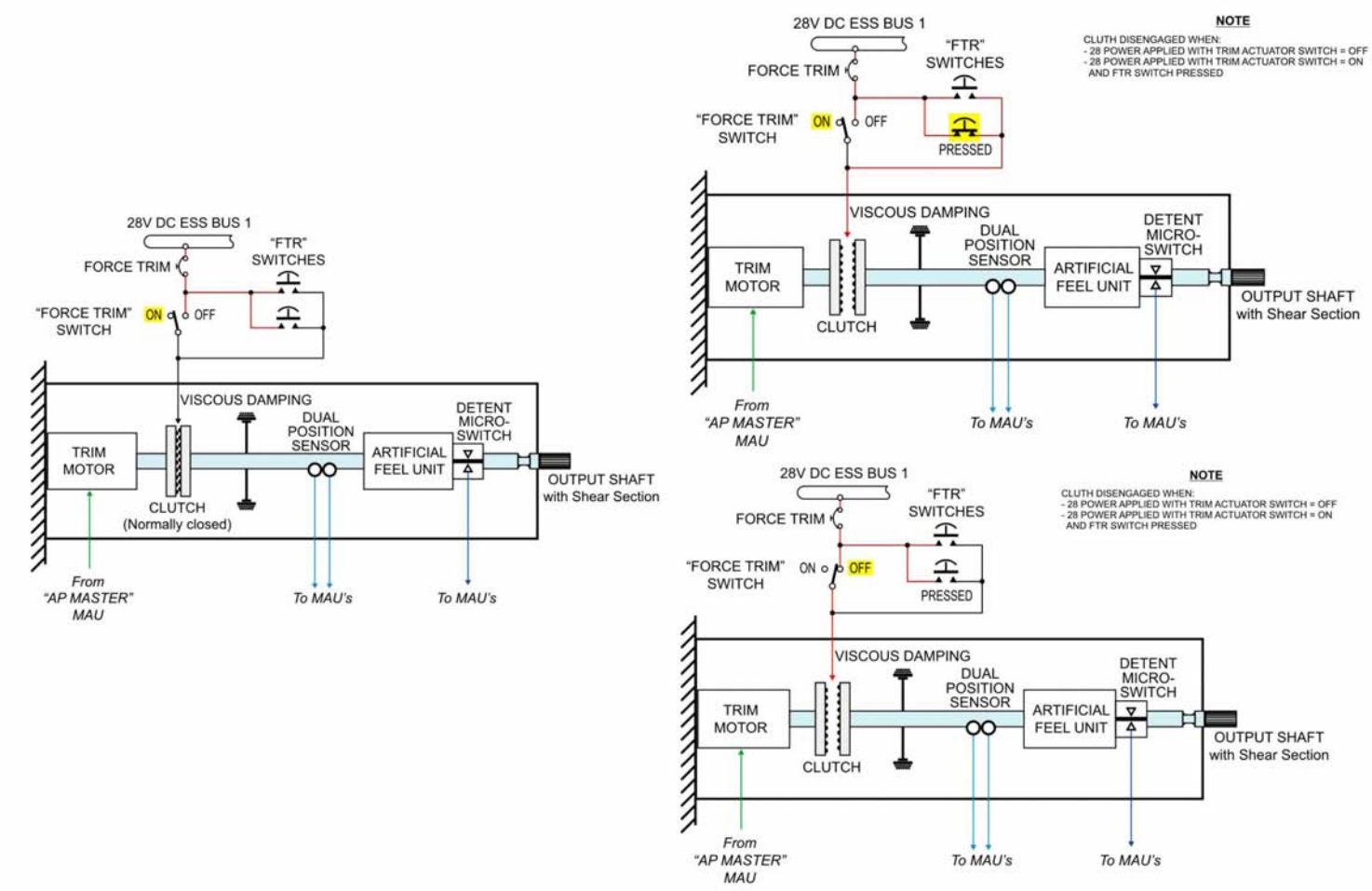
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PITCH ROLL AND COLLECTIVE TRIM ACTUATORS



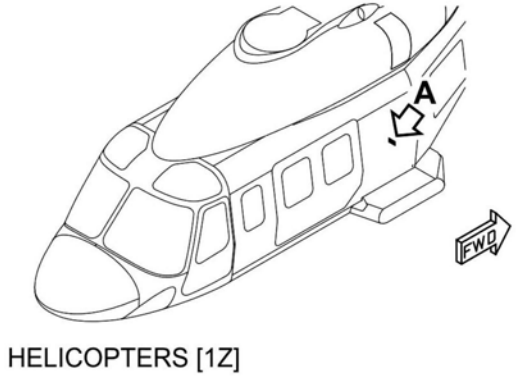
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YAW TRIM ACTUATOR

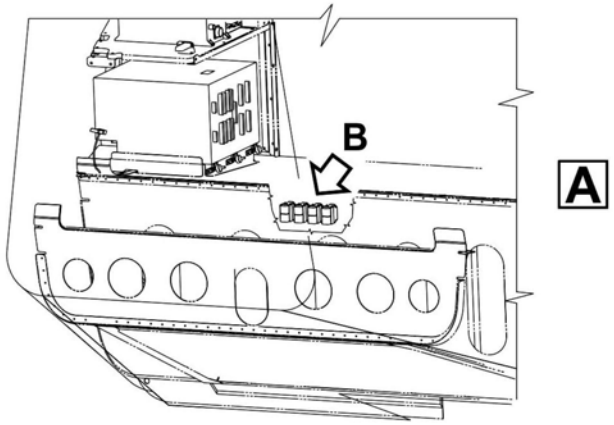


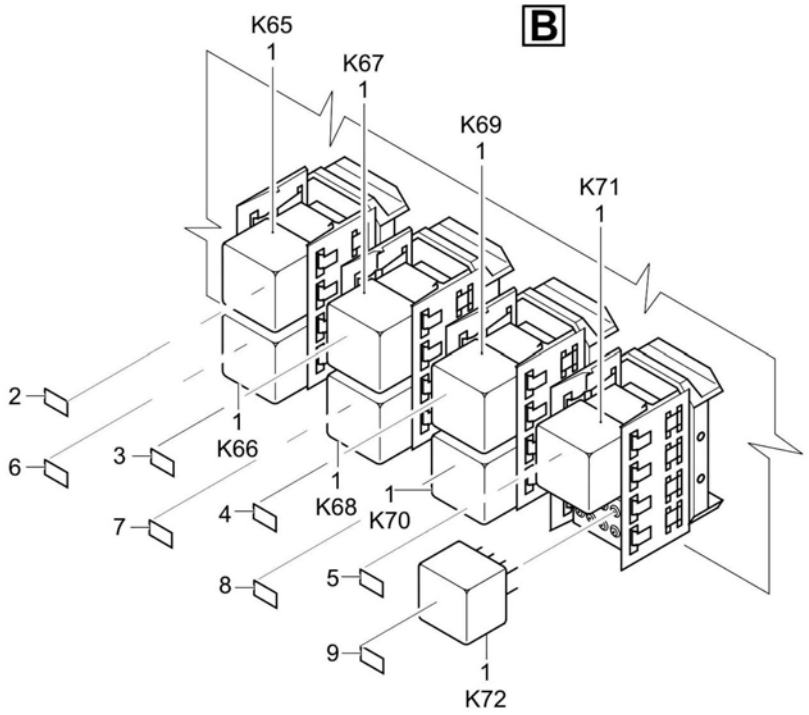
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TRIM ACTUATOR SCHEMATIC AND FORCE TRIM OPERATION



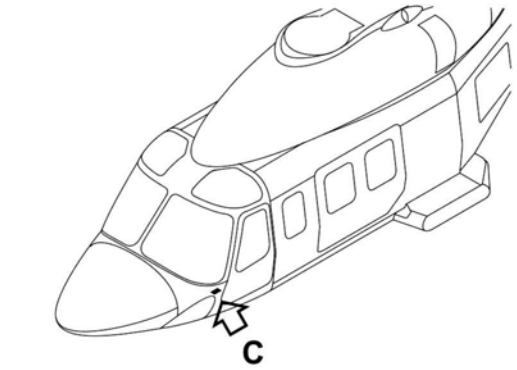
HELICOPTERS [1Z]



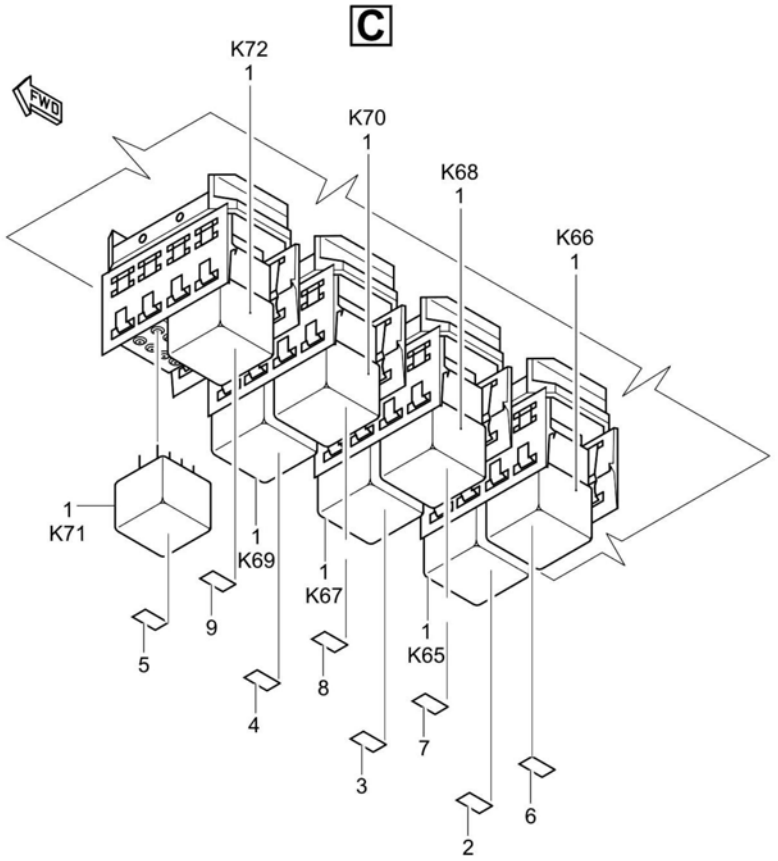


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SHORT NOSE – TRIM RELAYS



HELICOPTERS [2A]



ICN-39-A-220000-G-A0126-00113-A-01-1

LONG NOSE - TRIM RELAYS

AFCS FUNCTIONS

The AFCS provides the following functions that are available also in case of single AP or single FD operation:

- Pre-flight test (TEST)
- Hands-off Attitude Hold (ATT)
- Hands-on Stability Augmentation System (SAS)
- Yaw Control
- Collective Control
- Autotrim
- Trim Priority Function (Trim Master)
- Flight Director (FD) commands and coupling for automatic flight (including Torque Limiting)
- System and sensors monitoring

ATTITUDE HOLD (ATT) MODE

The ATT mode is automatically selected when an Autopilot is engaged (default at engagement).
It provides long-term pitch and roll attitude retention for hands-off flying and for Flight Director coupling in addition to providing stabilization (SAS function).
In the ATT mode the AP pitch and roll Force Trims must be engaged (FORCE TRIM switch on MISC panel at ON).
The pitch and roll attitude to be retained is memorized as that at the time of engagement.
Changes to the selected attitude can be made by:

- pressing the BEEP TRIM switch on the cyclic stick (typically for small changes), or
- pressing the Force Trim Release (FTR) switch on the cyclic stick and flying manually the helicopter to attain the desired attitude, then releasing the FTR to return to hands-off.

Pitch attitude beep rate is ±2°/s for airspeeds lower than 120 KIAS and ±1°/s for airspeeds higher than 140 KIAS.
With the airspeed between 120 and 140 KIAS, the pitch changes linearly between ±2°/s and ±1°/s.
Roll attitude beep rate is ±3°/s regardless of airspeed.
The pilot can override the AFCS ATT mode at any time by taking over the controls manually: if the FTR switch is not pressed, the detent switch inside the Trim Actuator permits “flying through the AP” without changing the selected attitude. When the helicopter is returned to hands-off the ATT mode brings the helicopter back to the memorized attitude.
In ATT mode, the AP pitch and roll channels can be coupled to the Flight Director (FD): in that case the selected attitude is changed by the FD commands.
When the aircraft is flying at high speed (IAS > 45 KIAS) with no FD roll mode engaged, the ATT mode holds roll attitude only if it is selected to be greater than 3°.
If the selected roll attitude is less than 3°, AP forces the roll attitude to 0° (wings level) and holds current aircraft heading through the yaw control channel (High Speed Wings-Level Yaw Heading Hold function).

STABILITY AUGMENTATION SYSTEM (SAS) MODE

The SAS improves the handling characteristics of the helicopter by damping the effects of the short-term external aircraft disturbances on pitch, roll and yaw axes and improves the controllability during low-speed manoeuvring or hovering flight.
The SAS function is active whenever the AP is engaged, either in ATT or in SAS mode.
SAS mode is intended for use where extensive aircraft manoeuvring is required and the pilot prefers to be hands-on without attitude retention.

SAS mode is selected by:

- pushing the SAS button on the auto-pilot controller, or
- setting the FORCE TRIM switch on Miscellaneous panel to OFF.

When SAS mode is selected, being it a hands-on control mode:

- the Autotrim is disabled
- the AFCS can be operated with FORCE TRIM switch either ON or OFF
- the ATT OFF caution message is displayed in the CAS window and the SAS annunciator is displayed in the ADI (PFD)

NOTE B

Each autopilot uses the on-side AHRS for both ATT and SAS mode computations; failure of an AHRS causes the disengagement of the on-side Autopilot.

YAW CONTROL

Autopilot Yaw Control channel is engaged automatically when an AP is selected and remains active in both ATT and SAS modes.

It provides airspeed-switched yaw coordination as required to maintain proper high and low speed directional control.

A hysteresis separates the low speed and high speed operating ranges: high speed operation is entered accelerating above 45 KIAS, low speed operation is entered decelerating below 41 KIAS.

The yaw control provides the following functions:

- At any speed:
  - Yaw Rate Damping
  - Collective-to-Yaw Cross-feed
- At low airspeed (IAS < 41 KIAS):
  - Low Speed Yaw Heading Hold
- At high airspeed (IAS > 45 KIAS):
  - High Speed Turn Coordination
  - Lateral Ball Trim
  - High Speed Wings-Level Yaw Heading Hold

YAW RATE DAMPING

Yaw Rate Damping (SAS Yaw channel) is active whenever the Autopilot is engaged; it provides stability augmentation about the yaw axis at all airspeeds and helps preventing the dutch roll tendency of the helicopter.

COLLECTIVE-TO-YAW CROSS-FEED

Collective-to-Yaw Cross-feed function is active whenever the Autopilot is engaged; it automatically compensates any collective position changes to counteract the effect of torque when the collective control is moved.

LOW SPEED YAW HEADING HOLD

Low Speed Yaw Heading Hold function automatically engages at low speed (IAS < 41 KIAS) if roll attitude is less than 3°, or when HOV mode is engaged on FD.
Yaw force trim must be engaged (CLTV/YAW TRIM switch on the Miscellaneous panel is at ON) for the function to operate.
Once the Low Speed Yaw Heading Hold function is active, it does not disengage in case roll attitude exceeds 3°.
The low speed yaw heading hold reference is established at the time this function becomes active and is indicated with a small magenta low speed heading bug displayed on the HSI.

Changes can be performed by repositioning the aircraft acting on the pedals or using the YAW beep switch on the collective grips (±3°/s).

HIGH SPEED TURN COORDINATION

High Speed Turn Coordination function operates at high speed only (IAS > 45 KIAS) during roll controlled turns to minimize slip or skid. A yaw rate command supplies this function from the calculated aircraft turn rate as a function of the bank angle and airspeed.

LATERAL BALL TRIM

Lateral Ball Trim function operates at high speed only (IAS > 45 KIAS); it allows the pilot to keep a constant slip or skid, either to offset a small accelerometer misalignment or to purposely offset the tail alignment with respect to the aircraft horizontal flight path.
Yaw force trim must be engaged (CLTV/YAW TRIM switch on the Miscellaneous panel is at ON) for the function to operate.
The magnitude of the slip or skid is set using the YAW beep switch on the collective grips (±0.5 ft/s²/s).

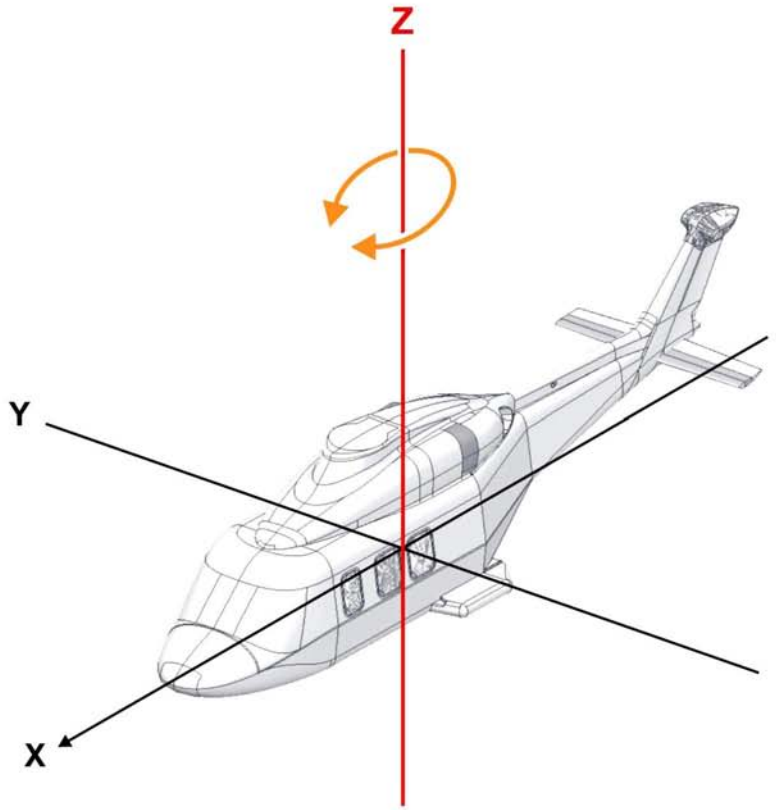
HIGH SPEED WINGS-LEVEL YAW HEADING HOLD

The High Speed Wings-Level Yaw Heading Hold function is active when the aircraft is flying at high speed (IAS > 45 KIAS) with no FD roll mode engaged and the selected roll attitude is less than 3°: the function automatically reduces the roll attitude to 0° (wings level) and then holds new aircraft heading through the yaw channel.
Yaw channel is also used to correct for heading errors when HDG mode is engaged on FD (coupled), if the roll attitude is less than 0.5° and heading error is less than 2°.
The roll attitude is commanded to wings level while yaw is driving heading.

AT ANY SPEED
• Yaw Rate Damping (Stabilization) • Collective-to-Yaw Cross-feed

AT LOW SPEED (IAS < 45 KIAS)
• Low-Speed Yaw Heading Hold

AT HIGH SPEED (IAS > 45 KIAS)
• Turn Coordination • Lateral Ball Trim • High Speed Wings-Level Yaw Heading Hold



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YAW CONTROL FUNCTIONS

COLLECTIVE CONTROL

The Collective Control function is activated by selecting 3-cue FD operation or by activating a collective only mode (RHT) while the CLTV/YAW TRIM switch on the Miscellaneous panel is at ON.
The Collective Control function provides automatic vertical control of the helicopter by means of a rotary Collective Actuator connected in parallel with the collective mechanical control system.
When the CLTV/YAW TRIM switch is turned OFF, the collective (and yaw) Force Trims are disengaged.
Changes to collective reference settings can be performed by repositioning the collective lever with the collective FTR pressed or using the CLTV beep switch on the collective grips.
The Collective Control function is disabled by deselecting the mode, uncoupling the FD from the AP or selecting the SAS mode.
The AP that is controlling the collective actuator is the Trim Master.
The Collective Control function limits collective position to prevent engine or mast torque exceedances. This limiting function will only be active when the automatic collective control function is engaged.

AUTOTRIM

The purpose of the automatic trim (AUTOTRIM) function is to keep the average position of the output shaft of the roll, pitch and yaw linear actuators near center to ensure optimum Linear Actuator authority in response to aircraft control commands; the AP Autotrim function commands the Trim Actuators to achieve that.
The Autotrim function on each axis is active when either or both Autopilots are engaged in ATT mode and the relevant axis Force Trim is engaged.
The Autotrim control reference is the Linear Actuator displacement from the center: when the displacement exceeds a percentage of the Linear Actuator stroke, the Trim Motor is commanded at an appropriate rate in the direction to re-center the Linear Actuator (different percentage values are set for pitch, roll and yaw axes).

TRIM PRIORITY FUNCTION (TRIM MASTER)

The Trim Priority Function has the purpose to provide and retain the Trim Master operation on controlling the Trim and Collective actuators.
For normal operation, when both autopilots are engaged, the autopilot that was engaged first retains the Trim Master function.
The indication of what AP is elected as Trim Master is shown in the Flight Controls synoptic page of the MFD.
As the Trim Master AP is disengaged (either manually or automatically due to a failure), the remaining AP is elected Trim Master.
The Trim Master function switching logic is overridden if one or more linear actuators fail.
A set of relays in the aircraft wiring are used to ensure that only the Trim Master AP provides commands to the pitch, roll, yaw and collective trim actuators so that they cannot be corrupted by the other AP.
Similarly, the trim end-around feedback is provided to identify trim and collective actuator movement and direction.

PRE-FLIGHT TEST (TEST)

Pre-flight test allows verifying autopilot control and monitoring functionality.
During the pre-flight procedure, as part of the System Checks, pilot is required to perform the AFCS TEST that is initiated by pressing the TEST button on the Autopilot Controller with the following conditions:

- Aircraft on the ground (WOW)
- Electrical and hydraulic power available
- Autopilots disengaged
- Hands off the flight controls (Force Trims on)

- Collective down (<10%)

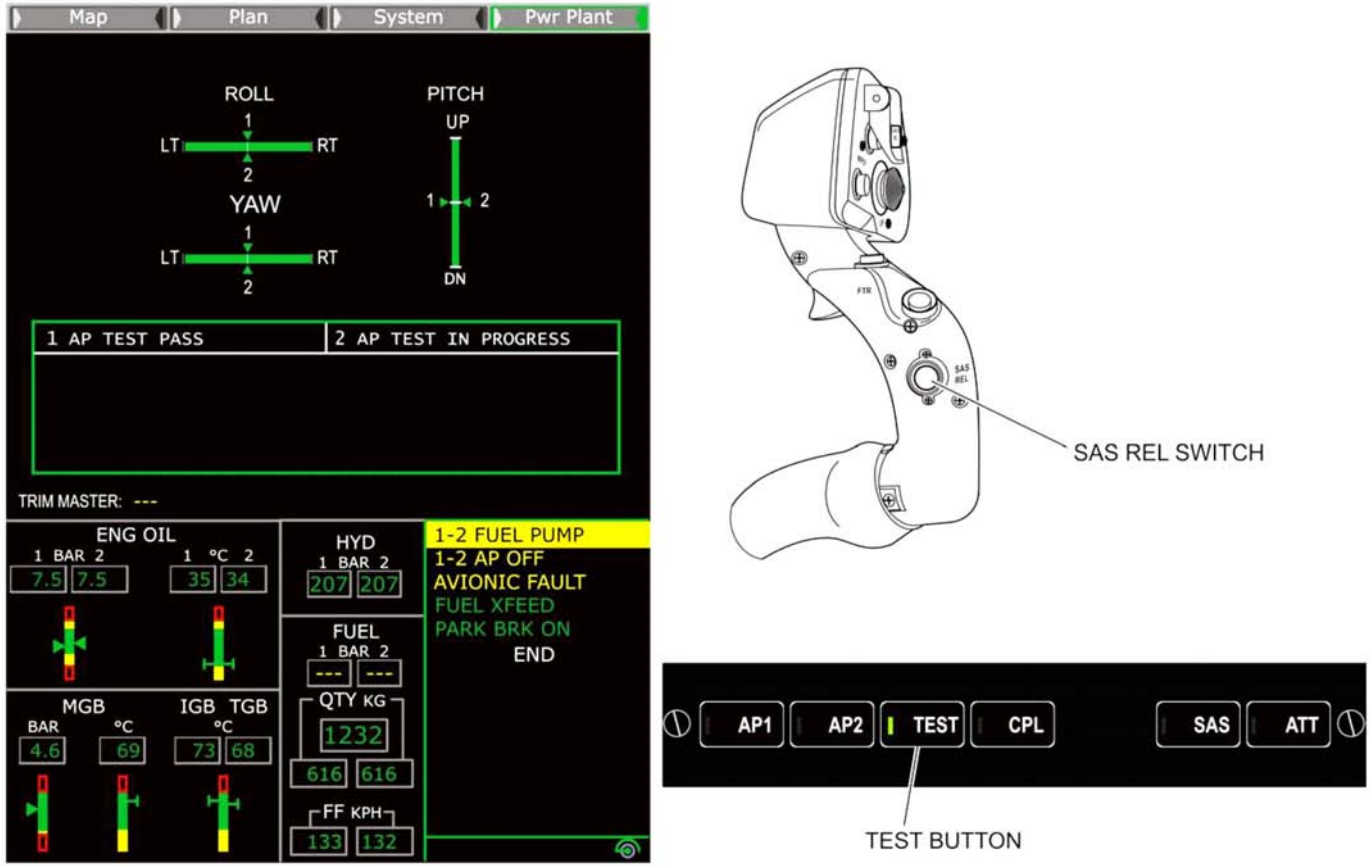
During the self-test, the each AP moves both Linear Actuators and Trim Actuators in opposite directions. This limits the amount of actual flight control movement that occurs during the test.
The pre-flight test verifies autopilot control and monitor functions reducing the possibility of having latent failures in critical system components.
The result of the test is displayed on the Flight Controls synoptic page (that is automatically selected on pilot MFD) and in the CAS window.
Refer to Section 2 of the RFM for the AFCS TEST procedure.

PRE FLIGHT TEST ANNUNCIATORS

While this test is running, the FLIGHT CONTROL synoptic page displays a text box containing one or more messages listed below in case of an improper operativity condition:

Message Wording	Message Conditions
• ACT CPLT POWER FAIL:.....	Copilot's linear actuators are not powered.
• ACT PLT POWER FAIL:.....	Pilot's linear actuators are not powered.
• 1(2) AUTOTRIM POWER FAIL: …	Trim actuators are not powered.
• 1(2) HYDRAULIC PRESSURE INVALID:	hydraulic pressure is not in normal operating range.
• 1(2) COLLECTIVE TOO HIGH:	Collective positive is too high.
• 1(2) TRIM OFF.....	Cyclic trim switch is off.
• 1(2) YAW TRIM OFF.....	Yaw trim switch is off.
• 1(2) COLLECTIVE OFF.....	Collective switch is off.
• 1(2) FTR ACTIVE.....	Cyclic FTR switch is pressed.
• 1(2) YFTR ACTIVE.....	Yaw FTR switch is pressed.
• 1(2) CFTR ACTIVE:.....	Collective FTR switch is pressed.
• 1(2) PITCH OUT OF DETENT: ….	Pitch trim actuator is being held out of detent.
• 1(2) ROLL OUT OF DETENT: …..	Roll trim actuator is being held out of detent.
• 1(2) YAW OUT OF DETENT: ……	Yaw trim actuator is being held out of detent.
• 1(2) COLLECTIVE OUT OF DETENT:	Collective actuator is being held out of detent.
• 1(2) PRESS AND RELEASE SAS REL SWITCH:.....	Pre Flight test request for pilot action.
• 1(2) SAS REL SW FAILURE:……	Pre Flight test did not detect pilot press of the SAS REL switch.
• 1(2) SAS RELEASE SWITCH ACTIVE:	Pre Flight test detected a press of the SAS REL switch when not expected.
• TEST WAITING:.....	Displayed while waiting the pilot to press the SAS REL switch.
• TEST IN PROGRESS.....	Displayed when on-side AFCS is executing the actuator tests.
• TEST COMPLETE:.....	Displayed when all tests have been completed.
• TEST STANDBY:.....	Displayed while waiting for the cross-side AFCS to complete actuator tests.
• TEST INHIBIT:.....	Displayed when test is halted due to an inhibit condition.

- TEST INVALID:..... On-side AFCS is invalid, while cross-side AFCS is running test.



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AFCS – PRE FLIGHT TEST

AUTOPILOT – OPERATION

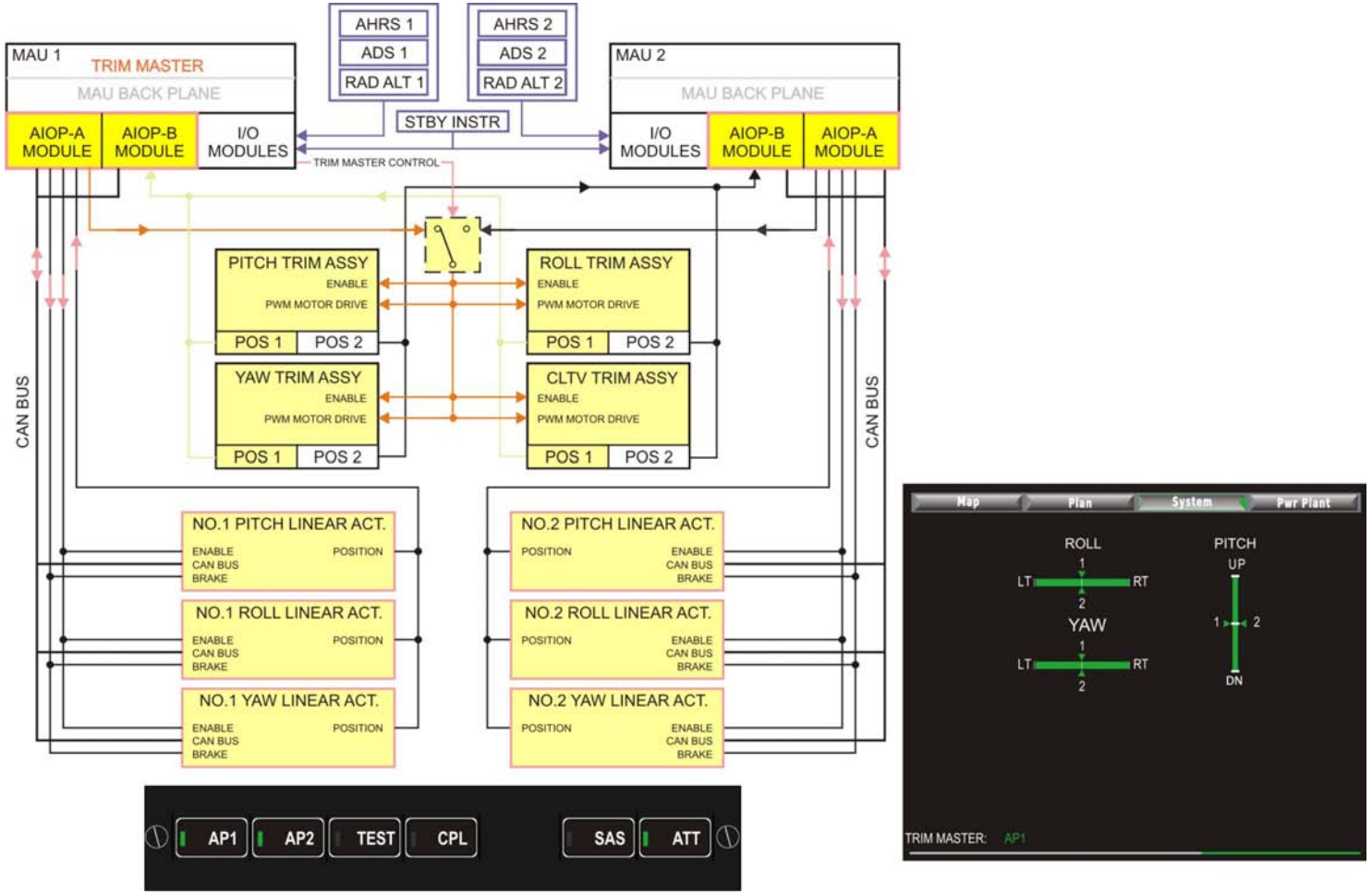
ENGAGING AND DISENGAGING THE AUTOPILOTS

The autopilots engage and disengage by pushing the AP1 and/or AP2 buttons on the autopilot controller.
Simultaneous disengagement of both AP 1 and AP 2 can also be commanded by pressing the SAS REL push-button on either cyclic stick.
ATT is the default mode when the autopilots are engaged. However, the autopilots are forced to SAS mode when the cyclic trim enable switch is OFF.
To prevent undesirable transients in the controls at engagement or disengagement, an easy-on/easy-off function drives the pitch, roll and yaw Linear Actuators gently back to center.
After disengagement and actuator centering, the 28 Volt brake excitation is removed, locking the Linear Actuators in the center position.
The first autopilot engaged is elected Trim Master and takes control of the four Trim Actuators by switching on or off a set of relays.
As the Trim Master AP is disengaged (either manually or automatically due to a failure), the remaining AP is elected Trim Master.

NORMAL OPERATION

In normal operation, each autopilot supplies half the total linear actuator system authority.
Engaging the AP activates the automated yaw control function and the Autotrim function.
The autopilot control authority as supplied by the dual series Linear Actuators is 20% of the full aircraft control authority for each axis (each section of a Linear Actuator providing 10% of the full aircraft control authority).
Flying manually, the pilot has the same control authority over the aircraft when the autopilot is engaged in SAS or ATT modes, or when the autopilot is disengaged.
AP 1 and AP 2 operate either with or without the flight director guidance modes active.

In normal operation, both autopilots are engaged to supply full dual system performance when coupled to the flight director.
The on-side AHRS is the main reference for AP operation; in case an unflagged miscompare occurs between the two AHRS, the electronic Standby Instrument input is used by the MAU to determine which AHRS has failed.
The on-side ADS is the reference for AP gain computations; in case of ADS dual failure or in case an unflagged miscompare occurs between the two ADS, a fixed airspeed value is used by the AP.



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AUTOPILOT OPERATION

AUTOPILOT FAILURE

When detecting failure affecting mode integrity, the AP of the affected single system disengages.
The following events also disengage the on-side AP on the corresponding AHRS:

- Invalid on-side attitude data
- Invalid on-side attitude rate data
- Invalid on-side yaw rate data

Failure of an AP is annunciated by CAS messages and the AUTOPILOT AUTOPILOT aural warning message.

SINGLE AUTOPILOT OPERATION

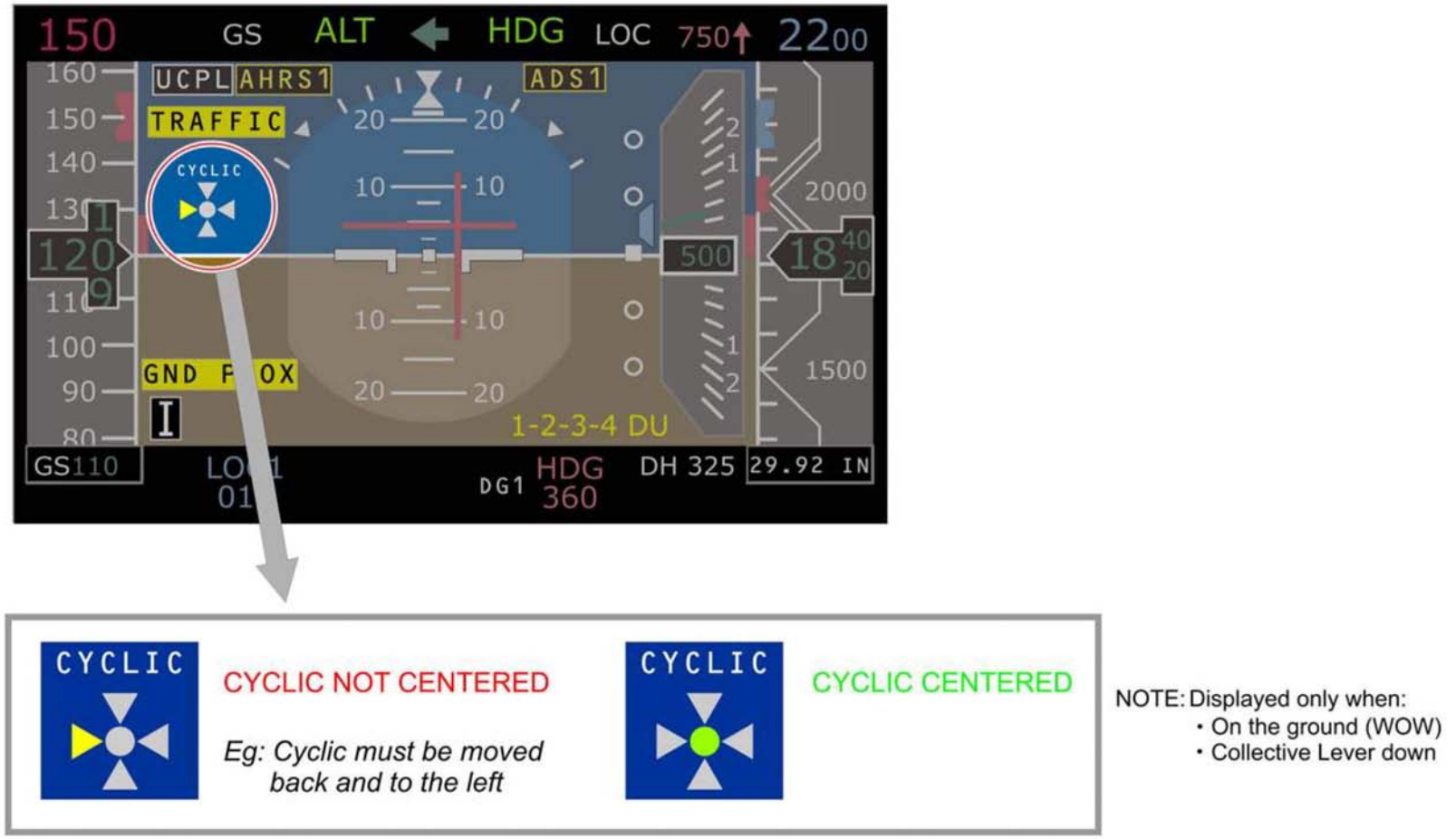
With only one autopilot engaged, the single system operates at full gain with half the dual system authority. When both autopilots are engaged, each single system gain is reduced to 50% so that each system supplies half of the required input. This results in full gain control with twice the single system authority.
Full yaw control and Autotrim functions are available also in single system operation.

AUTOPILOT OVERRIDE

The pilot has full authority with the AP engaged or disengaged and can immediately override the AFCS at any time by simply taking over the controls. With the autopilots engaged, FTR switches, AP disengage switches (SAS REL), Detent switches and actuator circuit breakers offer different ways to partially or completely override the AFCS.

CYCLIC POSITION INDICATOR

The cyclic position indicator is displayed on the ADI to help pilot center the cyclic controls before starting the engines, to ensure that the main rotor does not hit the static stops when rotating at low speed.
The indicator is only displayed when the helicopter is on the ground (WOW) with collective down (LVDT signal via EECs).
Indication is taken from the position sensors located inside the pitch and roll Trim Actuators.
The cyclic is centered when the indicator shows a green dot; amber arrows indicate which direction the cyclic stick must be moved to center the controls.



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CYCLIC POSITION INDICATOR

AUTOPILOT CONTROLS

AUTOPILOT CONTROLLER

1 AP 1 push-button

- PRESSED: ..... Illuminates the green annunciator light and engages the AP no.1.
  Pushing the button again extinguishes the green annunciator light and disengages the AP no.1.

NOTE C

When the pilot engages AP 1 then:

a. ATTITUDE mode is set as default mode

b. the yaw control functions are activated

c. the SAS mode is forced if the cyclic FORCE TRIM switch is set to OFF.

The AP can operate both with and without FD guidance. In normal operation, both AP 1 and AP 2 are engaged in order to supply full dual system performance while coupled to FD.

2 AP 2 push-button

- PRESSED: ..... Illuminates the green annunciator light and engages the AP no.2.
  Pushing the button again extinguishes the green annunciator light and disengages the AP no.2.

NOTE D

See NOTE for AP 1.

3 TEST push-button

- PRESSED: ..... Illuminates the green annunciator light and starts the Built-In Test (BIT).
  Pushing the button again extinguishes the green annunciator light and exits the BIT.

4 CPL push-button

- PRESSED: ..... When green annunciator is illuminated, uncouples the FD from AP and the annunciator extinguishes.

NOTE E

When a FD mode is engaged, FD automatically couples to AP and the CPL green annunciator illuminates.
Pushing the CPL button again illuminates the green annunciator light and manually re-couples the FD to the AP.

5 SAS push-button

- PRESSED: ..... Illuminates the green annunciator light and engages the SAS mode.

NOTE F

The SAS and ATT buttons are mutually exclusive and are used to select the SAS or ATTITUDE mode of operation of the AFCS. When the cyclic FORCE TRIM switch is set to OFF, the AP engages with the SAS mode active.

6 ATT push-button

- PRESSED: ..... Illuminates the green annunciator light and engages the ATTITUDE mode.

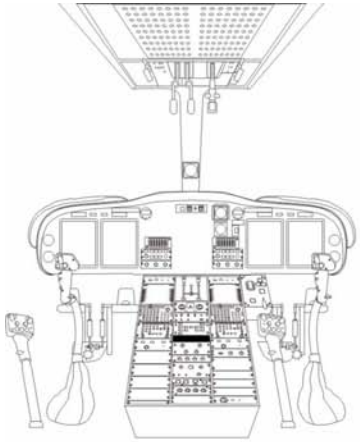
NOTE G

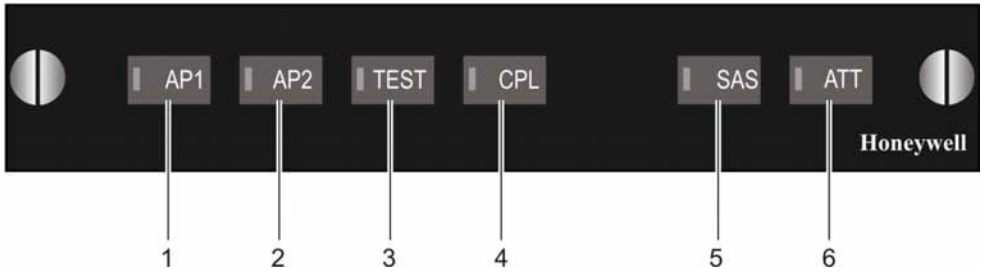
The SAS and ATT buttons are mutually exclusive and are used to select the SAS or ATTITUDE mode of operation of the AFCS.
The ATT mode is automatically engaged if at least one AP is engaged and the cyclic FORCE TRIM switch is set to ON.
The ATT mode disengages if the pilot:

a. engages the SAS mode or

b. sets the cyclic FORCE TRIM switch to OFF or

c. disengages both AP 1 and AP 2.





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AUTOPILOT CONTROLLER

GUIDANCE CONTROLLER

1 STBY pushbutton

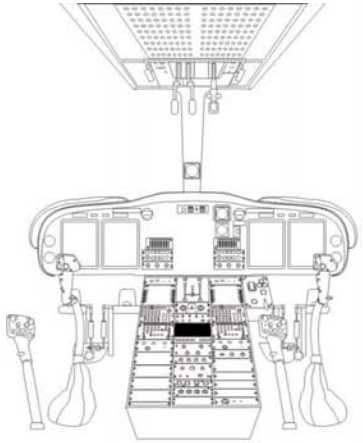
- PRESSED: ..... Illuminates the green annunciator light and cancels any selected active flight director modes.

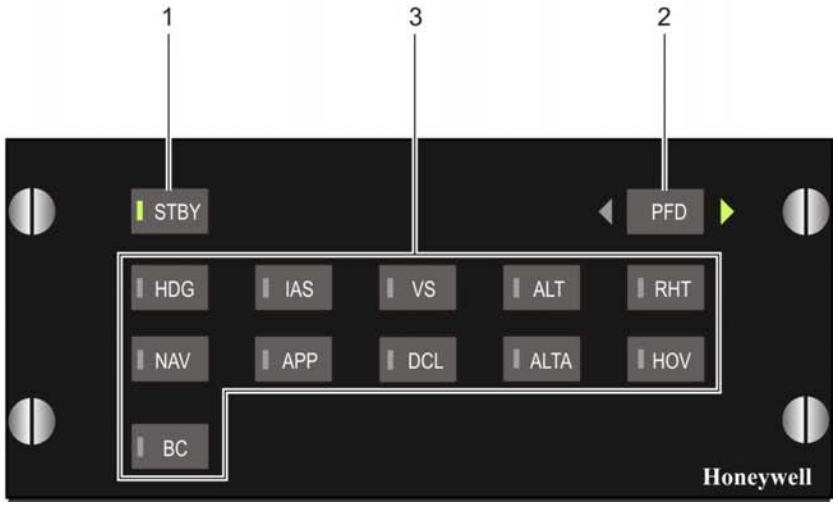
2 PFD pushbutton

- PRESSED: ..... Selects which PFD (left or right) supplies source data that is used by both flight directors and toggles the associated green arrow annunciator located on each side of the PFD button. The illuminated arrow indicates the selected PFD.

3 FD Mode pushbuttons

- PRESSED: ..... Illuminate the relevant green annunciator light and engage or arm the relevant Flight Director mode.
  Pushing any button again extinguishes the relevant green annunciator light and disengages the relevant Flight Director mode.





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GUIDANCE CONTROLLER

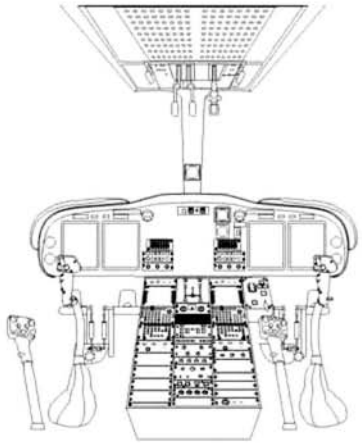
MISC CONTROL PANEL

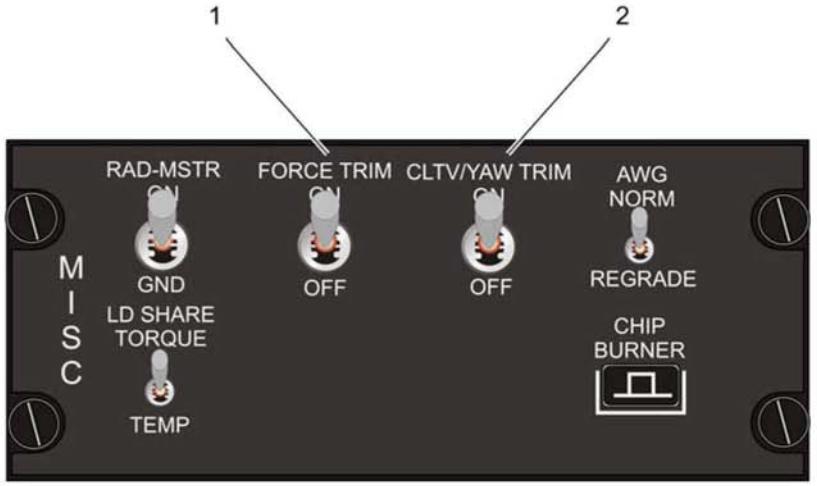
1 FORCE TRIM switch

- OFF: ..... Disengages the cyclic Force Trims (Pitch and Roll) and disables ATT mode of the AP
- ON : ..... Engages the cyclic Force Trims (Pitch and Roll) and enables ATT mode of the AP

2 CLTV / YAW TRIM switch

- OFF: ..... Disengages the collective and pedal Force Trims
- ON : ..... Engages the collective and pedal Force Trims





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MISC CONTROL PANEL

COLLECTIVE LEVER

1 BEEP CLTV / YAW TRIM switch (on PLT and CPLT collective grip)

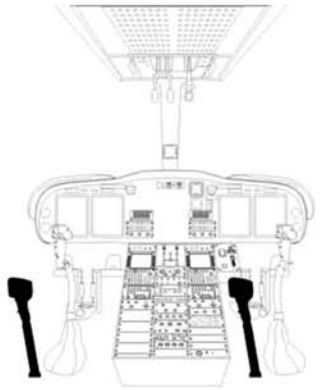
- DN / UP: ..... Allows trimming the collective axis if the CLTV / YAW TRIM switch on Miscellaneous control panel is ON. Changes the reference target parameter value if a FD collective mode is engaged.
- L / R: ..... Allows trimming the yaw axis if the CLTV / YAW TRIM switch on Miscellaneous control panel is ON

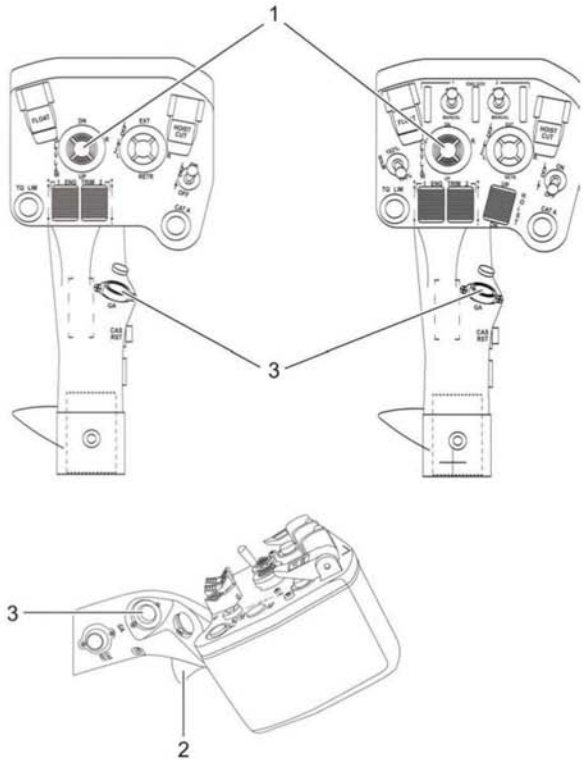
2 FTR (Force Trim Release) push-button switch (on PLT and CPLT collective grip)

- PRESSED: ..... Disengages the collective Force Trim suspending collective trimming and force feel
- RELEASED: ..... Re-engages the collective Force Trim restoring collective trimming and force feel.
  If a FD collective mode is engaged, the reference target parameter is reset to the current value

3 GA (GO AROUND) mode push-button switch (on PLT and CPLT collective grip)

- PRESSED: ..... Engages the GA mode





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COLLECTIVE LEVER

CYCLIC STICK

1 BEEP TRIM switch (on PLT and CPLT cyclic grip)

- DN / UP: ..... Allows trimming the pitch axis if the FORCE TRIM switch on Miscellaneous control panel is ON and ATT mode is selected on AP
- L / R: ..... Allows trimming the roll axis if the FORCE TRIM switch on Miscellaneous control panel is ON and ATT mode is selected on AP
- PRESSED: ..... Engages the HOV mode on FD (4 Axis Enhanced FD only)

2 FTR push-button switch (on PLT and CPLT cyclic grip)

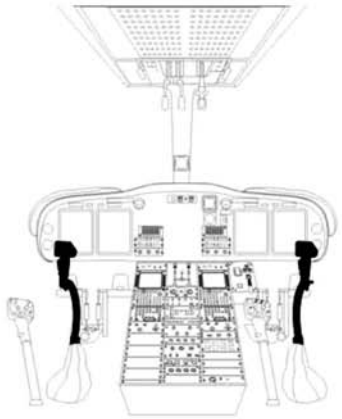
- PRESSED: ..... Disengages the cyclic Force Trims (Pitch and Roll) suspending collective trimming and force feel.
  Re-centers the Pitch and Roll Linear Actuators.
- RELEASED: ..... Re-engages the cyclic Force Trims (Pitch and Roll) restoring cyclic trimming and force feel.
  If a FD pitch or roll mode is engaged, the relevant reference target parameter is reset to the current value

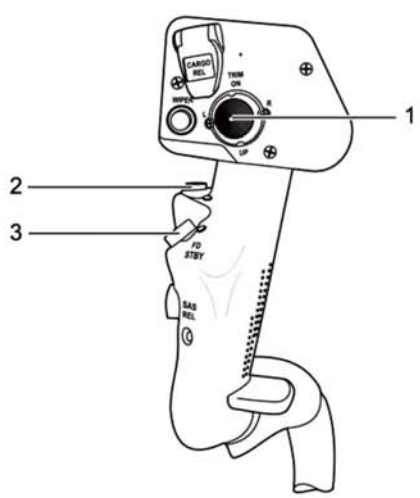
3 FD STBY (Flight Director Standby) switch (on PLT and CPLT cyclic grip)

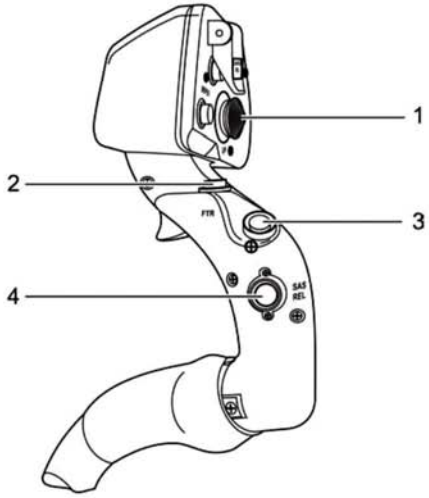
- PRESSED: ..... Illuminates the green annunciator light and cancels any selected active flight director modes

4 SAS REL push-button switch (on PLT and CPLT cyclic grip)

- PRESSED: ..... Disengages both Autopilots (AP 1 and AP 2)







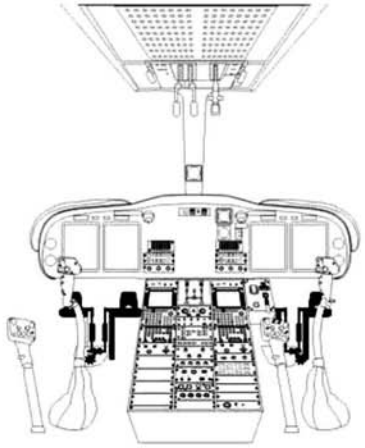
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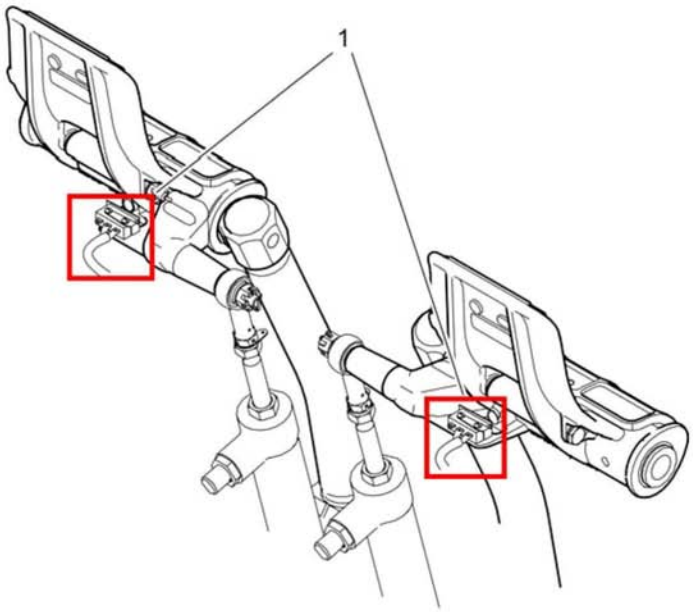
CYCLIC STICK

PEDALS

1 YAW force trim release switch (on PLT and CPLT pedals)

- PRESSED: ..... Disengages the pedal Force Trim (Yaw) suspending yaw trimming and force feel.
  Re-centers the Yaw Linear Actuator
- RELEASED: ..... Re-engages the pedal Force Trims (Yaw) restoring yaw trimming and force feel





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PEDALS

AUTOPILOT INDICATORS

1 SAS annunciator

- In view: ..... Indicates that AP is in SAS mode

2 UCPL annunciator

- In view: ..... Indicates that Flight Director is uncoupled (all axes) from the AP

NOTE H

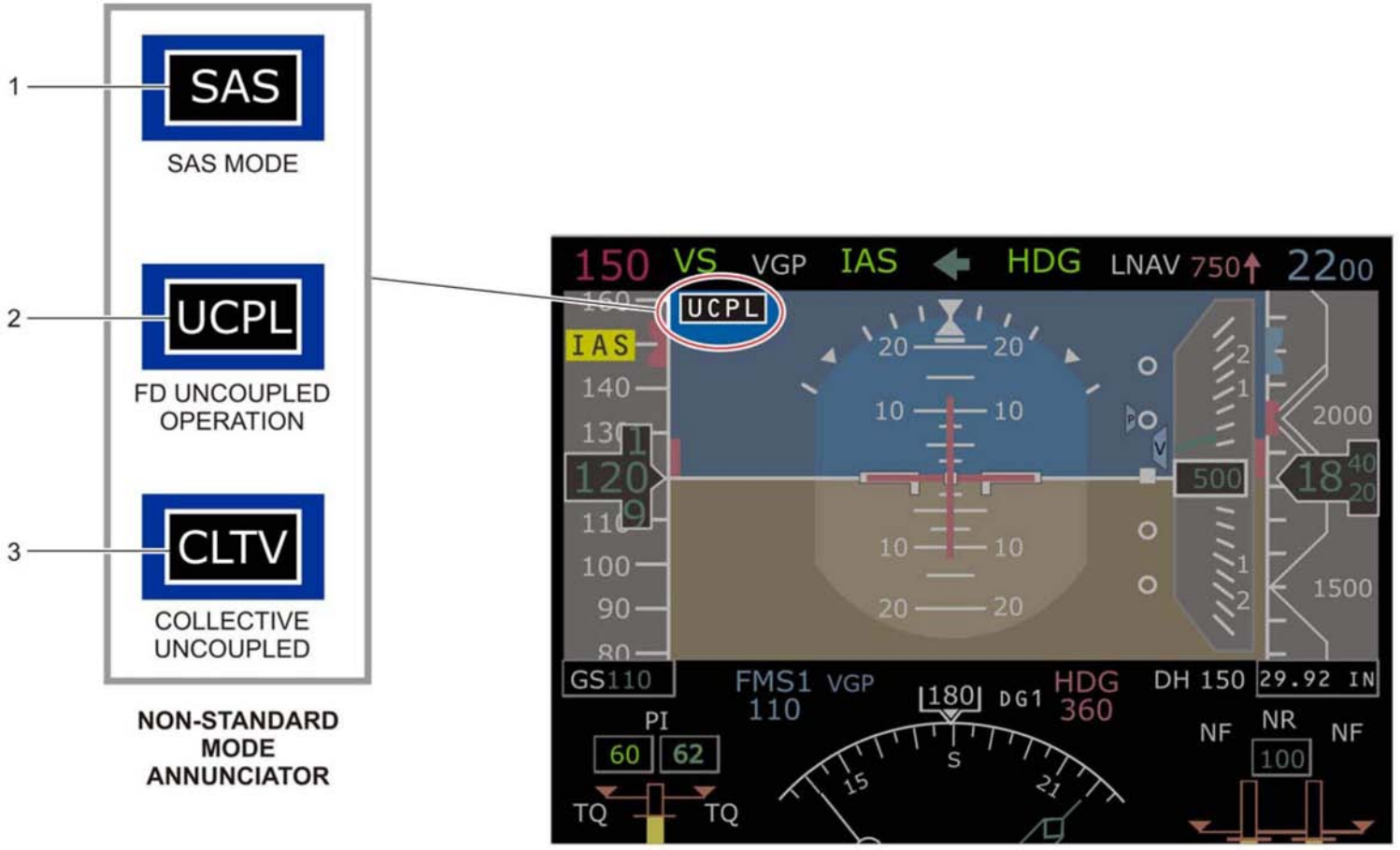
The UCPL annunciator is not in view when the AP is in SAS mode

3 CLTV annunciator

- In view: ..... Indicates that Flight Director is uncoupled from the AP on the Collective axis only

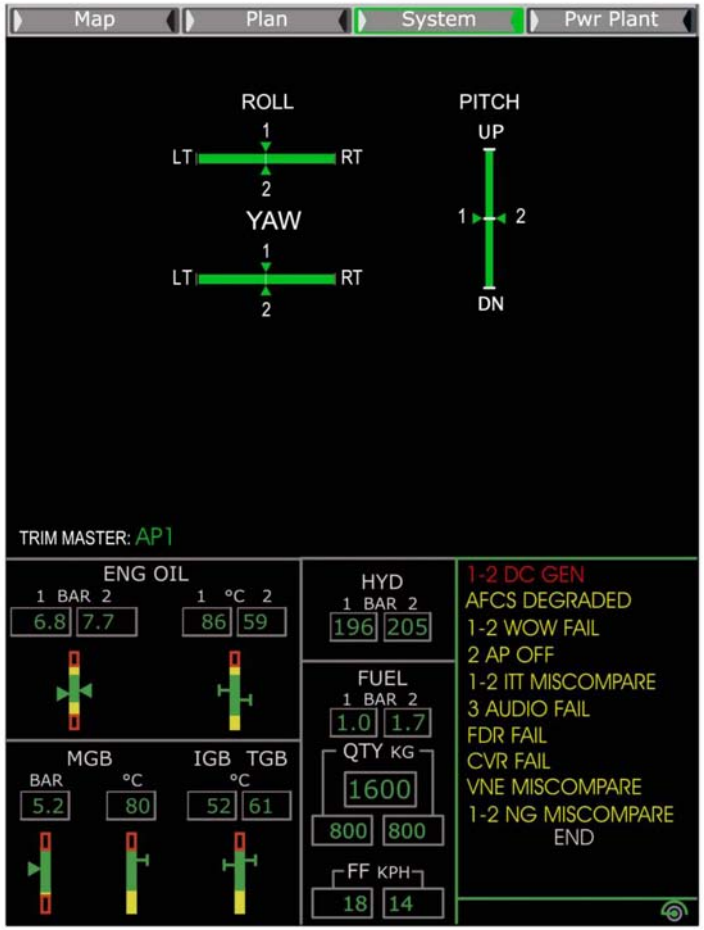
NOTE I

The CLTV annunciator is not in view when the AP is in SAS mode or FD is uncoupled on all axes



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AUTOPILOT INDICATORS



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AUTOPILOT SYNOPTIC PAGE

CAS CAUTION MESSAGES			
CAS CAPTION	**FAILURE DESCRIPTION**	**PROCEDURE NAME**	**AW139-RFM-4D**
1(2) AP FAIL + aural message + AFCS DEGRADED	Associated autopilot failure	AUTOPILOT FAIL	Section 3 EMERGENCY AND MAL-FUNCTION PROCEDURES
1(2) AP P(R)(Y) FAIL + aural message	Pitch (Roll) (Yaw) axis of AP 1(2) un-commanded disengagement	AUTOPILOT AXIS DISENGAGE	
1(2) AP OFF + aural message + AFCS DEGRADED	Associated AP not switched ON	AUTOPILOT OFF	

CAS CAUTION MESSAGES			
CAS CAPTION	**FAILURE DESCRIPTION**	**PROCEDURE NAME**	**AW139-RFM-4D**
2 AP P(R)(Y) TRIM OFF + aural message	Pitch (Roll) (Yaw) axis of AP 1(2) not engaged	AUTOPILOT AXIS OFF	
ATT OFF	Attitude system not engaged or Cyclic force trim failure if associated with cyclic freedom of motion in longitudinal and/or lateral direction with loss of function of FTR switch and cyclic beep trim. NOTE J With ATT system not engaged the aircraft flies in SAS mode only (SAS message on PFD)	ATTITUDE SYSTEM OFF and CYCLIC FORCE TRIM FAILURE	
1(2) TRIM FAIL	Trim system failure on all axes while AP 1(2) has priority (Trim Master)	AFCS TRIM FAILURE	
1 P(R)(Y) TRIM FAIL	Failure of Pitch (Roll) (Yaw) trim while AP 1 has priority	PITCH, ROLL, YAW TRIM FAIL	
2 P(R)(Y) TRIM FAIL	Failure of Pitch (Roll) (Yaw) trim while AP 2 has priority	PITCH, ROLL, YAW TRIM FAIL	

CAS CAUTION MESSAGES			
CAS CAPTION	**FAILURE DESCRIPTION**	**PROCEDURE NAME**	**AW139-RFM-4D**
MISTRIM	Linear actuator(s) not centered	MISTRIM	
1(2) SAS DEGRADED	Associated SAS degraded operation	SAS DEGRADED	
AFCS DEGRADED	Associated AFCS degraded operation	SAS DEGRADED	
1(2) AP TEST ABORT	AP TEST aborted by pilot action or air-craft lifted off before test completion	AP TEST ABORT	

CAS CAUTION MESSAGES			
CAS CAPTION	**FAILURE DESCRIPTION**	**PROCEDURE NAME**	**AW139-RFM-4D**
1(2) COLL FAIL **+ aural message**	Collective axis of AP1(2) fail NOTE K Collective FD mode annunciator amber (if a coupled collective mode is active)	SINGLE COLLECTIVE AUTOPILOT FAILURE	Supplement 40 4 AXIS BASIC FLIGHT DIRECTOR and Supplement 34 4 AXIS ENHANCED FLIGHT DIRECTOR — Section 3 EMERGENCY AND MALFUNCTION PROCEDURES
1-2 COLL FAIL **+ aural message**	Collective axis of AP1(2) failure on both AP 1 and AP 2 NOTE L Collective FD mode annunciator amber (if a coupled collective mode is active)	DUAL COLLECTIVE AUTOPILOT FAILURE	

COLLECTIVE FORCE TRIM OFF OR FAIL

When a FD collective mode is engaged and coupled and the collective trim is switched OFF (CLTV/YAW TRIM switch on the Miscellaneous Control Panel) or fails:

- a chime sound is generated
- the CLTV annunciation illuminates on the top left of the ADI display
- the CLTV/YAW OFF green advisory illuminates on the CAS
- Collective modes are available uncoupled only

CYCLIC FORCE TRIM FAILURE

Cyclic force trim failure is a disconnection of the longitudinal and/or lateral clutches. The failure is usually joined to the caution

ATT OFF and the cyclic moves freely in pitch and/or roll axis with loss of function of the cyclic trim release (FTR switch) and cyclic beep trim system.
In these conditions the cyclic must be used hands-on to prevent it from moving away the selected position.

AFCS QUICK DISCONNECT PROCEDURE

For situations where faults are suspected in the AFCS, but with no CAS cautions illuminated, and the AP functions need to be disengaged, all AP/AFCS functions can be disconnected by pressing the SAS REL button on the cyclic grip.

4-AXIS FLIGHT DIRECTOR – GENERAL

The dual Flight Directors (FD1 & FD2) provide lateral and vertical guidance commands that are normally coupled to the Autopilots for automatic flight path control.

The 4 Axis (3 cue) FD provides lateral modes operating on the roll axis and vertical modes operating on the pitch and collective axes.

The 4 Axis Enhanced Flight Director provides all the functions of the 4 Axis Basic Flight Director plus the Auto Hover/Velocity Hold mode (HOV).

References:

- 4 Axis Basic FD: ..... RFM Supplement 40
- 4 Axis Enhanced FD:..... RFM Supplement 34

The 4 Axis Enhanced FD requires the following optional equipment to be installed in place or in addition to the Basic FD:

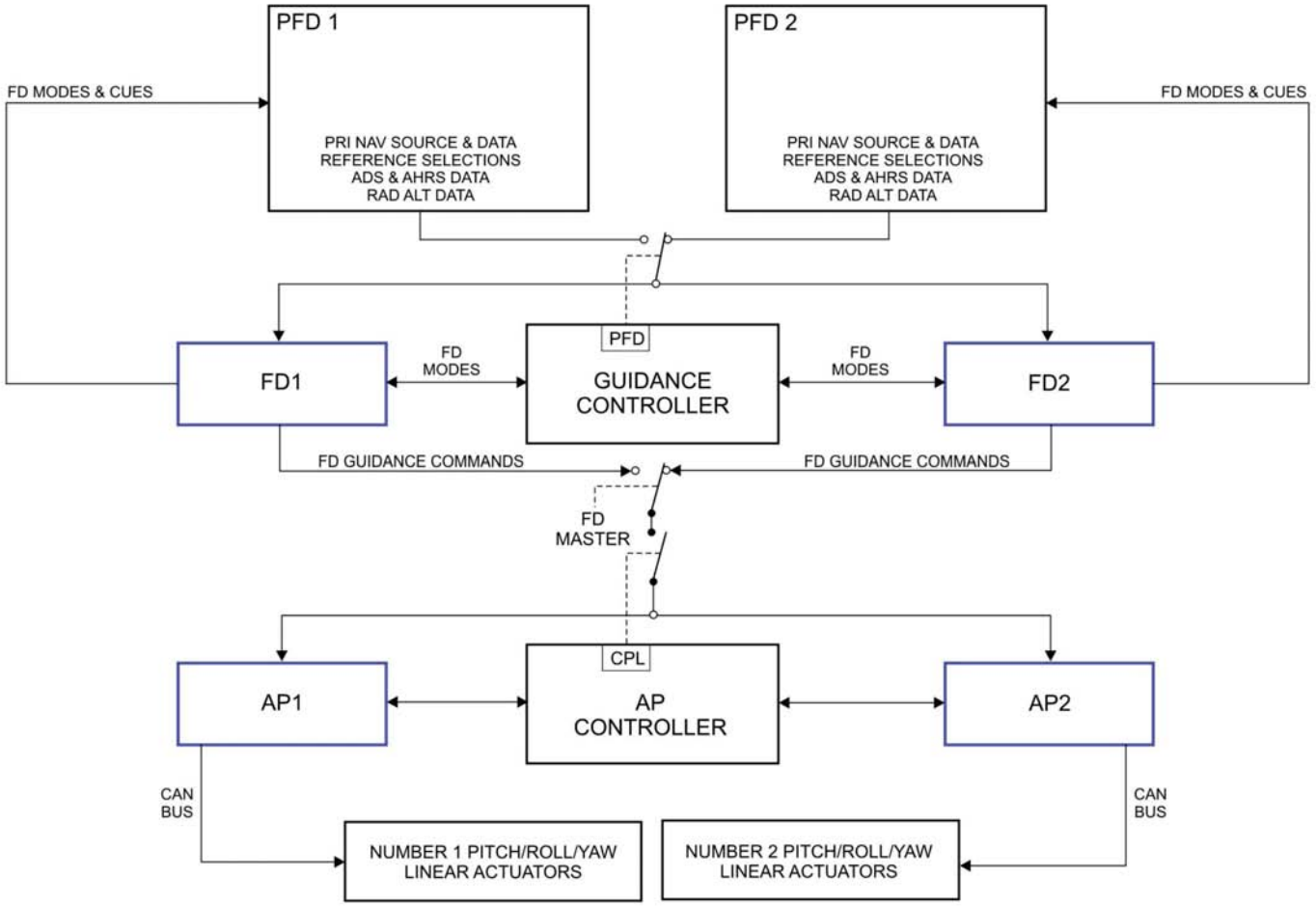
- AHRS LCR-93 (in place of LCR-92) to provide AFCS with hybrid Along-Heading and Across-Heading velocities and accelerations (in addition to LCR-92 data)
- 5-way cyclic BEEP TRIM switches (in place of 4-way switches)

FD MASTER

At power up one of the two FD is automatically selected and configured as Master (priority). At every power up the FD selected as Master is alternated: the selection is not visible to the pilot.

PFD SELECTION (PILOT-IN-COMMAND)

Both FD1 and FD2 use the navigation source and data reference presented on the selected PFD. .
PFD selection (which typically matches the “Pilot Flying” or “Pilot in Command” determination) is controlled by the PFD pushbutton on the Guidance Controller.
The selected PFD is indicated by the PFD couple arrow in the center of the top line of both PFD’s and by either the LH or the RH arrow-shaped LED annunciator aside the PFD pushbutton on the Guidance Controller.
Whenever the PFD pushbutton is pressed to toggle LH/RH PFD selection all FD modes are disengaged (STBY illuminates on the Guidance Controller).
FD modes remain engaged in case the selected PFD fails: the paired MFD reverting in Composite format keeps operating as the selected PFD.



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FLIGHT DIRECTOR FUNCTIONAL DIAGRAM

4 AXIS FLIGHT DIRECTOR - OPERATION

ENGAGING AND DISENGAGING FD MODES

Flight Director modes are engaged and disengaged via the Guidance Controller (FD Control Panel). Each mode pushbutton green light illuminates when the mode is armed or engaged (captured).
The engagement of the FD modes is made with the Guidance Controller (GC) and the communication to the AIOP modules is through the CIO modules.
Each mode may be deselected by pressing the associated pushbutton on the Guidance Controller.
All engaged modes may be deselected simultaneously by using the STBY button on the Guidance Controller or the FD STBY button located on pilot or copilot cyclic stick grips.
GA (Go Around) mode can only be engaged by using the GA button on pilot or copilot collective lever and disengaged by engaging a different vertical mode or setting FD to STBY.
Armed and captured FD modes are displayed with messages along the top line on the PFD:

- collective and pitch mode messages on the left
- lateral mode messages on the right

Armed modes are shown in white small font; captured (or engaged) modes are shown in green medium font.
When transitioning from no mode to engaged, from armed to captured or change from one mode to another mode a chime sounds and the message flashes normal/reverse video for 6 seconds then becomes steady (normal video).
When changing from captured to no mode a chime sounds and the message flashes on and off for 6 seconds then extinguishes.
Pitch and roll command bars and/or collective cue are displayed on the PFD by the AFCS when a Flight Director mode is active on the relevant channel.
When there is no FD mode active the command bars/ collective cue are not displayed.
When HOV mode is the active flight director mode, the pitch and roll command bars are not displayed.
All modes are disengaged when toggling the PFD button on Guidance Controller.
All modes - except HOV, ALT and RHT - must not be engaged below 60 KIAS and are automatically disengaged when airspeed reduces below 55 KIAS. HOV, ALT and RHT can be engaged between 0 KIAS and Vne.
All modes have a Minimum Use Height limit: see RFM Supplement 40 (or 34), Section 1 – Limitations.

FD COUPLING AND UNCOUPLING

When a FD mode is engaged (captured) the coupled (CPL) function is also automatically engaged if the AP is in ATT mode and the relevant Force Trim is engaged: the CPL pushbutton green light on the Autopilot Controller illuminates. When coupled, both autopilots drive their actuators to satisfy the references provided by the priority FD.
Selecting a FD mode by pressing the pushbutton on the Guidance Controller with ATT OFF (i.e. SAS mode or both Autopilots off) will engage the FD in the Uncoupled mode.
The FD may be manually uncoupled by pressing the CPL pushbutton on the Autopilot Controller when illuminated: the CPL green light extinguishes and the UCPL annunciator appears on both PFD's; on the ADI the command bars and/ or collective cue remain in view and the pilot can manually fly the commands. When the AP reverts to SAS no UCPL caption will appear on the PFD.
In either case (Coupled or Uncoupled) pitch, roll command bars/collective cue are presented on each PFD, depending on the mode selected.
The coupled function is also automatically forced to Uncoupled whenever:

- the AP is not in ATT Mode
- moving the CLTV/YAW TRIM switch to OFF when a collective FD mode is engaged

In case of single AP operation, when a FD mode is engaged the coupled function remains available.

Autopilots ignore the FD commands when pilot flies the helicopter manually against any artificial feel device while the FD is coupled, because of the detent microswitch inside the Trim Actuator. FD remains coupled and no change in FD settings occurs.

FD AURAL TONES

MODE CHANGE TONE

A chime sounds whenever any of the following occurs:

- Autopilot mode changes from ATT to SAS or vice versa
- A FD mode changes status from disengaged to engaged or vice versa
- A FD mode changes status from armed to captured
- Coupling status (CPL) changes from Coupled to Uncoupled or vice versa

Arming of modes does not trigger the tone.
When a collective FD mode is engaged and coupled, the tone is triggered by moving the CLTV/YAW TRIM switch to OFF.

ALTITUDE REFERENCE CHANGE TONE

An aural tone (“bip-bip”) sounds for any change in altitude or radar height mode reference datum.

ALTITUDE ALERTER

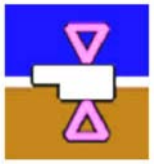
When ALT (Altitude Holding) or RHT (Radar Height Holding) modes are engaged, an aural warning message “ALTITUDE – ALTITUDE” is generated whenever aircraft exceeds a predetermined altitude/height deviation.

FLIGHT DIRECTOR INDICATORS

1 COLLECTIVE REFERENCE MARKERS

(In view only if a FD Collective mode is engaged)
The collective reference markers (fixed) are set as two hollow magenta triangles pointing towards each other: they represent the required position of the collective lever to fulfil the FD collective command

2 COLLECTIVE CUE



(In view only if a FD Collective mode is engaged). The collective cue represents the current position of the collective lever relative to reference markers.
If collective cue is shown below the reference markers, the collective lever is to be pulled up, and viceversa.

3 ENGAGED VERTICAL COLLECTIVE MODE Annunciator
Shows what collective mode is engaged (captured)

- Green: ..... Flight Director is operating normally

- Amber: ..... Collective trim actuator is failed or collective pitch mode (ie ALT) is affected by the FD torque limiting function

4 ARMED VERTICAL MODE Annunciator (white)
Shows what vertical mode(s) is(are) armed (collective and/or pitch)

5 ENGAGED VERTICAL PITCH MODE Annunciator
Shows what pitch mode is engaged (captured)

- Green: ..... Flight Director is operating normally
- Amber: ..... Both linear actuators are failed or vertical pitch mode (ie IAS) is affected by the FD torque limiting function

6 ENGAGED LATERAL MODE Annunciator
Shows what roll mode is engaged (captured)

- Green: ..... Flight Director is operating normally
- Amber: ..... Both linear actuators are failed

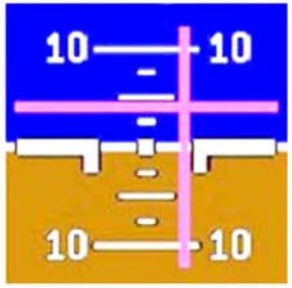
7 ARMED LATERAL MODE Annunciator (white)
Shows what lateral mode is armed (roll)

- Green: ..... Flight Director is operating normally
- Amber: ..... Both linear actuators are failed

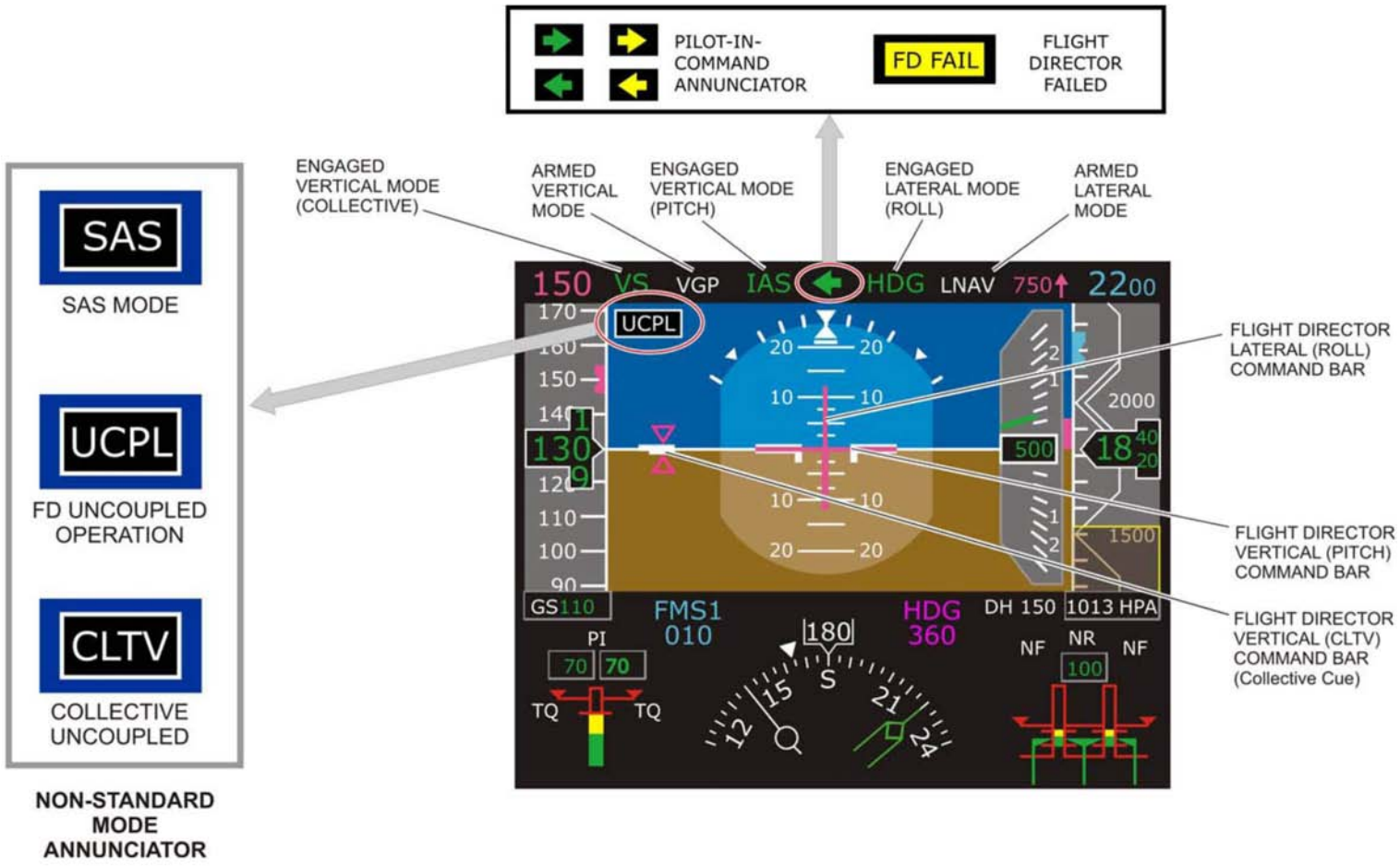
8 FLIGHT DIRECTOR SOURCE SELECT ARROW AND STATUS
The source select arrow points towards the PFD that is supplying navigation and mode select data to both flight directors.

- Green: ..... Associated Flight Director is valid
- Amber: ..... Associated Flight Director is invalid
- FD FAIL flag: ..... Both Flight Directors invalid (replaces the arrow annunciator)

9 FD COMMAND BARS

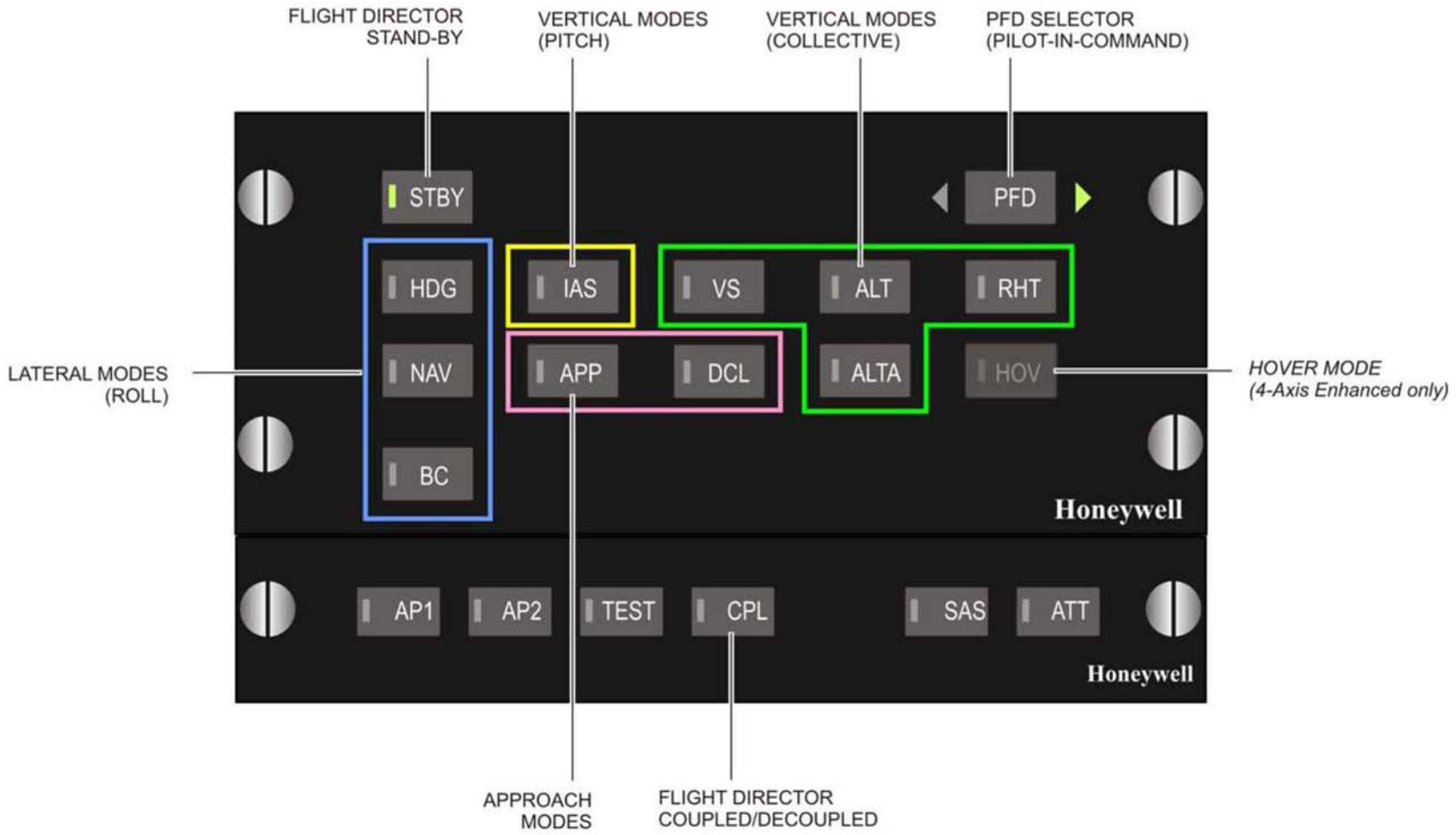


Show the amount of roll (Lateral command bar) or pitch (Vertical command bar) guidance command from the Flight Director.
Either command bar is in view only if a relevant channel mode (roll or pitch) is engaged and is read with respect to the ADI aircraft symbol.
When a FD mode is engaged, and remains uncoupled from the AP, it permits the pilot to manually fly the aircraft using the directional cues presented by the command bars.
To follow the command bars, the pilot manually flies the aircraft to where the command bars intersect on the ADI.
In the example aside, the aircraft is below and to the left of the desired course and altitude. The pilot should execute a climbing, right-hand turn to place the aircraft on course and at the designated altitude.



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FLIGHT DIRECTOR INDICATIONS



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4- AXIS BASIC FLIGHT DIRECTOR MODE SELECTION

Mode	Function	Control Axis	PFD caption	
			ARM	CAPTURE
HDG	Heading select	Roll and Yaw	N/A	HDG
ALT	Altitude Hold	Collective	N/A	ALT
IAS	Indicated Airspeed Hold	Pitch	N/A	IAS
NAV	Lateral navigation	Roll	LNAV VOR LOC	LNAV VOR LOC
APP	Lateral Approach	Roll for VOR or LOC	VAPP LOC	VAPP LOC
	Vertical Approach	Collective for Glide Slope or VGP	GS VGP	GS VGP
PRV	Preselect VOR-ILS approach while in LNAV or Preselect VGP	N/A	VAPP LOC GS VGP	N/A

Mode	Function	Control Axis	PFD caption	
			ARM	CAPTURE
DCL	ILS Deceleration VGP Deceleration	Pitch	DCL	DCL
BC	Back Course Approach	Roll	BC	BC
ALTA	Altitude Acquire	Collective	N/A	ALTA
VS	Vertical Speed Hold	Collective	N/A	VS
GA	Go-Around	Pitch for AS function/ Collective for VS function	N/A	GA
VGP	Vertical Glide Path	Collective	VGP	VGP
RHT	Radar Height Hold	Collective	N/A	RHT
HOV *	Hover / Velocity Hold	Pitch / Roll	N/A	HOV
OS **	Over Station mode	Roll	N/A	OS
ALVL **	Auto-Level mode	Pitch / Collective	N/A	ALVL
SBY	Clear all FD modes	N/A	N/A	N/A

* Only for 4-axis Enhanced FD

** Provided automatically

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FLIGHT DIRECTOR MODES



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4-AXIS BASIC FD MODES

COLLECTIVE PI LIMITING FUNCTION

During collective coupled operation the collective movement is limited by the following Power Index (PI) values:

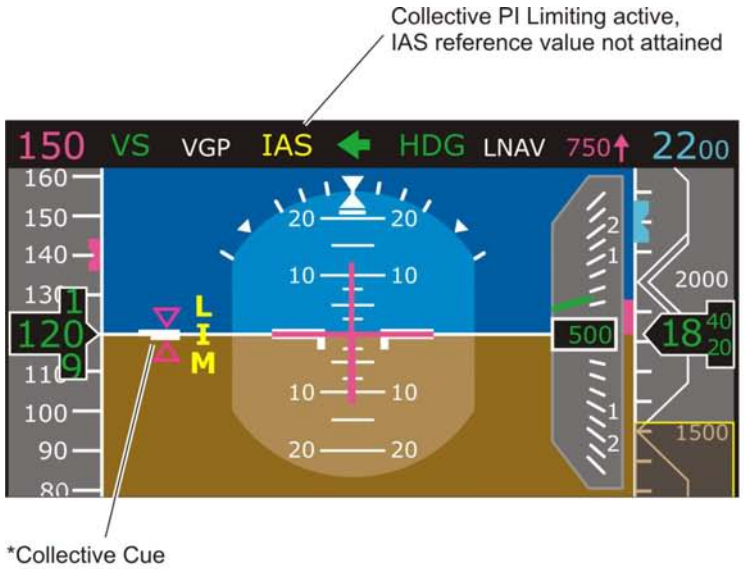
- Maximum 97% AEO (reduced to 95% at altitudes above 10000 ft Hp)
- Maximum 106% AEO for airspeed less than 60 KIAS (5 MIN message displayed beside collective cue)
- Maximum 140% OEI
- Minimum 5% AEO
- Minimum 10% OEI

When PI limiting function is active and is limiting maximum collective movement, an amber LIM caption is displayed beside the collective cue.
If PI limiting is active with IAS engaged and the required reference for VS, ALTA, RHT, GA, or ALT cannot be achieved then the airspeed will automatically reduce as necessary to achieve the collective mode reference.
If the collective mode reference cannot be maintained when the airspeed has reduced to a minimum of 80 KIAS, this airspeed will be maintained (IAS caption is displayed in green) and the collective mode reference will be reduced (the collective mode caption is displayed in amber).

CAUTION

IF PI LIMITING IS ACTIVE WITH ALT OR RHT ENGAGED AND THE REFERENCE BAROMETRIC ALTITUDE OR RADIO HEIGHT CANNOT BE MAINTAINED, THE AURAL WARNING “ALTITUDE – ALTITUDE” WILL WARN THE PILOT WHEN THE MAXIMUM ALLOWED DEVIATION FROM THE REFERENCE SETTING HAS BEEN EXCEEDED.

When flying at high altitude (above 15000 ft), selecting the LD SHARE switch on MISC panel to TORQUE improves the helicopter manoeuvering during automatic turns.

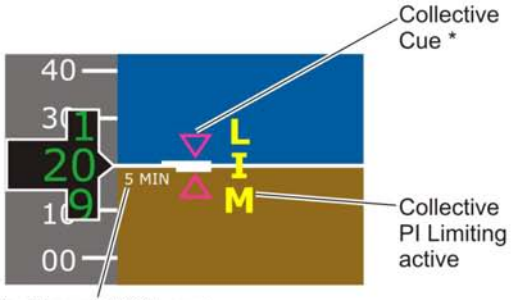


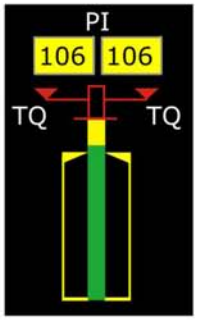


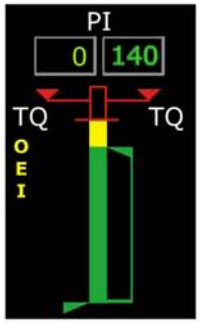
PI LIMITING

- **Max 97% AEO** (reduced to 95% at altitudes above 10000 ft Hp)
- **Max 106% AEO for IAS < 60 KIAS** (5 MIN message displayed beside collective cue)
- **Max 140% OEI**
- Min 5% AEO
- Min 10% OEI

Minimum IAS = 80 KIAS







*In UCPL operation when collective cue is below reference arrows collective must be raised and viceversa

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COLLECTIVE PI LIMITING

FD MODES

The following descriptions assume the modes are coupled, unless stated otherwise.

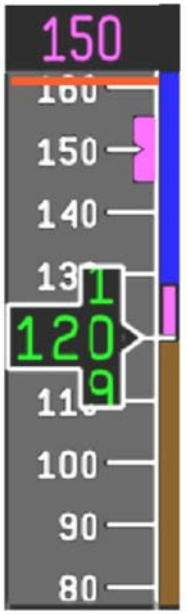
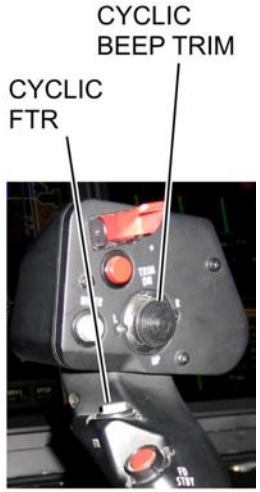
HDG — HEADING HOLD MODE		
Description	The HDG mode provides the capability to steer the aircraft to capture and hold the Selected Heading reference displayed as the Heading Bug and associated digital readout on the HSI. Aircraft axis control is performed via roll attitude. The HDG select mode is supported with turn coordination functionality on the yaw axis.	
Engagement	• Direct:..... Press HDG button on Guidance Controller • Automatic:..... HDG mode automatically engages when any of the following mode is selected (and armed): VOR, VAPP, LOC, BC, LNAV When engaged Heading Bug and Selected Heading readout on both HSIs are shown in magenta.	

HDG — HEADING HOLD MODE		
References and performance	The Selected Heading reference is the Heading Bug on the HSI. Heading bug can be set and adjusted (even if HDG mode is not engaged) by either: • rotating the HEADING knob on either pilot or copilot Remote Instrument Controller • setting either pilot or copilot cyclic Beep TRIM switches to L or R (Beep rate: ±3°/s for first 3 seconds and then ±10°/s) NOTE M Only one Selected Heading is available on the helicopter; both pilot and copilot are always enabled to change it. Heading bug can be synchronized with the actual heading (even if HDG mode is not engaged) by either: • pressing the PUSH SYNC button on HEADING knob on either pilot or copilot Remote Instrument Controller • pressing the cyclic FTR button on either pilot's cyclic while HDG mode is engaged • Heading bug is also synchronized with the actual heading when HDG mode is automatically engaged. In HDG mode turns are performed at standard rate 1 (i.e. 3°/s).	



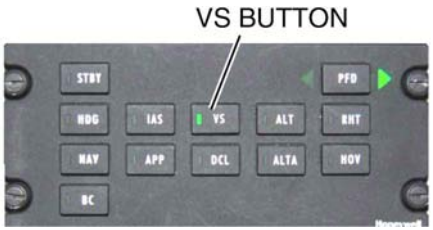
ICN-39-A-220000-G-A0126-00143-A-01-1

IAS — INDICATED AIRSPEED HOLD MODE		
Description	The IAS hold mode generates pitch commands to maintain a selected aircraft airspeed.	
Engagement	• Direct:..... Press IAS button on Guidance Controller • Automatic:..... IAS mode automatically engages when ALTA mode is selected or DCL mode is armed	
Initial conditions	IAS mode maintains the airspeed existing at the time of engagement.	

IAS — INDICATED AIRSPEED HOLD MODE		
References and performance	When engaged the airspeed reference is displayed as a set bug and a digital readout in magenta on the Airspeed Indicator on PFDs. The airspeed reference bug is adjusted by either: • setting either pilot or copilot cyclic Beep TRIM switches to UP (to reduce IAS) or DN (to increase IAS) (Beep rate: ±3.5 kts/s) • repositioning the cyclic, with the FTR button depressed, to the required airspeed and releasing FTR button. The airspeed reference is adjustable via the cyclic beep switch between 60 KIAS and (Vne – 5) KIAS. If airspeed, at time of engagement, is higher than (Vne – 5) KIAS, then it is automatically reduced to (Vne – 5) KIAS. Note that the IAS reference will not increase when Vne increases. The IAS reference is resynchronized when there is a transition from FD Un-coupled to FD Coupled.	 



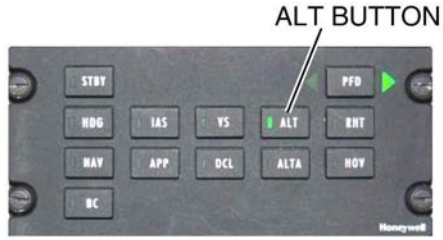
ICN-39-A-220000-G-A0126-00150-A-01-1

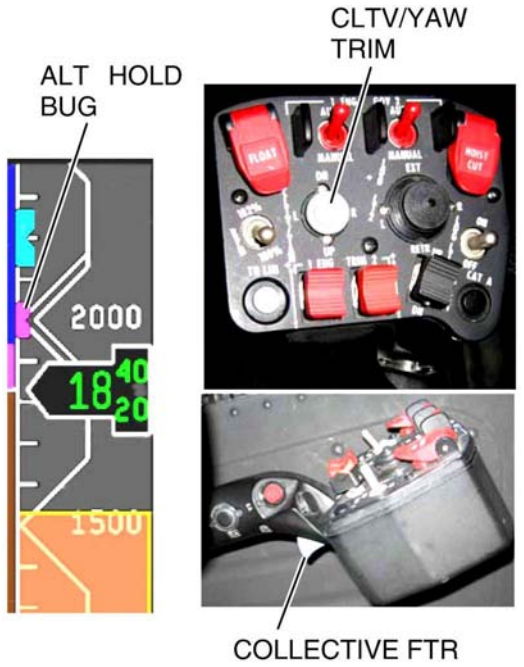
VS — VERTICAL SPEED HOLD MODE		
Description	The VS hold mode generates collective commands to maintain a selected aircraft vertical speed.	
Engagement	• Direct:..... Press VS button on Guidance Controller • Automatic:..... VS mode automatically engages when ALTA mode is selected	

VS — VERTICAL SPEED HOLD MODE		
References and performance	VS mode maintains the vertical speed existing at the time of engagement. When engaged the vertical speed reference is displayed as a set bug and a digital readout in magenta on the Vertical Speed Indicator on PFDs. The vertical speed reference bug is adjusted by either: • setting either pilot or copilot collective Beep Trim CLTV/YAW switches to UP (to increase VS) or DN (to reduce VS) (Beep rate: ±150 fpm/s) • pressing the collective FTR switch while flying to the desired vertical speed, and then releasing the FTR. The VS mode reference can be set in the range of –1500 fpm to +2000 fpm. The VS reference is resynchronized when there is a transition from FD Uncoupled to FD Coupled.	



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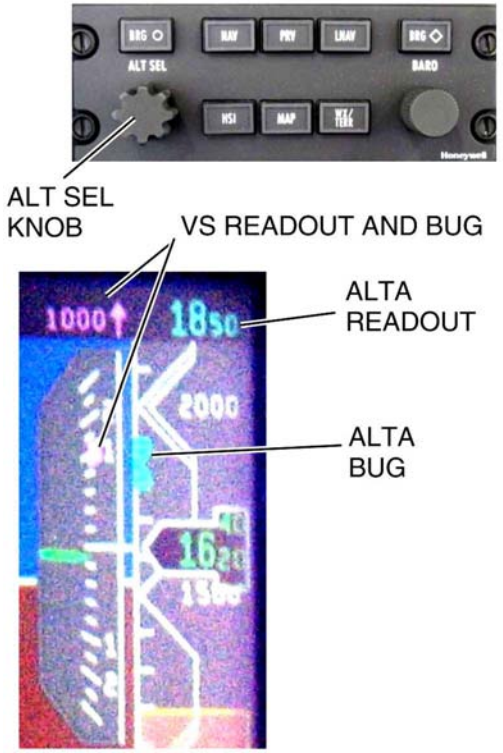
ALT — BARO ALTITUDE HOLD MODE		
Description	The ALT hold mode generates collective commands to maintain a selected aircraft barometric altitude.	
Engagement	• Direct:..... Press ALT button on Guidance Controller • Automatic:..... ALT mode automatically engages when ALTA mode has acquired the pre-selected altitude	

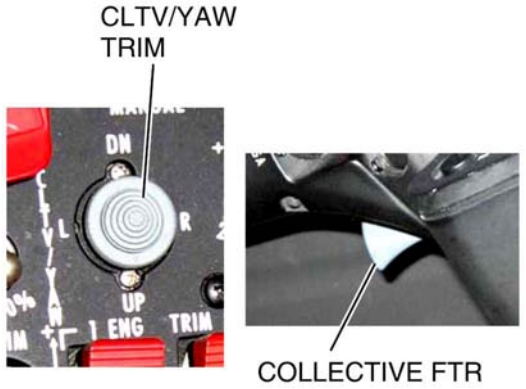
ALT — BARO ALTITUDE HOLD MODE		
References and performance	ALT mode maintains the barometric altitude existing at the time of engagement. When engaged the baro altitude hold reference is displayed as a small magenta bug on the Barometric Altimeter on PFDs. The altitude hold reference bug is adjusted by either: • setting either pilot or copilot collective Beep Trim CLTV/YAW switches to UP (to increase altitude) or DN (to reduce altitude) (Beep rate: ±50 ft/s) • pressing the collective FTR switch while flying to the desired barometric altitude, and then releasing the FTR. Changing the ALT reference causes a Reference Change aural tone to be played when the reference begins to change. While the reference is being changed and for 5 seconds after finishing, no Reference Change aural tone is played. Adjustment of the barometric setting value displayed on the selected PFD will induce a corresponding change of the indicated altitude. When changing the barometric setting value with the ALT mode engaged, the mode will command a climb or descent as necessary to return to the barometric altitude corresponding to the respective value indicated at the time ALT mode was engaged, last synchronized or beeped. ALT mode can be engaged with HOV mode as an alternative to the RHT mode.	



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ALTA — ALTITUDE ACQUIRE MODE		
Description	The ALTA mode generates collective commands to climb or descend towards the pre-selected barometric altitude reference set with the Altitude Pre-Select knob (ALT SEL) on the Display Controller.	
Engagement	• Direct:..... Press ALTA button on Guidance Controller	

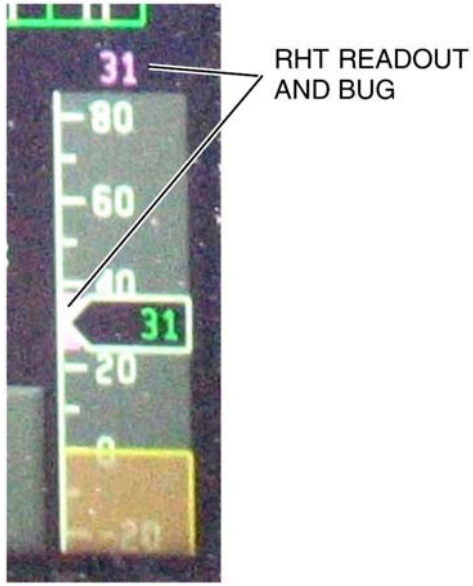
ALTA — ALTITUDE ACQUIRE MODE		
References and performance	At power up, the digital readout for the ALTA reference will be displayed as dashes (invalid) until the ALT SEL knob is moved at least one click. When ALTA mode is engaged: • a vertical speed of either +1000 fpm or –750 fpm in the direction of the target altitude is automatically set • IAS mode is automatically engaged holding the current airspeed (60 KIAS minimum) The ALTA mode vertical speed reference is displayed as a set bug and as a digital readout on the PFD. The vertical speed reference for the ALTA mode can be changed by the pilot using one of the following means: • setting either pilot or copilot collective Beep Trim CLTV/YAW switches to UP (to increase VS) or DN (to reduce VS) (Beep rate: ±150 fpm/s) • pressing the collective FTR switch while flying to the desired vertical speed, and then releasing the FTR. The VS mode reference can be set in the range of –1500 fpm to +2000 fpm.	

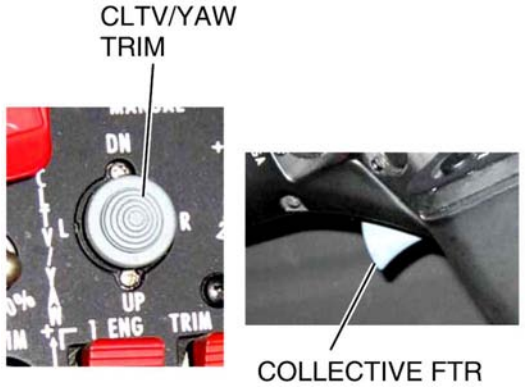
ALTA — ALTITUDE ACQUIRE MODE		
	The minimum vertical speed reference is 100 fpm in the direction of the target altitude Changing the altitude preselect reference while ALTA is engaged, will cause the mode to attempt to capture the new reference once the altitude preselect knob is released. As the aircraft approaches and captures the pre-selected altitude, the ALTA mode automatically transitions to the ALT mode and IAS mode remains engaged.	

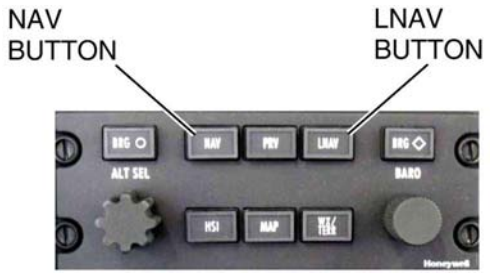


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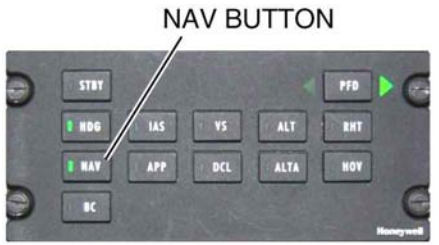
RHT — RADIO HEIGHT HOLD MODE		
Description	The RHT hold mode generates collective commands to maintain a selected aircraft radio height.	
Engagement	• Direct:..... Press RHT button on Guidance Controller • Automatic:..... RHT mode automatically engages when • HOV mode has engaged • Collective is beeped in ALVL mode	

RHT — RADIO HEIGHT HOLD MODE		
References and performance	RHT mode maintains the radio height existing at the time of engagement. When engaged the radio height hold reference is displayed set bug and a digital read-out in magenta on the Radio Altimeter on PFDs.	

RHT — RADIO HEIGHT HOLD MODE		
	The radio height hold reference bug is adjusted by either: • setting either pilot or copilot collective Beep Trim CLTV/YAW switches to UP (to increase height) or DN (to reduce height) (Beep rate: ±50 ft/s) • pressing the collective FTR switch while flying to the desired radio height, and then releasing the FTR.	

NAV MODE — VOR / LOC / LNAV FUNCTIONS		
Description	The NAV mode generates roll commands to steer the aircraft to capture and hold the Selected VOR Course, ILS Localizer or FMS Desired Track dependent on the Navigation system selected as Primary Navaid. Primary Navaid is selected by the pilot in command via the Display Controller (DC) and displayed as the CDI and the associated digital readouts on the HSI. • NAV button on DC:..... VOR or LOC • LNAV button on DC..... FMS	

<table>
<tr><th colspan="3">NAV MODE — VOR / LOC / LNAV FUNCTIONS</th></tr>
<tr><td>Engagement</td><td>• Arming:..... Press NAV button on Guidance Controller.
• NAV mode is armed (VOR, LOC or LNAV)
• HDG mode is automatically engaged and the present heading held

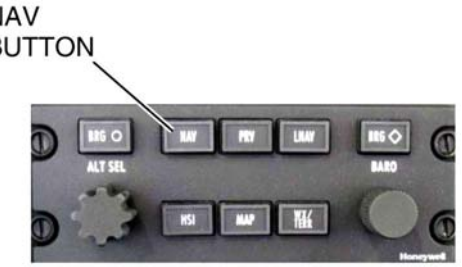
<u>NOTE N</u>
If LOC is selected as Primary Navaid, NAV mode selection does not provide Glide Slope guidance</td><td> 
</td></tr>
<tr><td>References and performance</td><td>The VOR / LOC Course can be set and adjusted by the COURSE knob on the Remote Instrument Controller (RIC).
The FMS Desired Track is automatically computed by the FMS.
VOR Course only can be synchronized with the current VOR bearing value by pressing the PUSH DIR button on COURSE knob on the RIC.
• Capture:..... Automatic capture of the Selected VOR Course, ILS Localizer or FMS Desired Track
• occurs at lateral beam sensing point (navigation source must be valid),
• disengages the HDG mode
• activates the VOR, LOC or LNAV mode, respectively</td><td> 
</td></tr>
</table>

NAV MODE — VOR / LOC / LNAV FUNCTIONS		
	When VOR mode is captured, the VOR deviation (difference between VOR bearing and course selection) is gain-programmed as a function of distance from the station (DME and/or FMS). If distance is not available, the gain-programming uses the default values optimized for cruise. The AFCS uses a course error signal to immediately correct the short-term heading disturbances such as wind gust. With a crosswind, a course error offset (crab angle) is computed and used by the FD to keep the aircraft on course.	

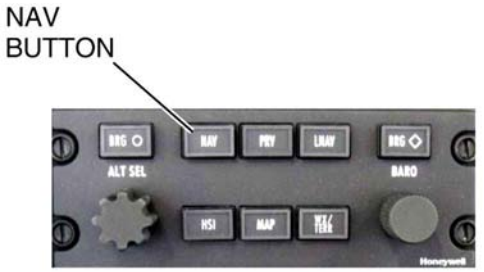


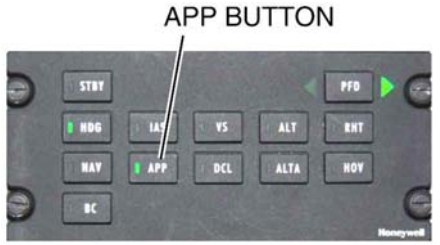


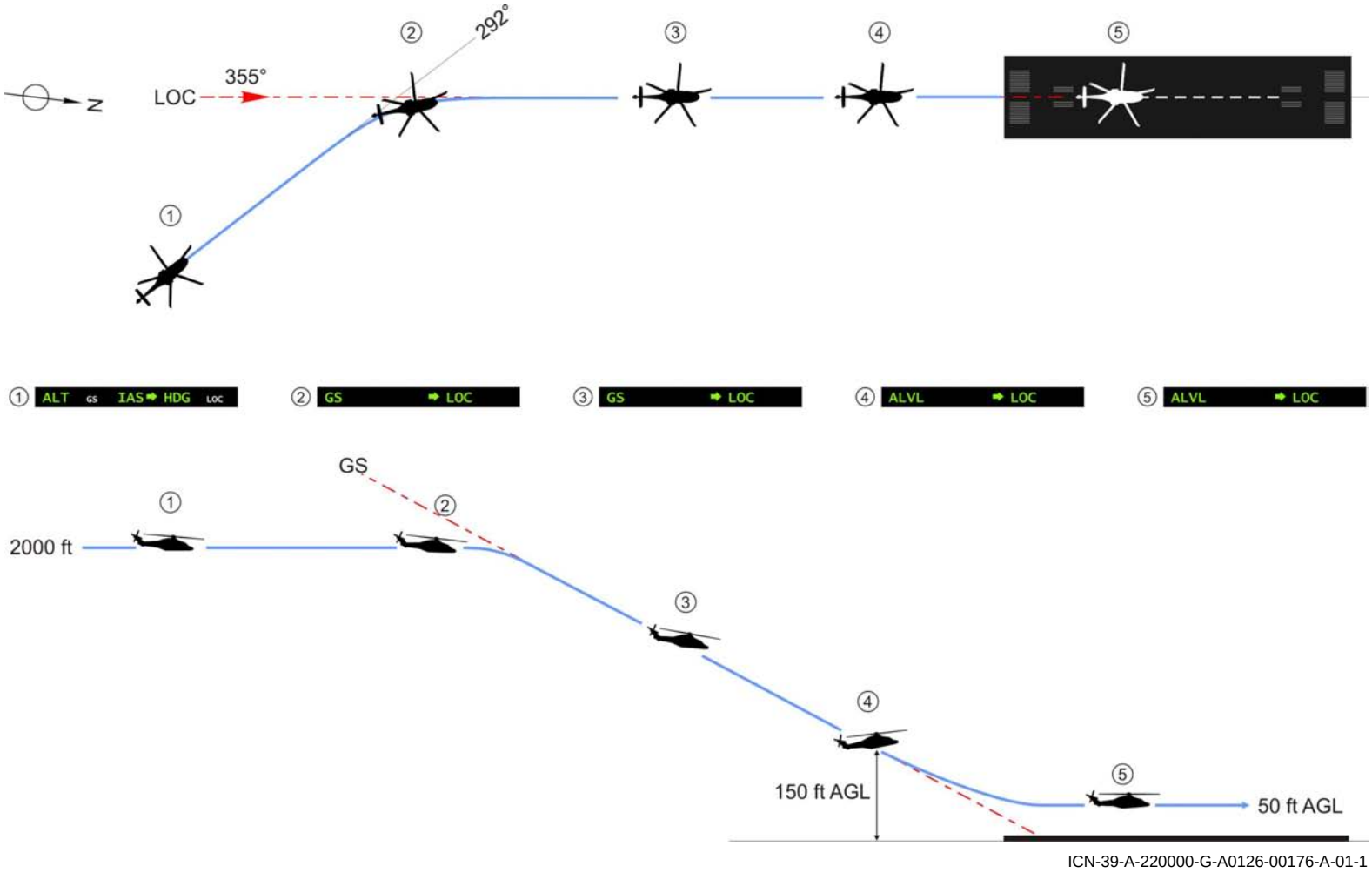
ICN-39-A-220000-G-A0126-00169-A-01-1

APP MODE — VOR APPROACH FUNCTION		
Description	The VAPP (VOR Approach) mode provides the capability to steer the aircraft to fly a VOR non-precision approach. VOR must be selected by the pilot in command as the Primary Navaid via the NAV button on the Display Controller (DC) and displayed as the CDI and the associated digital readout on the HSI. • Aircraft axes control is performed via roll attitude for VAPP.	
Engagement	• Arming:..... Press APP button on Guidance Controller • VAPP mode is armed • HDG mode is automatically engaged and the present heading held	

APP MODE — VOR APPROACH FUNCTION		
References and performance	The VOR Course can be set and adjusted by the COURSE knob on the Remote Instrument Controller (RIC). VOR Course can also be synchronized with the current VOR bearing value by pressing the PUSH DIR button on COURSE knob on the RIC. At VOR radial capture, HDG mode automatically disengages and VAPP mode automatically engages, so the helicopter turns to track the selected radial.	

APP MODE — ILS APPROACH FUNCTION		
Description	The APP mode provides the capability to steer the aircraft to capture and hold the ILS Localizer (LOC) and Glide Slope (GS). LOC must be selected by the pilot in command as the Primary Navaid via the NAV button on the Display Controller (DC) and displayed as the CDI and the associated digital readout on the HSI. Aircraft axes control is performed via: • roll attitude for LOC • collective input for GS	

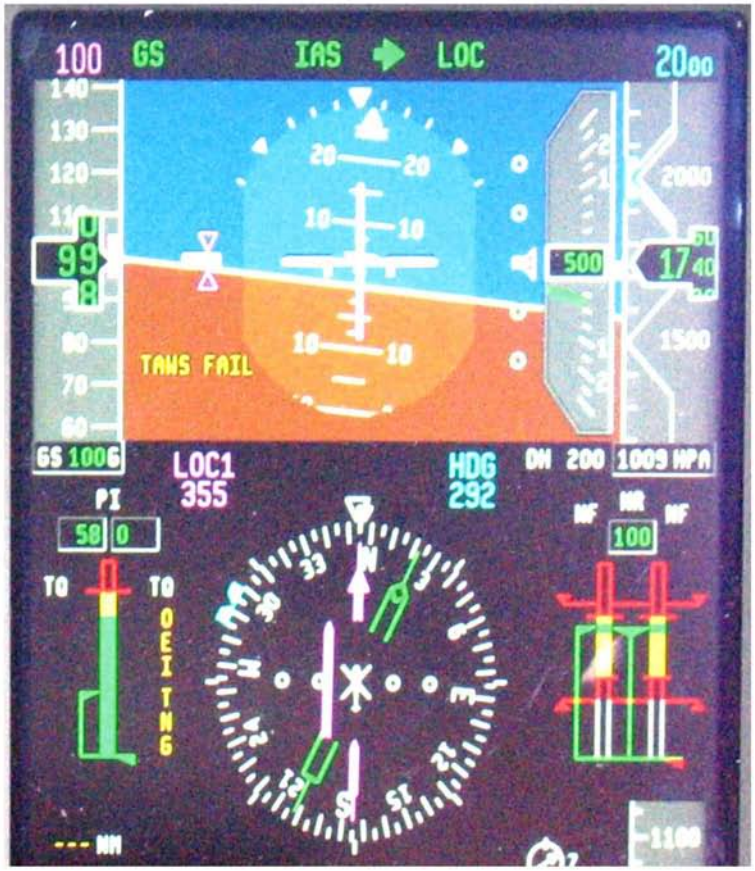
APP MODE — ILS APPROACH FUNCTION		
Engagement	• Arming:..... Press APP button on Guidance Controller • GS and LOC modes are armed • HDG mode is automatically engaged and the present heading held	
References and performance	The LOC inbound (front) course can only be set and adjusted by the COURSE knob on the Remote Instrument Controller (RIC). At LOC capture, HDG mode automatically disengages and LOC mode automatically engages, so the helicopter turns to track the selected localizer. At GS capture, any selected collective mode automatically disengages and GS mode automatically engages, so the helicopter adjusts collective to hold the glide slope. Upon completion of ILS Approach, ALVL mode is automatically engaged.	



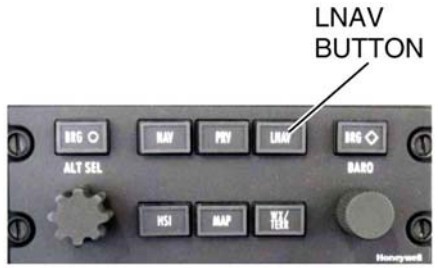
ICN-39-A-220000-G-A0126-00176-A-01-1

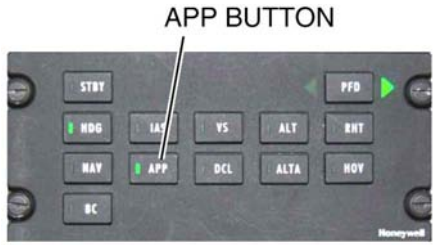
ILS PATTERN



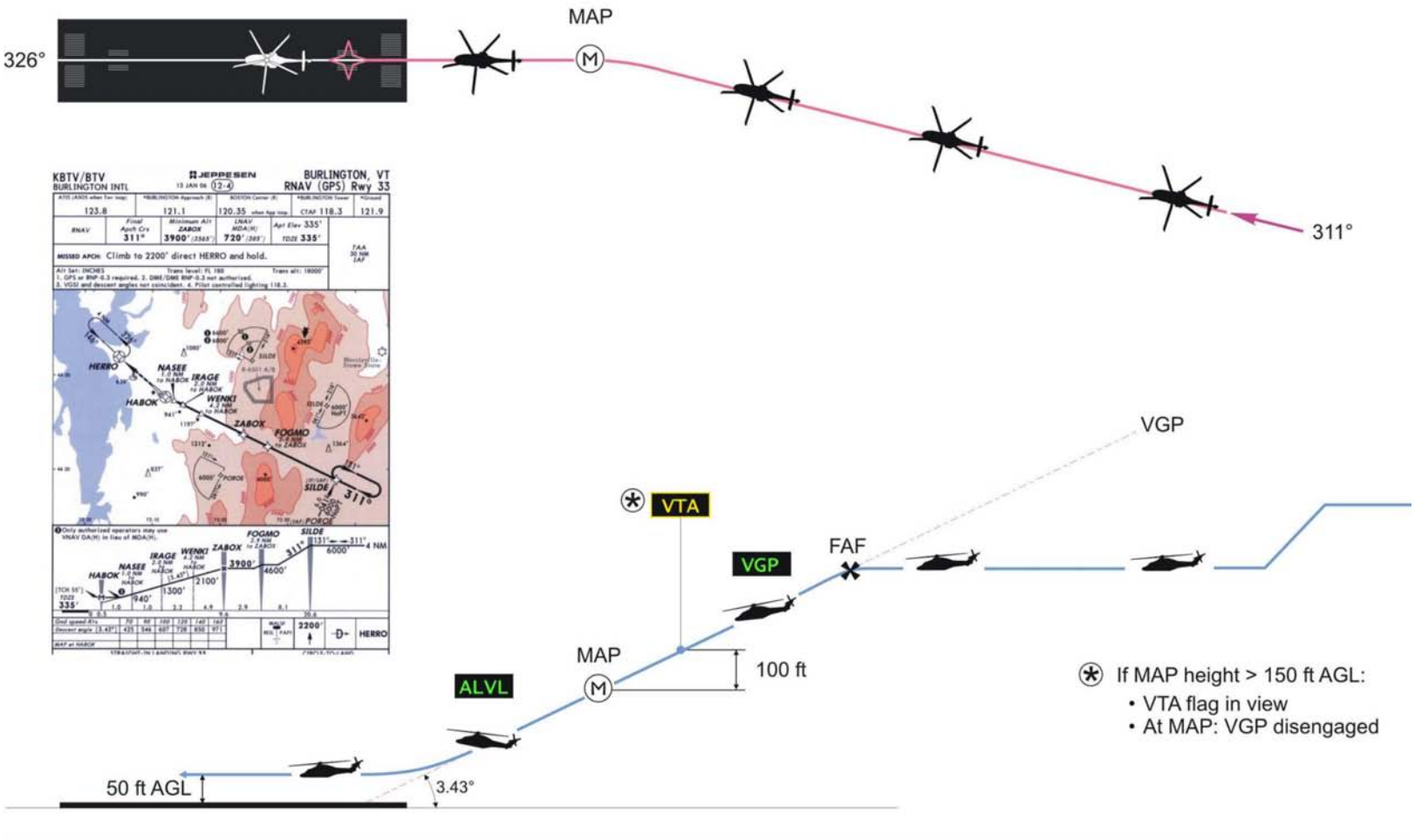


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APP MODE — VGP APPROACH FUNCTION (GPS/RNAV APPROACH)		
Description	The APP mode produces a non precision approach similar to an ILS approach when using a non localizer based approach from the data base with FMS selected as the Primary Navaid (LNAV button on DC). The Vertical Glide Path (VGP) is anchored at the BOD (Bottom of Descent) —ie the touchdown point— with an angle equal to that specified in the data base at the corresponding waypoint. Aircraft axes control is performed via: • roll attitude for LNAV lateral steering • collective input for VGP	

APP MODE — VGP APPROACH FUNCTION (GPS/RNAV APPROACH)		
Engagement	• Arming:..... Press APP button on Guidance Controller • VGP mode is armed • HDG mode is automatically engaged and the present heading held Conditions for arming VGP mode: • FMS is selected as Primary Navaid and a published non-Localizer approach is selected as Arrival in the Active FPL • FMS is not in DR (Dead Reckoning) mode • Helicopter is within 30 nm of destination • Altitude and angle constraint values have not been changed • If there is an altitude constraint at the FAF (Final Approach Fix), the helicopter must be at the FAF altitude • No vertical direct-to the FAF has been executed NOTE O If APP pushbutton is pressed when Primary Navaid is FMS and VGP is not available, the VGP UNAVAILABLE message is displayed in the MCDU scratch-pad.	

APP MODE — VGP APPROACH FUNCTION (GPS/RNAV APPROACH)		
References and performance	On the glide slope pointer a letter P is displayed to indicate VGP. When VGP mode is armed, both letter P and pointer appear on the left hand side of vertical deviation scale. When captured the symbol moves to the right hand side of the vertical deviation scale. When flying to the MAP (Missed Approach Point) or within 5 nm from the FAF, any selected collective mode automatically disengages and the VGP mode is automatically captured, so the helicopter adjusts collective to hold the VGP. Conditions for capturing VGP mode: • VGP mode must be armed • FMS active waypoint is the MAP or helicopter is within 5 nm of the FAF • If holding, helicopter must be established on the inbound course to the FAF • Helicopter is able to capture the final approach slope As the helicopter approaches the runway threshold, ALVL mode is automatically engaged and collective is commanded for an asymptotic flare. If MAP height (in the Navigation Data Base) is greater than 150 ft AGL: • when helicopter is 100 ft above the MAP the VTA (Vertical Track Alert) flag appears on the ADI • when helicopter is at the MAP the VGP mode disengages	



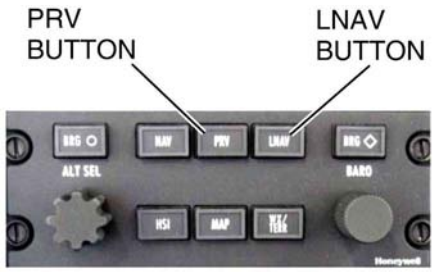
ICN-39-A-220000-G-A0126-00181-A-01-1

VGP PATTERN



VGP CAPTURED

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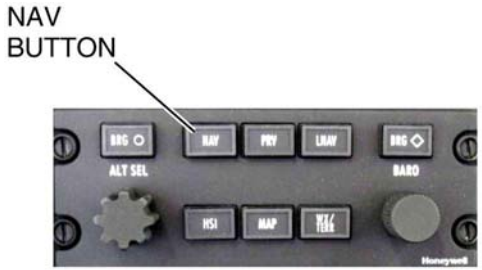
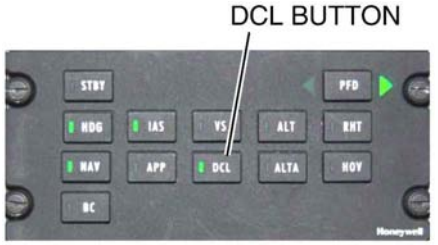
APP MODE — PREVIEW FUNCTION		
Description	The Preview function is a way to preselect ILS or VOR Approach functions while FD is still engaged in LNAV. The Preview function is selected by pressing the PRV pushbutton on the Display Controller with FMS already selected as the Primary Navaid (LNAV button on DC).	
Engagement	• Arming:..... Press APP button on Guidance Controller. • GS and LOC modes are armed • Helicopter keeps flying the LNAV leg	

APP MODE — PREVIEW FUNCTION		
References and performance	Selected Preview Course is displayed on the HSI as an additional thin cyan CDI and a small-font digital readout. Preview VOR/LOC Course can be set and adjusted by the COURSE knob on the Remote Instrument Controller (RIC). Preview VOR Course can also be synchronized with the current VOR bearing value by pressing the PUSH DIR button on COURSE knob on the RIC. If the APP pushbutton is pressed the FD arms LOC and GS or VAPP functions depending on the NAV receiver tuning (ILS or VOR). At the capture of the LOC or VAPP functions the FD disengages LNAV and engages LOC-GS or VAPP modes.	

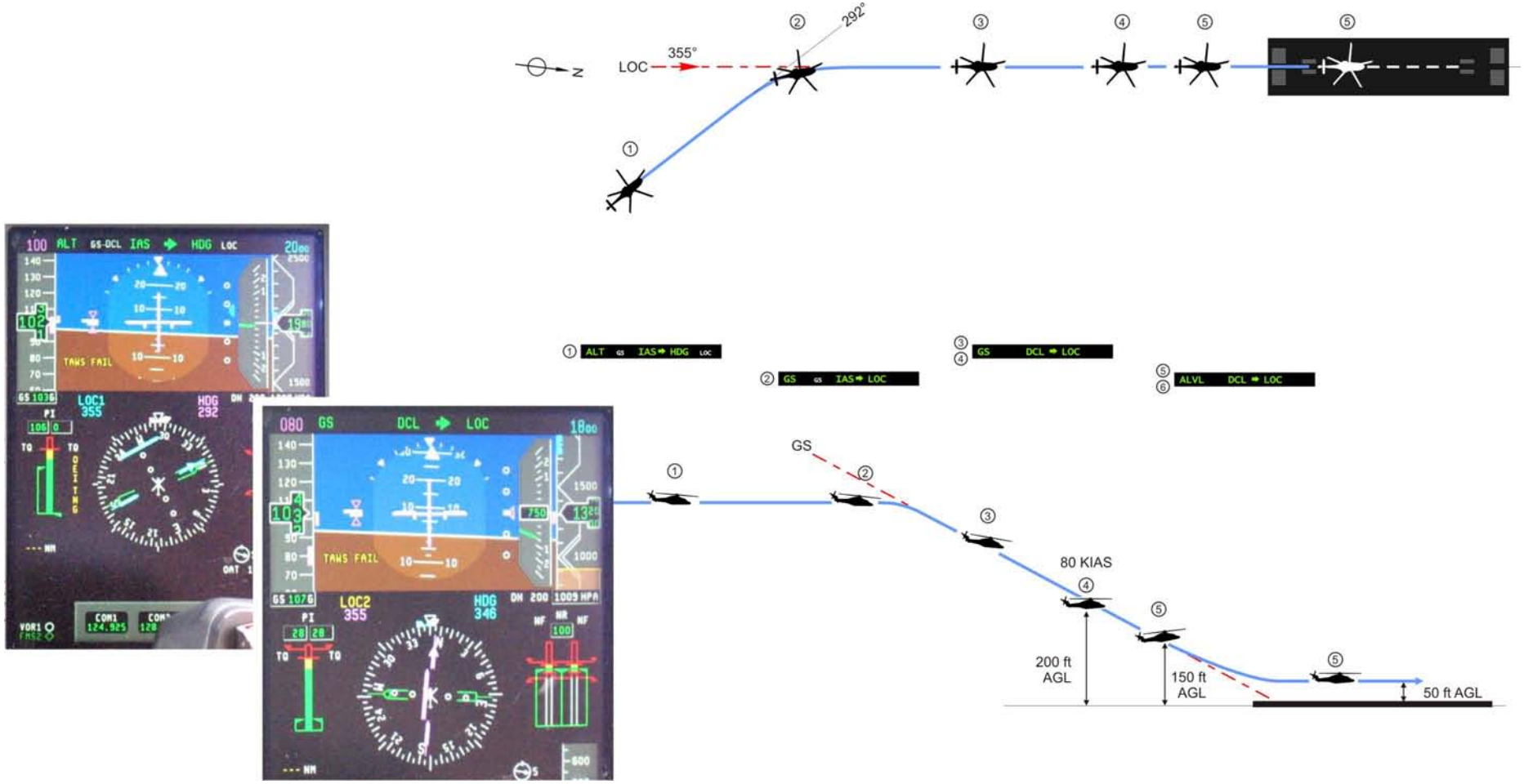




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DCL — ILS DECELERATION MODE		
Description	The DCL ILS mode is the same as the APP mode plus automatic deceleration from present speed down to approximately 80 KIAS as the aircraft reaches 200 ft AGL. LOC must be selected by the pilot in command as the Primary Navaid via the NAV button on the Display Controller (DC) and displayed as the CDI and the associated digital readout on the HSI. Aircraft axes control is performed via: • roll attitude for LOC • collective input for GS • pitch attitude for airspeed	
Engagement	• Arming:..... Press DCL button on Guidance Controller. • GS-DCL and LOC modes are armed • HDG mode is automatically engaged and the present heading held • IAS mode is automatically engaged and the present airspeed held	

DCL — ILS DECELERATION MODE		
References and performance	The LOC inbound (front) course can only be set and adjusted by the COURSE knob on the Remote Instrument Controller (RIC). At LOC capture, HDG mode automatically disengages and LOC mode automatically engages, so the helicopter turns to track the selected localizer. At GS capture, any selected collective mode automatically disengages and GS mode automatically engages, so the helicopter adjusts collective to hold the glide slope.	
	After GS capture and at a computed condition for correct approach and deceleration the IAS mode is disengaged and the DCL mode engaged. The airspeed reference bug moves to 80 KIAS and the digital reference is set to 80 KIAS. These values can be adjusted using the cyclic beep trim at which time the IAS mode automatically engages replacing the DCL mode. Upon completion of ILS Approach, ALVL mode is automatically engaged.	



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BACK COURSE (BC) MODE

The BC mode allows the aircraft to capture and track the inbound back course of the localizer for approach to a runway by providing the lateral guidance in the roll axis.
After tuning, the course pointer on the PFD must be set to the runway inbound course, with the heading bug set to the necessary intersection heading. When the BC button is pressed on the GC, HDG mode is automatically engaged and BC armed. The BC mode operates in the same way as the LOC mode with the deviation and course signals that are opposite to make a BC approach to the localizer.

GA — GO AROUND MODE

The Go-Around mode (GA) provides aircraft control for an automatic ascent profile. The function is intended primarily for use during the approach phase of flight to initiate a missed approach.
GA mode is engaged by pressing he GA pushbutton on either collective; any other mode is then disengaged.
Upon engagement the GA mode commands collective for a 1,000 ft/min climb, roll for wings-level and pitch for an airspeed of 80 KIAS or the current airspeed, whichever is higher.

OS — OVER STATION MODE

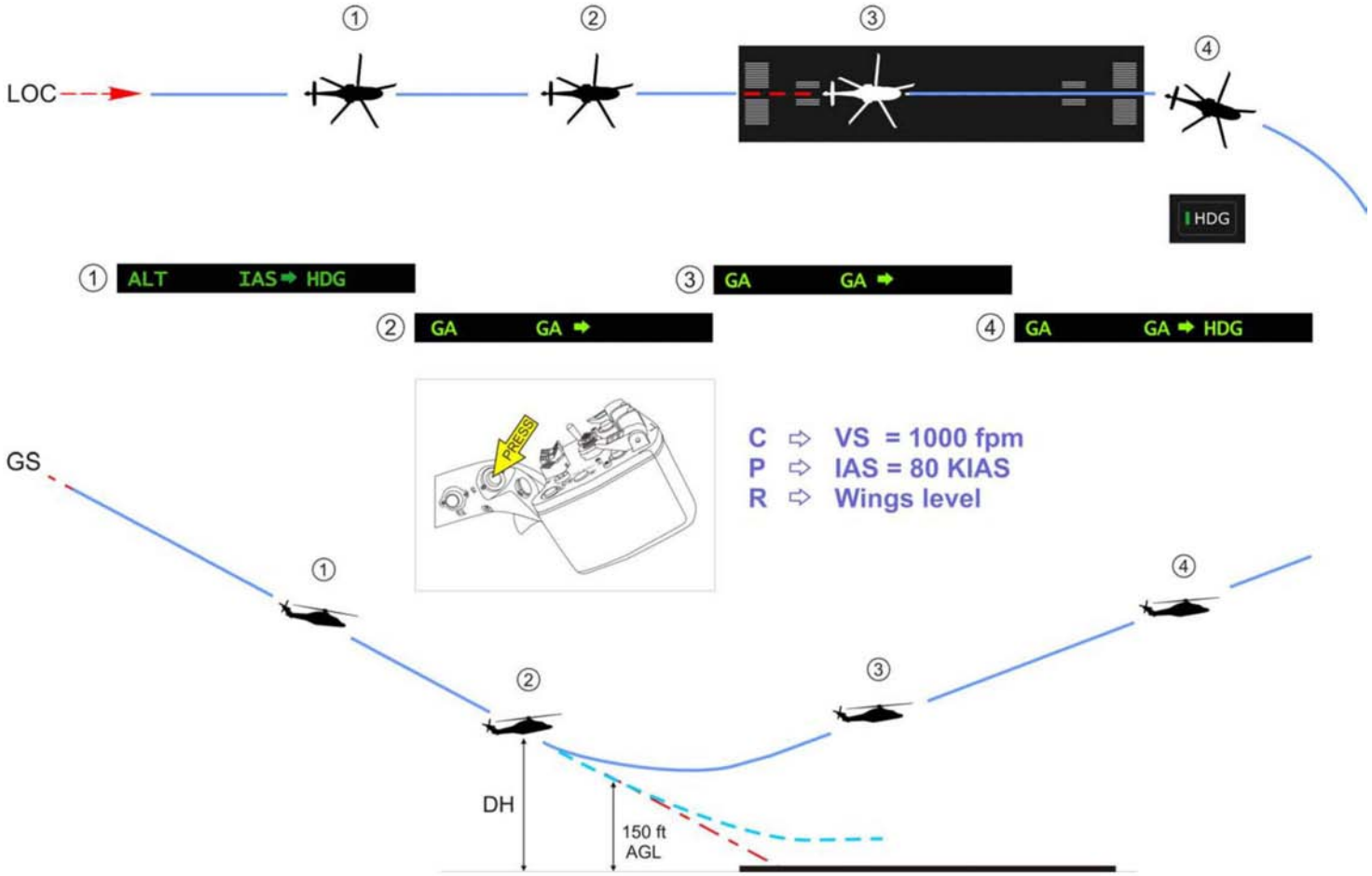
When overflying a VOR station, with VOR or VAPP mode engaged, the VOR equipment detects the aircraft being within the zone of ambiguity.
The VOR deviation signal is removed from the command until the VOR signal has stabilized after passing over station, when the OS mode disengages returning to VOR or VAPP mode automatically.
While over the station, course change may be made by selecting a new course via the COURSE knob on the RIC: aircraft is steered heading towards the selected course until OS mode is replaced by VOR or VAPP.

ALVL — AUTOLEVEL MODE

If the APP (LOC/GS or VGP) mode is continued, to below landing minimum, this safety feature is automatically activated at 150 ft AGL and an asymptotic flare to a radar height of 50 ft AGL will be initiated.
The green GS or VGP caption on the PFD will be replaced by ALVL.

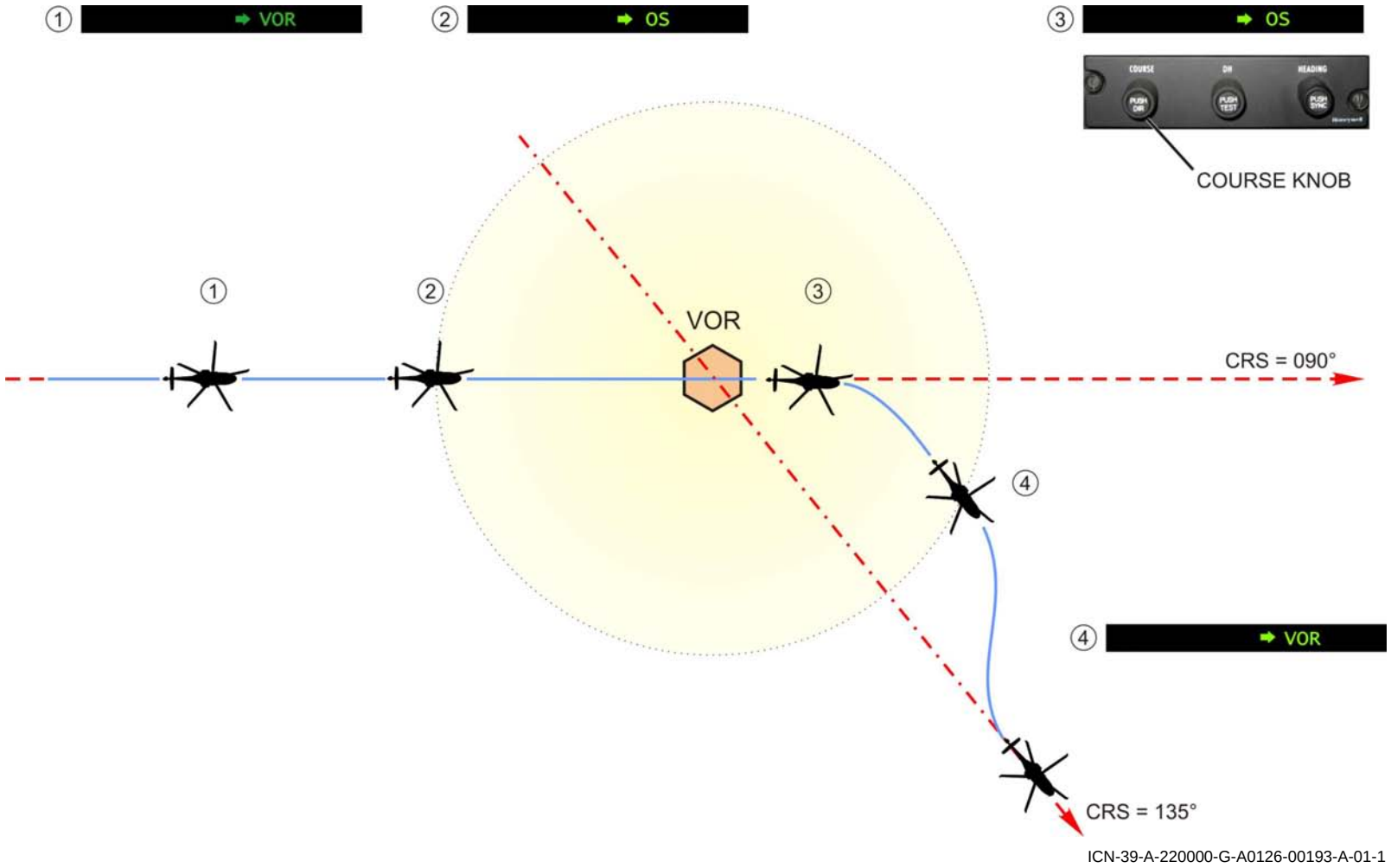
CAUTION

IN VGP MODE THE ALVL WILL NOT ACTIVATE IF THE MISSED APPROACH POINT (MAP) IS HIGHER THAN 150 FT AGL. FOR THIS CASE THE VGP MODE WILL DISENGAGE AT THE MAP (PRECEDED BY A VERTICAL TRACK ALERT (VTA) CAPTION DISPLAYED ABOVE THE VERTICAL GUIDANCE SCALE ON PFD) AND A CHIME IS GENERATED.



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GO-AROUND MODE



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OVER STATION MODE

HOV — HOVER/VELOCITY HOLD MODE

The HOV mode utilizes the blended AHRS-GPS ground velocity information to provide commands that maintain longitudinal and lateral aircraft ground velocities for hovering and low speed flying.
The HOV can be manually engaged by pressing the HOV push button on the Guidance Controller or by pressing the beep trim (Chinese-Hat fifth position) on the cyclic hand-grip.
At the engagement the HOV mode velocity references are set to zero. The engagement of HOV mode will automatically engage RHT mode if a valid radar altimeter signal is available and within the threshold limits.
The HOV mode velocity references may be changed by the pilot using any of the following means:

- pressing the cyclic FTR switch, flying to the desired lateral and longitudinal velocities and then releasing the cyclic FTR switch
- pressing the cyclic beep switch forward, aft, right, or left to increase the velocity reference in the direction of the switch press
- pressing the cyclic beep fifth position switch to promptly zero the velocity reference

The HOV mode velocity references are limited to a maximum of 60 knots forward and 40 knots left, right, and aft (while aft the velocity reference is limited to 40 knots total vector amplitude).

CAUTION

IN ALT MODE THE VOICE MESSAGE “ALTITUDE ALTITUDE” IS TRIGGERED WHEN ALTITUDE EXCEEDS THE REFERENCE ALTITUDE BY ± 150 FT. THEREFORE, IF ALT MODE IS ENGAGED AS AN ALTERNATIVE TO RHT AT A HEIGHT BELOW 300 FT, SET DH AT A VALUE 10 FT BELOW THE REFERENCE HEIGHT IN ORDER TO HAVE AN ADDITIONAL HEIGHT DEVIATION EXCEEDANCES CUE.

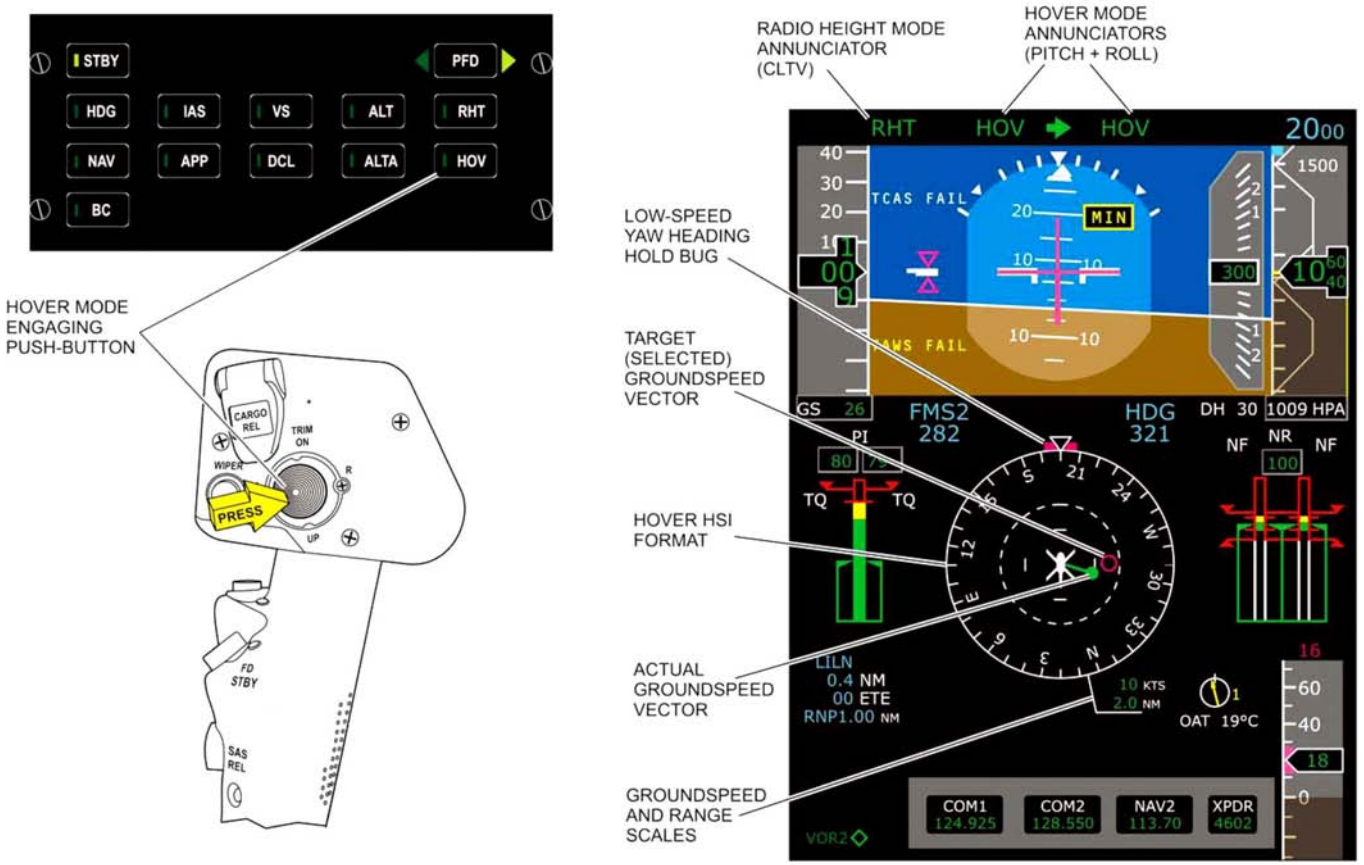
CAUTION

THE HOV MODE MAINTAINS A GROUNDSPEED REFERENCE THEREFORE PILOT MUST ENSURE THAT CROSSWIND AND REAR WIND SPEED LIMITS ARE NOT EXCEEDED. IF WIND LIMITS ARE EXCEEDED DIRECTIONAL CONTROL MAY NOT BE MAINTAINED.

CAUTION

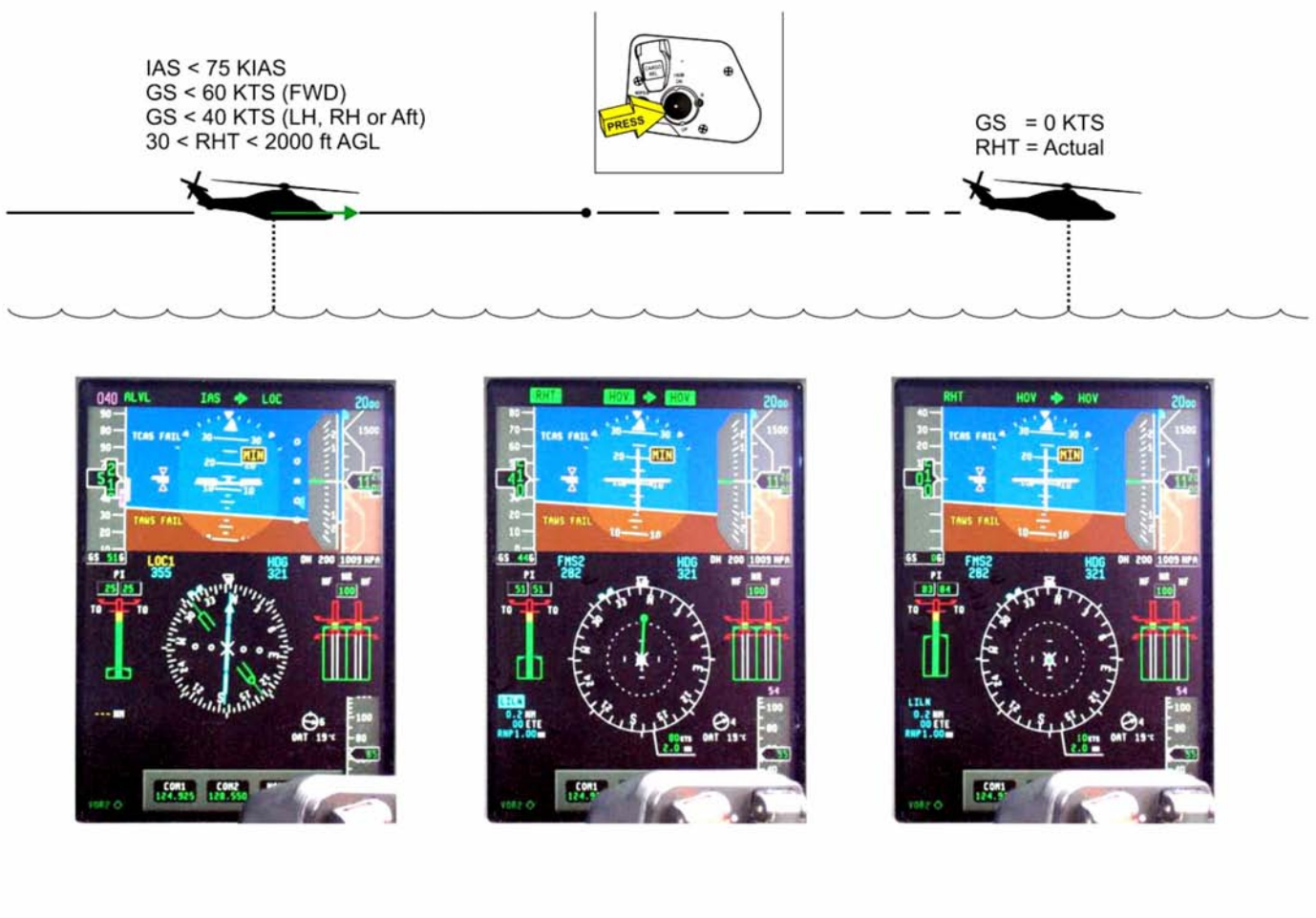
WHEN HOV MODE IS ENGAGED ABOVE 2000 FT AGL THE ALT MODE DOES NOT AUTOMATICALLY ENGAGE. THEREFORE THE PILOT MUST CONTROL COLLECTIVE MANUALLY OR ENGAGE ALT MODE.

The hover symbology automatically appears on the PFD when HOV mode is engaged. The display shows the aircraft velocity vector and the reference velocity symbol.



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HOVER MODE



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HOVER MODE OPERATION

<table>
<tr><th colspan="2">FLIGHT DIRECTOR AURAL MESSAGES</th></tr>
<tr><th>VOICE ONLY MESSAGES</th><th>DESCRIPTION</th></tr>
<tr><td>FD CHIME</td><td>Change state of any FD mode:
engaged to disengaged or viceversa
arm to captured
coupled to uncoupled or viceversa</td></tr>
<tr><td>BIP-BIP</td><td>Change in altitude or radar height reference datum (target altitude or height)</td></tr>
<tr><td>ALTITUDE - ALTITUDE</td><td>Aircraft exceeded the altitude deviation, dependent on the selected mode, by:
ALT ± 150 ft from reference
RHT
<table><tr><th>Reference Height (ft)</th><th>Deviation (ft)</th></tr><tr><td>30</td><td>11</td></tr><tr><td>50</td><td>15</td></tr><tr><td>100</td><td>20</td></tr><tr><td>150</td><td>25</td></tr><tr><td>200</td><td>30</td></tr><tr><td>250</td><td>40</td></tr><tr><td>500</td><td>70</td></tr><tr><td>1000</td><td>135</td></tr><tr><td>1500</td><td>200</td></tr><tr><td>2000</td><td>260</td></tr></table></td></tr>
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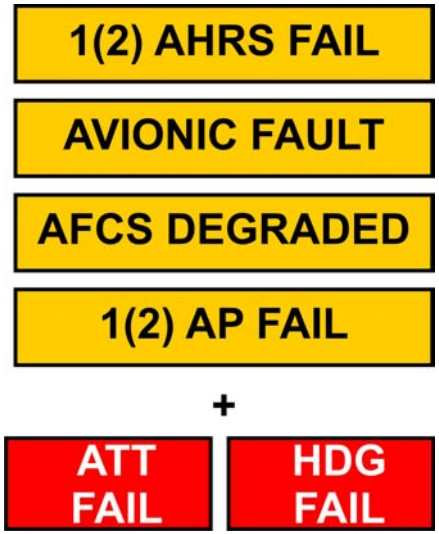
FLIGHT DIRECTOR FAILURE – CAS WARNING AND CAUTION MESSAGES

No CAS Warning or Caution messages are generated to indicate FD malfunctions. However indication on both PFD displays indicate the state of the flight director system.

GUIDANCE CONTROLLER FAILURE

In case of guidance controller failure, recognised as non functionality of panel pushbuttons (that is modes cannot be changed or disengaged using the panel pushbuttons), the FD may be disengaged using the cyclic FD STBY button.

CAS CAPTION	FAILURE DESCRIPTION	PROCEDURE NAME	AW139-RFM-4D
	replaces green arrow above attitude indicator on PFD that has failed Flight Director	SINGLE FLIGHT DIRECTOR FAILURE	Supplement 34 Section 3 EMERGENCY AND MAL-FUNCTION PROCEDURES
	replaces FD arrow above attitude indicator on both PFD displays	DUAL FLIGHT DIRECTOR FAIL	
	loss of PFD, the associated MFD will automatically configure to Composite mode maintaining the FD engaged on the same references as the PFD	FAILURE OF SELECTED PFD UNIT WITH FLIGHT DIRECTOR ENGAGED	

CAS CAPTION	FAILURE DESCRIPTION	PROCEDURE NAME	AW139-RFM-4D
	associated AHRS failure and subsequent loss of AP 1(2) + aural message and loss of attitude, heading and slip skid data on Left (Right) PFD	AHRS FAILURE WITH FD ENGAGED	

CAS CAPTION	FAILURE DESCRIPTION	PROCEDURE NAME	AW139-RFM-4D
1(2) ADS FAIL	NOTE P For on-side ADS failure on the non selected PFD there is no effect on FD modes. Refer to Sect. 3 Basic RFM + disengagement of FD vertical modes (ALT, IAS, VS) + Audio Chime and loss of  Airspeed Altitude Vertical speed Data on Left(Right) indicators	ON SIDE ADS FAILURE ON SELECTED PFD WITH FD ENGAGED	Supplement 40 Section 3 EMERGENCY AND MALFUNCTION PROCEDURES

CAS CAPTION	FAILURE DESCRIPTION	PROCEDURE NAME	AW139-RFM-4D
ATT OFF	+ SAS message displayed on both PFD's NOTE Q With ATT system not engaged the aircraft flies in SAS mode only	ATTITUDE SYSTEM OFF	
VTA	on attitude indicator during VGP approach when RAD ALT height between Missed Approach Point (MAP) and MAP+100 ft	VERTICAL TRACK ALERT CAPTION	
RAD Message replaces RAD ALT height information on both PFD RHT disengages with audio chime	Failure of both RAD ALT systems **CAUTION** WHEN BOTH RAD ALT FAIL, THE 150 FT AURAL WARNING MESSAGE DOES NOT FUNCTION AND THE LANDING GEAR CAUTION WILL BE DISPLAYED IF THE LDG GEAR IS RETRACTED REGARDLESS OF HEIGHT	DOUBLE RAD ALT FAILURE WITH RHT ENGAGED	

CAS CAPTION	FAILURE DESCRIPTION	PROCEDURE NAME	AW139-RFM-4D
RAD	message on RAD ALT display RHT disengages with audio chime miscompare between RAD ALT 1 and 2 information	RAD ALT MISCOMPARE WITH RHT ENGAGED	

LIMITATIONS

Refer to AW139-RFM-4D Supplement 34 SECTION 1 for the discussion of

- WEIGHT AND CENTRE OF GRAVITY LIMITATIONS
- FLIGHT DIRECTOR MODES LIMITATIONS
- FLIGHT DIRECTOR MODES ENGAGEMENT LIMITS AND MINIMUM USE HEIGHT (MUH)
- REQUIRED EQUIPMENT
- VOR LIMITATIONS
- ILS APPROACH MODE LIMITATIONS
- VGP LIMITATIONS

VELOCITY ANNUNCIATOR	
VEL FAIL	When the valid velocity data from both AHRS is not more available, the velocity vector on the hover display is removed and a red inverse video (white letter with a red background) VEL FAIL annunciator is displayed below and to the left of the compass rose. The HOV mode disengages automatically.
VEL TEST	When velocity is reported in test from an AHRS source, a red inverse video (white letters with a red background) VEL TEST annunciator is displayed below and to the left of the compass rose.

CHAPTER 23

COMMUNICATIONS

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SECTION 00 - COMMUNICATIONS

DESCRIPTION

Communications contains the systems necessary for communication in an aircraft, between different aircraft and between the aircraft and ground stations. These systems include components that supply voice and data communications.

Communications has the systems that follow:

- Modular Radio System
- Speech Communications
- Audio Integrating

Modular Radio System

The modular radio system is a multi-slot cabinet and network interface module.

Speech Communications

Speech communications use voice-modulated, electromagnetic waves to transmit and receive voice and data between different aircraft and between the aircraft and ground stations.

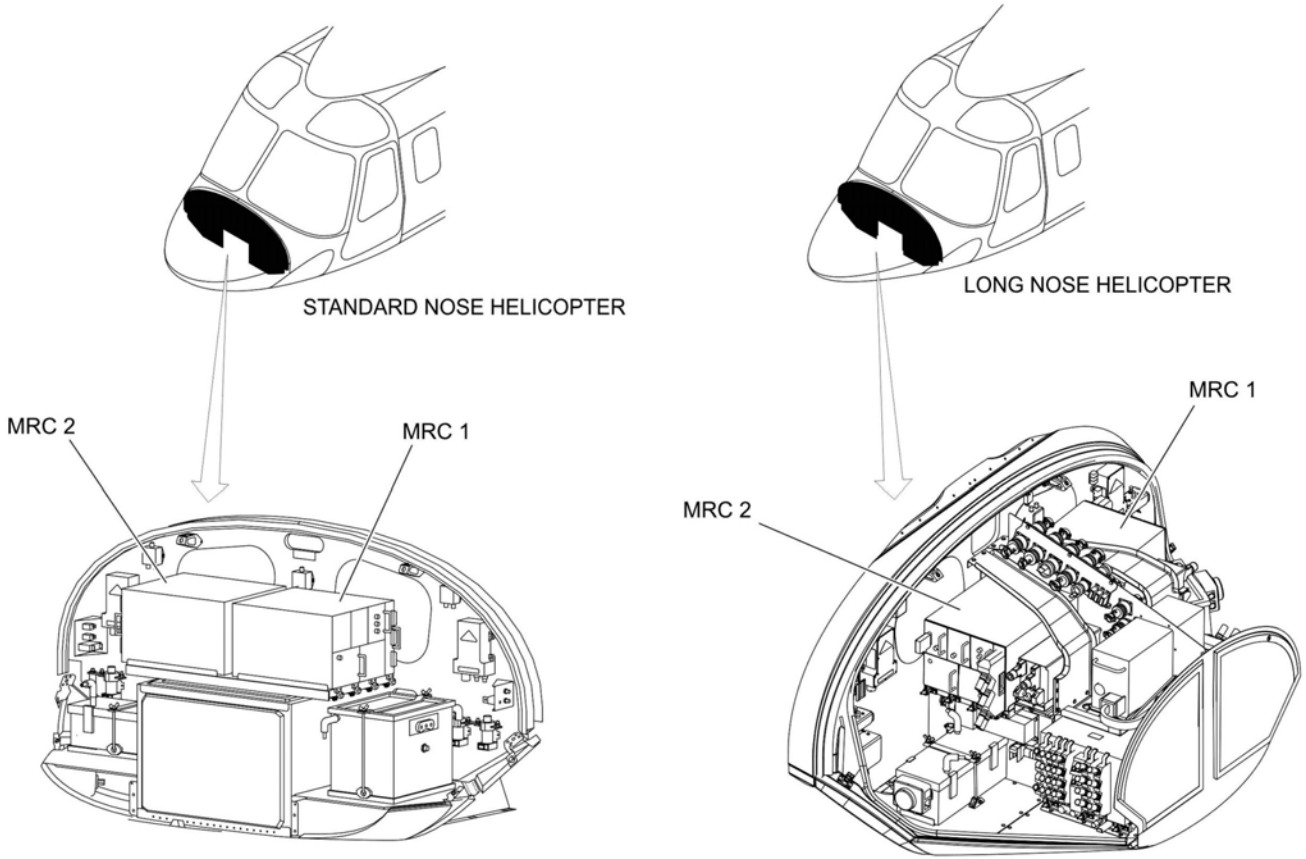
Audio Integrating

The audio integrating system controls the audio signals of the communication and navigation receivers/transmitters.
The Communication System is a part of the PRIMUS EPIC system.

MODULAR RADIO SYSTEM

The modular radio system is composed of two remote mounted Modular Radio Cabinets (MRC) that, with related controls, displays and antennas, supply aircraft radio functions, standard for navigation and communication systems, that include VOR, ILS, ADF, DME, VHF COM, and air traffic control (ATC) XPDR.

- MRC1 is installed in the nose bay left side.
- MRC2 is installed in the nose bay right side.

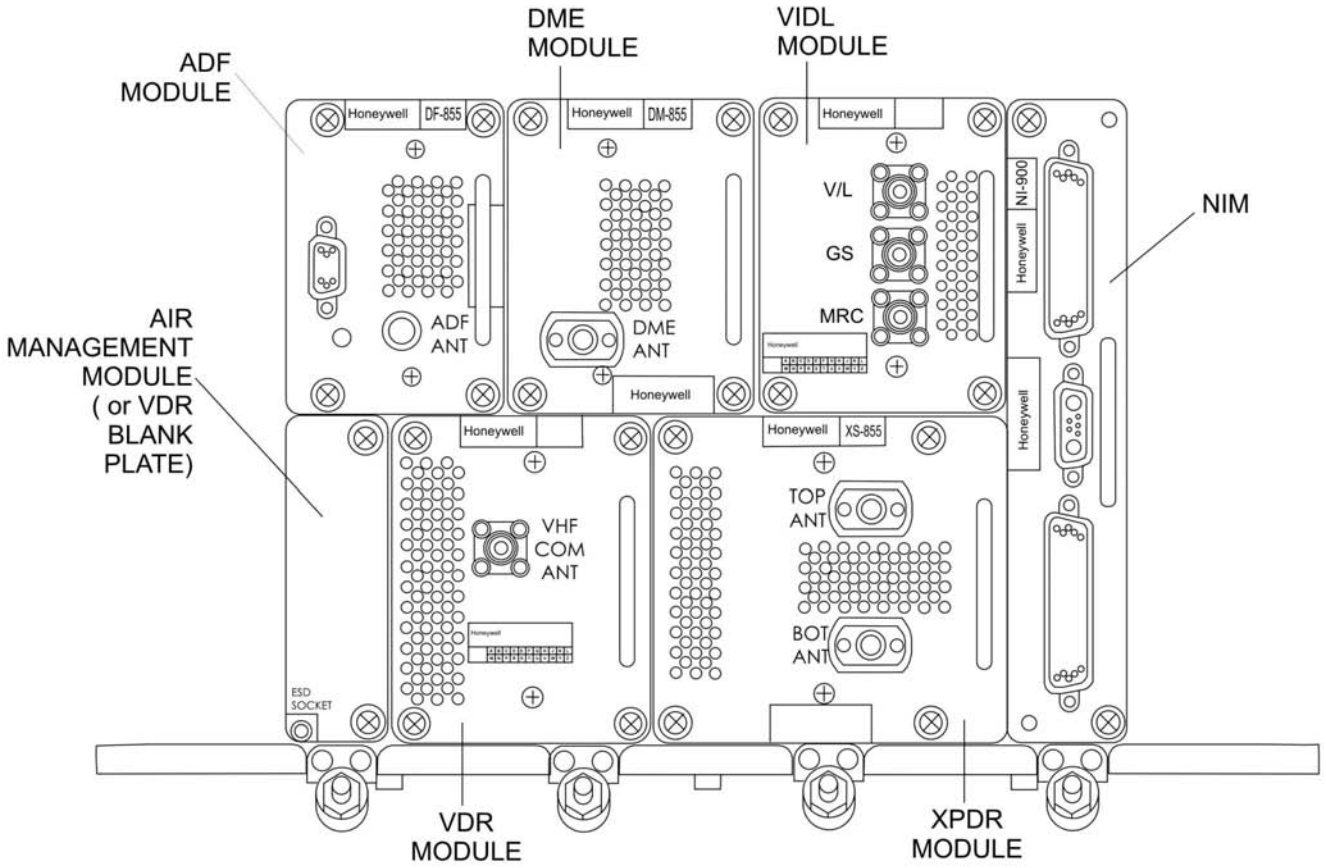


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MODULAR RADIO CABINET-LOCATIONS

The MRCs are line replaceable units (LRUs) and they contains line replaceable modules (LRMs).
The modules are as follows:

- VDR Module (COM)
- ADF Module
- DME Module
- VIDL Module (NAV)
- XPDR Module
- NIM Module



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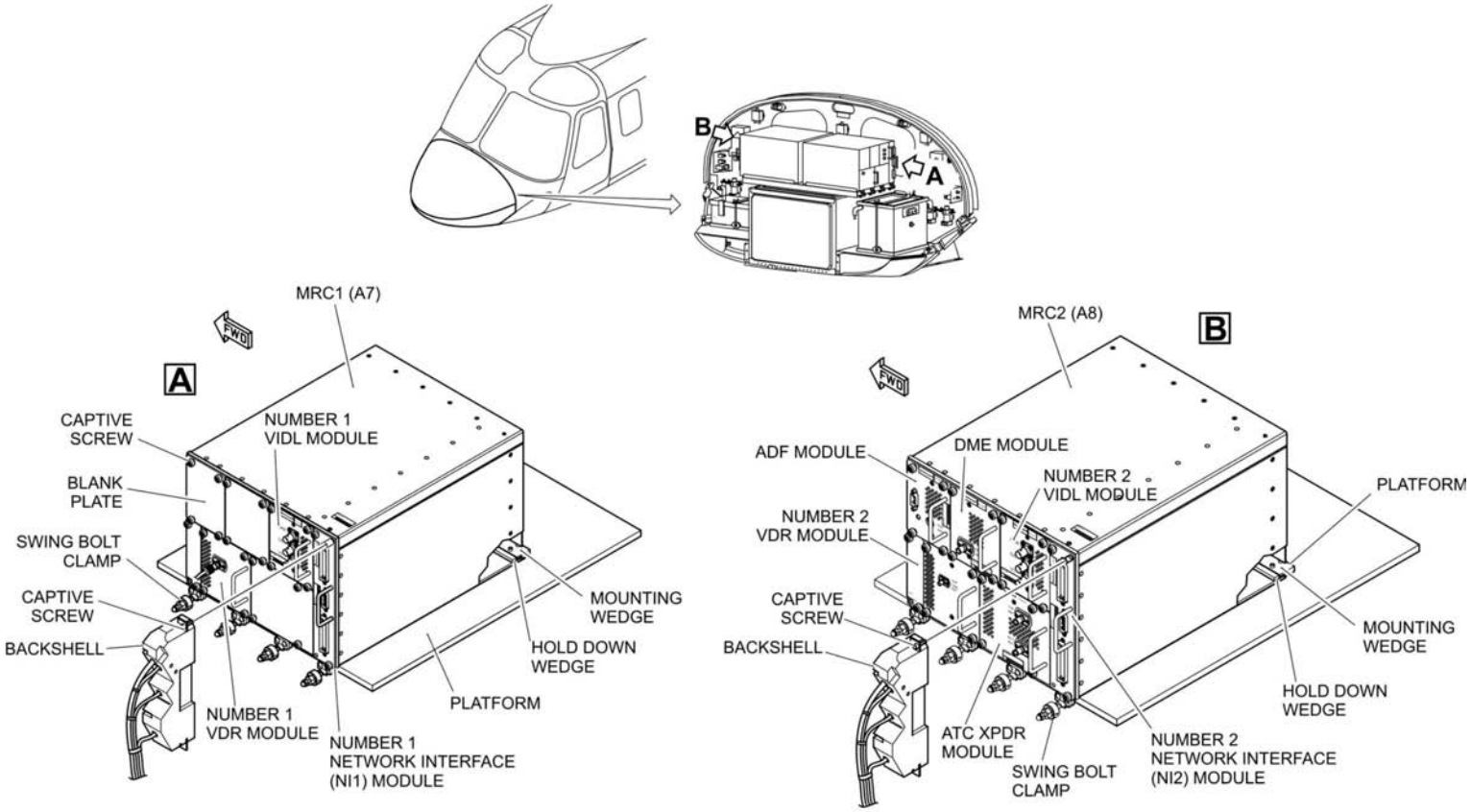
MODULAR RADIO CABINET – LOCATION OF MODULES

Each MRC contains a network interface module (NIM) and up to five radio modules.
Each module has a self-contained power supply, radio frequency (RF) receivers/transmitters; signal processing, backplane connectors, a front mounting plate, front connectors and circuit cards necessary to operate the module.
The radio modules connect to the NIM through the MRC backplane.
Two fans attached to the rear cover of the MRC supply cooling air to the NIM and radio modules.
Each MRC has an aircraft personality module (APM), on its backplane, APM 3 and 4 respectively in the MRC 1 and 2, for storage of setting and option files as part of the PRIMUS EPIC system. The APM data are stored in nonvolatile memory (NVM) and can be loaded with the Central Maintenance System (CMS).
MRC1 Standard configuration:

- VDR module..... Qty 1
- VIDL module..... Qty 1
- NIM module..... Qty 1

MRC2 Standard configuration:

- VDR module..... Qty 1
- VIDL module..... Qty 1
- ADF module..... Qty 1
- DME module..... Qty 1
- XPDR module..... Qty 1
- NIM module..... Qty 1

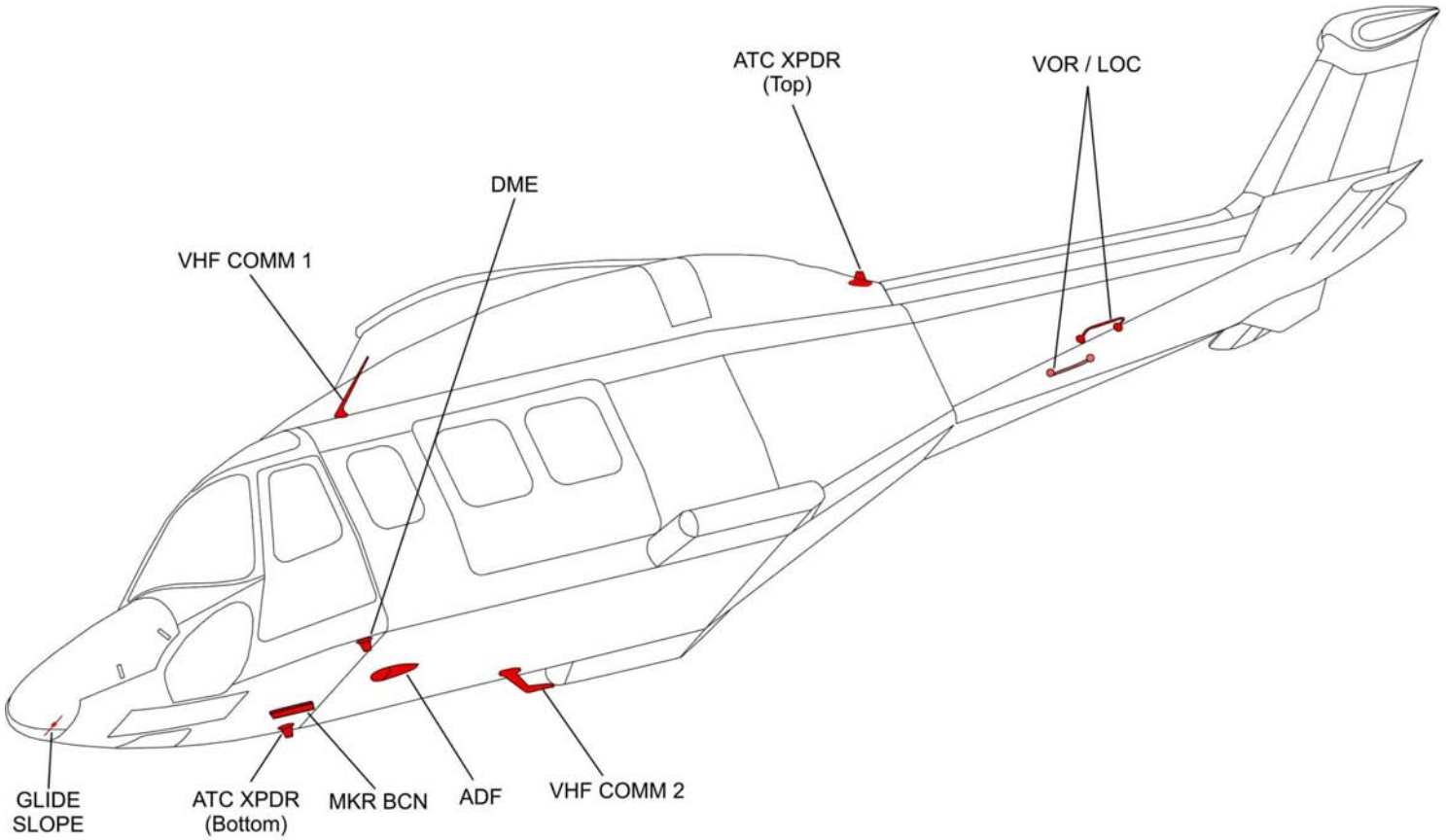


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MRC1 AND MRC2 STANDARD CONFIGURATION

ANTENNAS LOCATION

The radio modules are connected to the antennas.



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ANTENNAS LOCATION

NIM MODULE

The NIM supplies power to the radio modules and provides the following functions:

- radio data
- CMS functions
- radio tuning

RADIO DATA

Each NIM communicates to the rest of the PRIMUS EPIC system through the Avionics Standard Communication Bus version D (ASCB-D) and the Local Area Network (LAN).
Each radio module interfaces to the NIM through separate serial receive (RX) and transmit (TX) data buses called the RCB and internal LAN.
The NIM supplies the path for the discrete signals, analog audio signals that have been digitized and other signals necessary for the LRMs in the MRC.
The NIM also supplies analog to digital conversion of radio audio on the digital audio bus. The NIM has spare inputs for audio signals from external systems such as a third COM and High Frequency (HF) radios (optional equipment).
The digital audio bus from each MRC connects to each audio panel in the cockpit. This gives the pilot and copilot independent control of both sides of the audio.

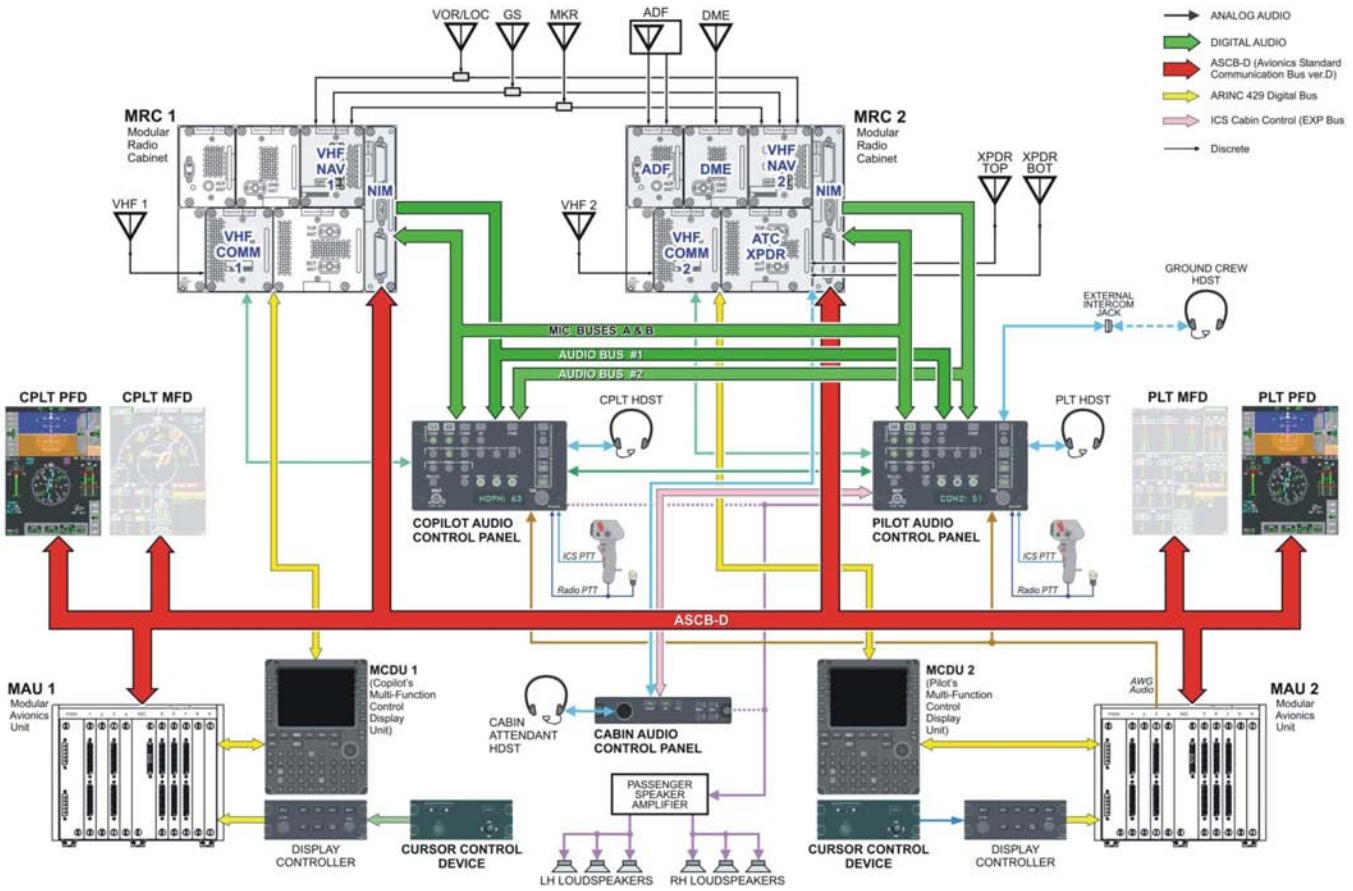
CMS FUNCTIONS

The NIM supplies a serial interface to APM installed on the MRC motherboard and uses the data from the APM to configure the radio modules in the MRC.
The NIM has a temperature sensor to monitor the MRC temperature through the RCB and controls the operation of the fans.

Another function of the NIM is to monitor the data that is transmitted across the digital buses and make sure that the buses operate correctly. If a malfunction is found, invalid signals are generated.

RADIO TUNING

Frequency, channel, and mode control of the modular radio system are supplied by Multifunction Control Display Units (MCDUs) and Curse Control Devices (CCDs).
The CCDs with the PFDs or the MCDUs can tune, control the modes, and show the displays for the modular radio system.
The MCDU is the primary tuning source for the modular radio system and the primary tuning is done through the ASCB-D.
There is also a backup MCDU tuning mode. This backup mode lets the pilot or copilot tune the on-side COM and NAV through a direct ARINC 429 link from the MCDU to the radios.



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MODULAR RADIO SYSTEM BLOCK DIAGRAM

SPEECH COMUNICATION

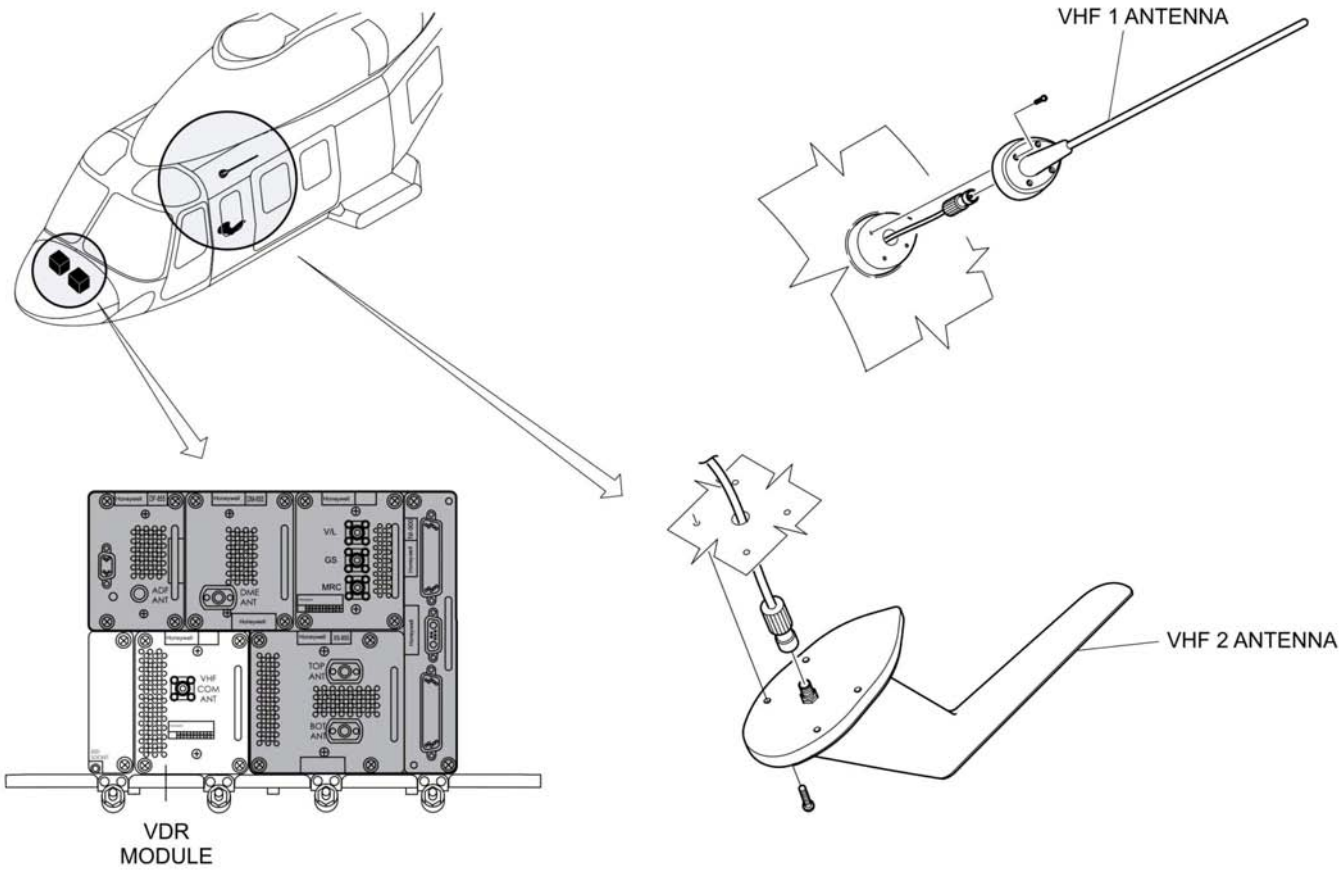
Speech communications has the system that follows:

- The Very High Frequency (VHF) Data Radio (VDR) system
- The V/UHF system (optional)
- The HF system (optional)

VDR SYSTEM

The very high frequency (VHF) data radio (VDR) system supplies communication (COM) between different aircraft, and between aircraft and ground stations. The VDR system uses amplitude modulation (AM) to transmit and receive voice signals for the cockpit headsets, but not at the same time (simplex operation).
The VDR system has the components that follow:

- VDR Module, Installed in Each MRC
- VHF COM Antennas.



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VDR SYSTEM-COMPONENTS LOCATION

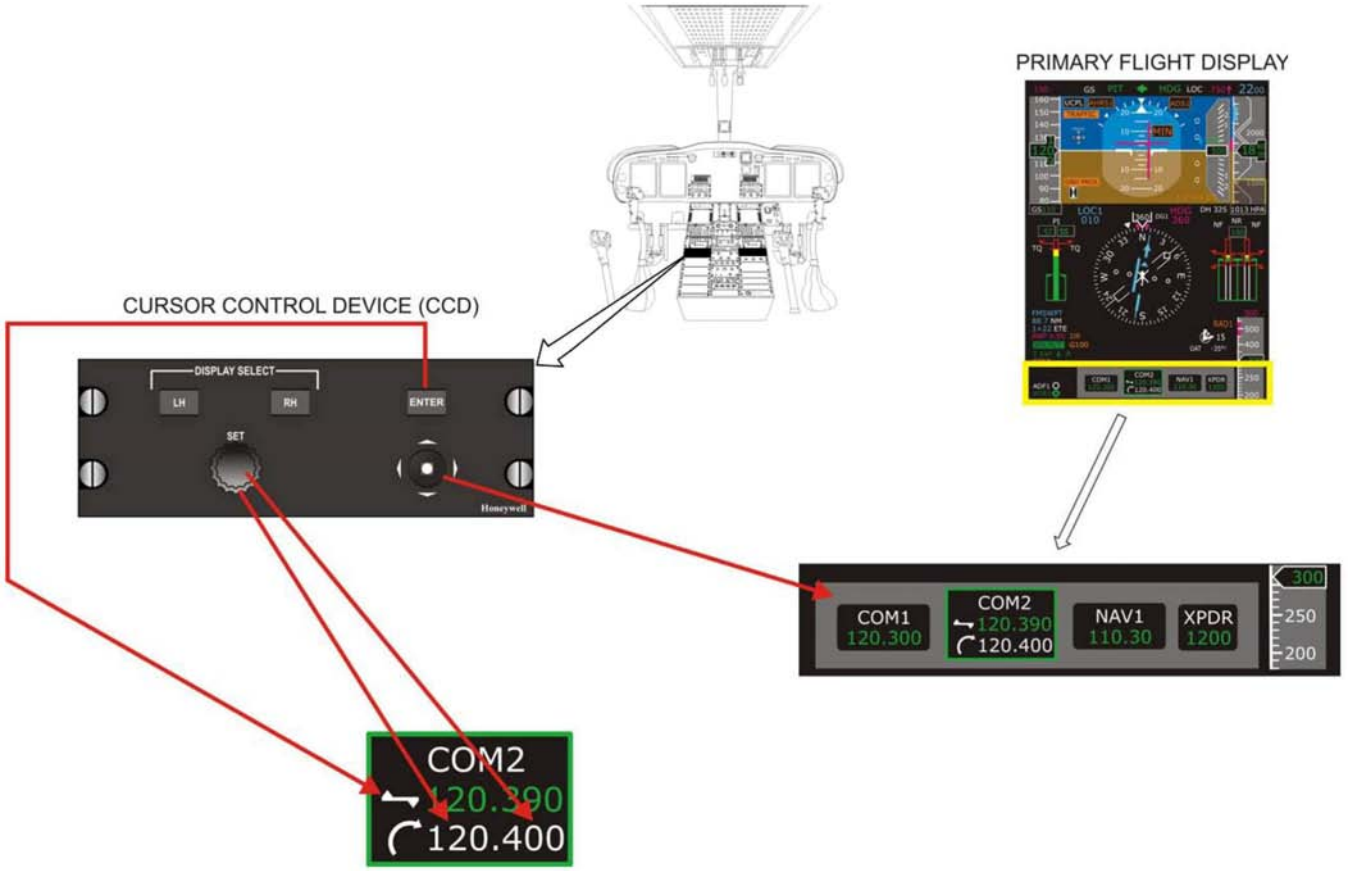
The VDR module is a VHF COM transceiver.
The VDR system operates in the frequency range from 118.000 to 136.975 MHz, with channel separations at 8.33 or 25 kHz.
The VDR system tunes in increments of 8.33 kHz. The VDR system only transmits data on 25-kHz channels.
Each VHF COM module receives frequency and squelch control signals through the NIM in the MRC.
The VDR system transmits and receives data through the radio control bus (RCB) to the network interface module (NIM). The NIM supplies the interface to the ASCB-D. The digital audio bus transmits the VDR audio signals to each audio panel in the audio system.
The transmitter has a timeout function that causes the VDR module to go back to the receive mode automatically after it transmits continuously for 2 min. This function helps to prevent interference or intermittent operation (COM) when more than one VDR system uses the same frequency.
The VDR module uses +28 V dc through NIM, but can be energized independently. The power on/off control signal goes directly through the NIM to the VDR module without change.
This low-power control signal controls the power supply of the VDR module.
The VDR module has other standard functions that include a nonvolatile maintenance log and an internal monitor circuit. The maintenance log records the possible data errors and makes them available to maintenance personnel. The internal monitor circuit monitors the circuit performance.

RADIO TUNING

The MCDU or the PFDs with cursor control devices (CCD) can tune, control the modes, and show the displays for the VDR system.
Primary tuning goes through the ASCB-D. Also, a backup MCDU tuning mode lets you tune the onside COM, NAV and XPDR modules through a direct ARINC 429 link from the MCDU to the radios.
Radio tuning information is displayed in the bottom center of the HSI.

Displaying tuning information on the PFD eases COM/NAV radio management. The currently tuned frequency for each radio is displayed on the PFD.
Using the CCD joystick and set knob, the pilot is able to tune the following on the PFD:

- COM1 radio
- COM2 radio
- NAV radio for the selected NAV source that is displayed on the PFD
- Transponder (XPDR) code



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RADIO TUNING USING CCD



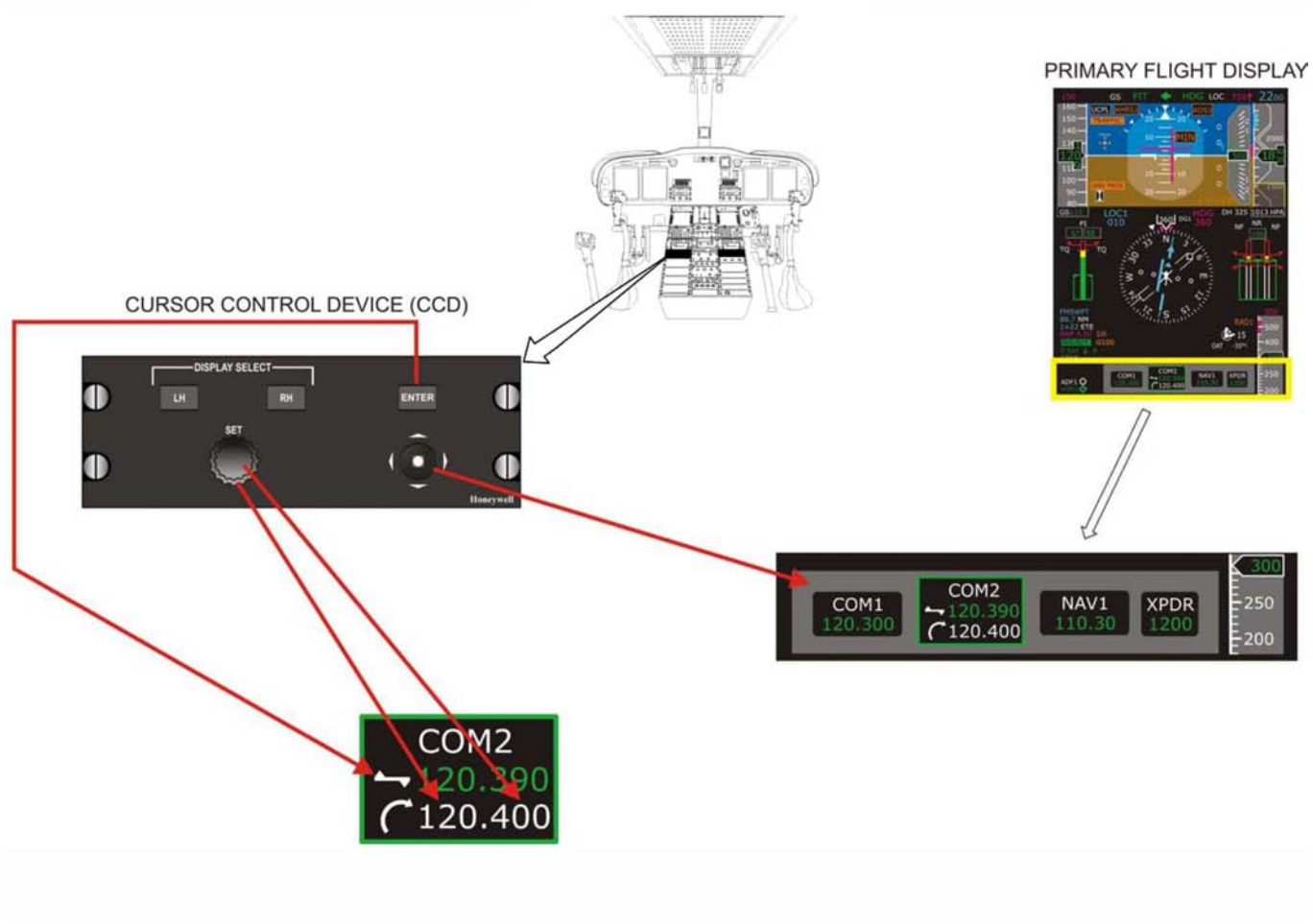
Green Cursor Box: The green cursor box is positioned to highlight the function using the CCD. Moving the joystick left or right shifts the cursor box between the COM and NAV radios.
When a frequency box has been made active to enable frequency changes, both the active and the preset (standby) frequencies are visible.



Set Knob Symbol: The set knob symbol prompts the pilot to use the SET knob on the CCD to change the frequency highlighted by the cursor.
Turning the larger knurled knob clockwise increases and counterclockwise decreases the whole number (Mhz) component of radio frequencies.
The smaller knob segment controls the decimal number (Khz) component of frequency ranges when selecting a radio frequency.



Swap Symbol: The swap symbol is visible next to the active frequency in the frequency box. It alerts the pilot that the active frequency can be replaced with the white positioned below it. This process of replacing one frequency with the other in the frequency box is referred to as swapping.
Pushing the ENTER button on the CCD swaps the current and preset frequencies for the radio.

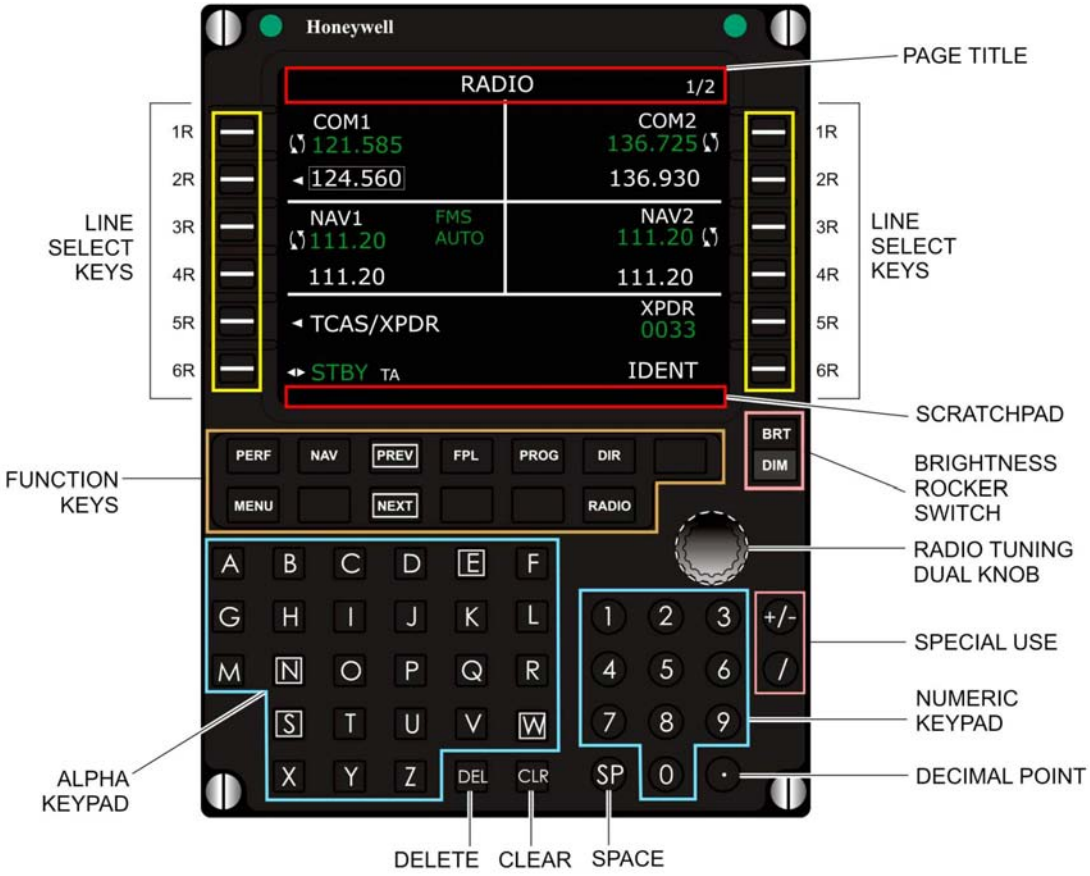


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RADIO TUNING USING CCD

MCDU

The MCDU is the primary control unit for the modular radio system. The MCDU controls all COM and NAV radios in the MRC. The MCDU controls the modes, frequencies and codes of all the modules in the MRC.
The Radio 1/2 page in the MCDU shows the related labels and frequencies for each VDR radio.
The top line shows the radio identification label (for example, COM 1 for VDR 1 and COM 2 for VDR 2).
The active frequency is in green.
The preset (standby) frequency is in white.



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MODULAR RADIO SYSTEM CONTROLS-MCDU



Cursor: The cursor box highlights the value in the active field. Pushing the adjacent line select key (LSK) of the other preset frequency changes position of the Cursor.



Tuning Curl: This icon indicates that the data value highlighted by the format cursor can be changed by turning the MCDU tuning knob.



Swap Frequencies: This symbol indicates exchanges between the active and preset frequencies for the associated radio that can be made.
This effectively saves the currently active frequency in the preset memory and tunes the radio to the frequency previously stored as the preset.

◀

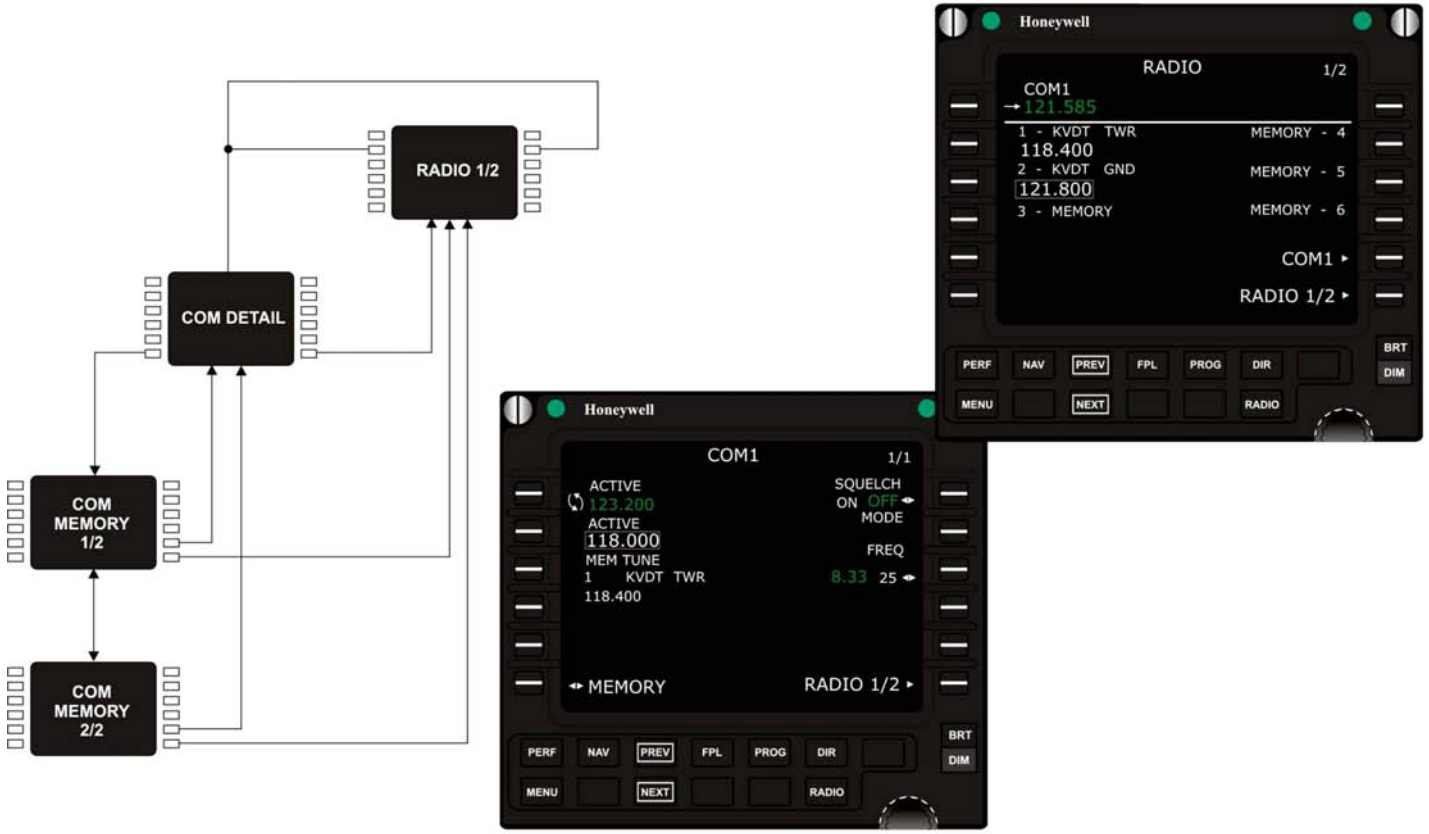
Page Indicator: When this icon is displayed, pushing the adjacent line select key (LSK) changes the display to another page. The page to be displayed is labeled explicitly or it is a detail page for the radio in the associated field.
The COM detail page is accessed by selecting the line key associated with the adjacent preselect boxed frequency in the main MCDU Radio page 1/2. It is used to access the controls specific to VHF communications radios, including squelch, operating mode, frequency spacing and a quick method for retrieving 12 frequencies from memory.
The COM detail page permits selection of 8.33 kHz or 25 kHz COM tuning and the selected spacing is green.
The COM pages on the MCDU also give access to the locations of the frequencies kept in memory.

◀▶

Exclusive Selection: This icon is displayed next to a list of mutually exclusive options. Each time the adjacent LSK is pushed, the next item in the list is selected, wrapping around to the first when the last option is reached. The selected value is displayed in green large font. The unselected items are displayed in small white characters.



Copy Value: This icon is used on the memory pages to indicate that the frequency highlighted by the cursor is copied into the active frequency for the associated radio.

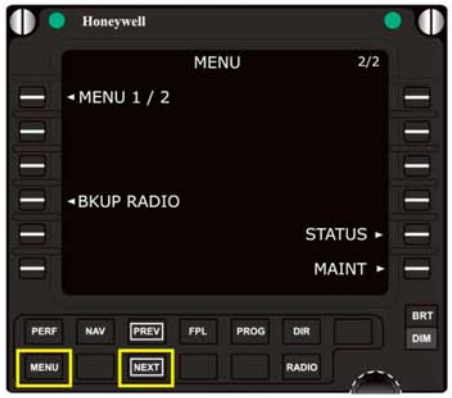


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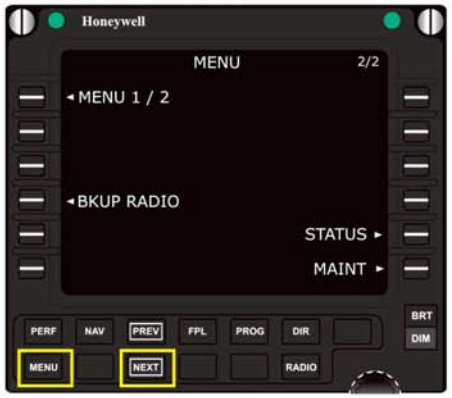
MCDU- VHF COM DETAIL AND MEMORY PAGES

The MCDU backup radio tuning page is given to permit direct tuning of the on-side COM, NAV and XPDR. This backup tuning mode is used when an MAU, or NIM fails or a loss of the ASCB. The backup tuning mode can be used for control of radios on the ground without applying power to the rest of the avionics system. The MCDU backup tuning page is displayed when the MCDU loses ARINC 429 communication with the on-side MAU.
In addition, the page is displayed when the BKUP RADIO menu button is selected on the MCDU menu page. The MCDU backup tuning page gives control of the on--side COM and NAV frequencies using the keyboard or the radio tuning knob. In addition, it controls the transponder code and operating mode for the on--side transponder. When operating the backup tuning mode, the altitude data from the ASCB is assumed to be unavailable and the available modes for selection are limited to standby and altitude off.

MCDU 1



MCDU 2



MCDU 1



MCDU 2



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BACKUP RADIO TUNING



RADIO 1 / 2 Annunciator Description

Annunciator	Description
25K	Indicates that the associated VHF COM radio is set to 25 Khz frequency spacing. When not present, the radio is tuning with 8.33 Khz frequency spacing. This is selected on the COM 1 / 2 page.
DME H xxx	This alert indicates that the VHF navigation radio is tuning the correpsonding DME receiver independently of the primary navigation frequency.
IHBT	This annunciator indicates that tuning of the radio is inhibited, usually from a remote source (such as an emergency tuning function).
IHBT	Indicates that the microphone button on the radio has been down long enough that the radio has identified it as stuck in the transmit position.
SQ	This annunciator indicates that the squelch feature for the radio is active. Squelch is set turned ON and OFF by the COM 1 / 2 page.
SQ	This annunciator indicates that the radio is currently transmitting.

Scratchpad Annunciator Description

ALERT	Description
DELETE	This annunciator indicates that the value in the scratchpad is deleted.
INVALID ENTRY	This annunciator indicates that the entry in the scratchpad is invalid.
SQNO RESPONSE FROM RADIO	This annunciator indicates that the squelch noise is coming from the radio.
STUCK MICROPHONE	This annunciator indicates that the microphone key is stuck.

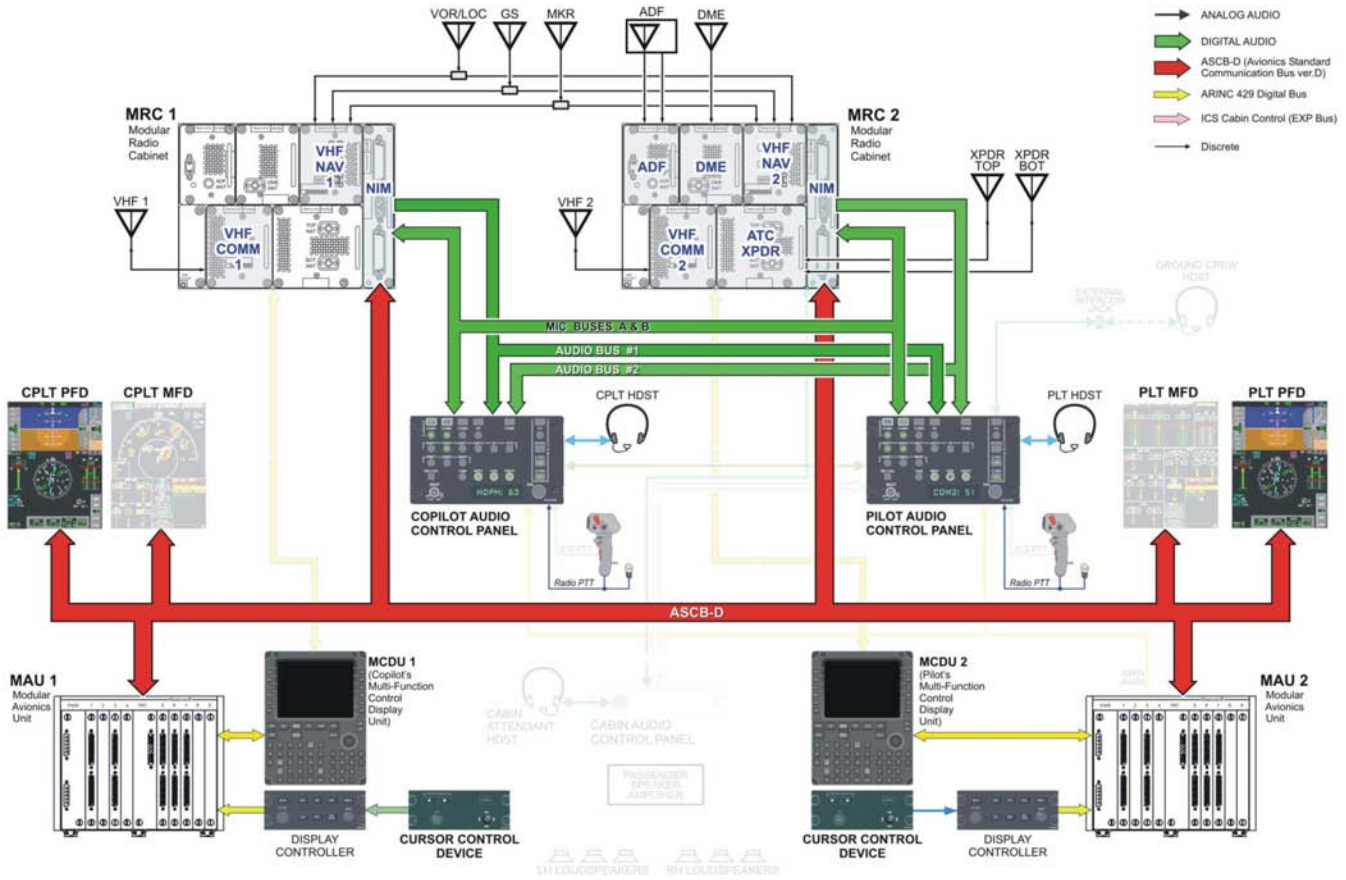
ICN-39-A-230000-G-A0126-00208-A-01-1



RADIO 2 / 2 Annunciator Description

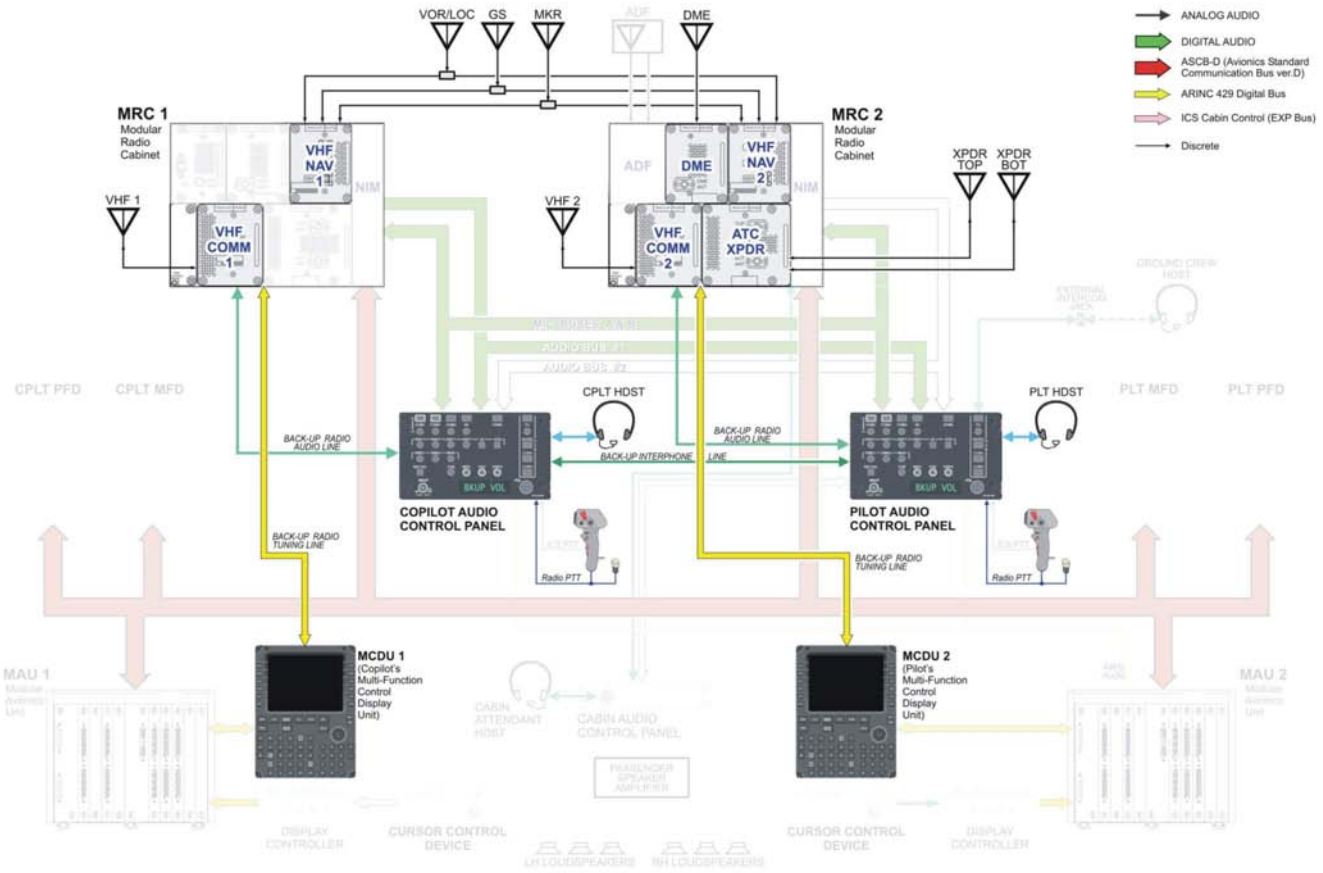
ALERT	Description
ANT	The ADF radio is in antenna mode.
BFO	The ADF radio is operating in BFO mode.
CW	This annunciator indicates that the radio is in a continouos wave mode.
IHBT	Indicates that tuning of the radio is inhibited, usually from a remote source (such as an emergency tuning function). Transmitting and/or receiving can be inhibited periodically when the other HF radio has recently performed a transmit operation.
ITU	This annunciator indicates that the radio frequency is an ITU frequency.
LO	The radio is set to low squelch.
MED	The radio is set to medium squelch.
RX	Indicates that the radio is currently receiving.
SQ	This annunciator indicates that the squelch feature for the radio is active.
TX	This annunciator indicates that the radio is currently transmitting.
TX MIN	The radio is transmitting with low power.
TX MED	The radio is transmitting with medium power.
VOICE	The ADF radio is in voice mode.

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MODULAR RADIO SYSTEM BLOCK DIAGRAM – NORMAL RADIO OPERATION



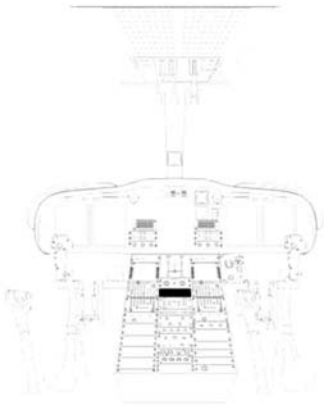
ICN-39-A-230000-G-A0126-00210-A-01-1

MODULAR RADIO SYSTEM BLOCK DIAGRAM –BACKUP RADIO

RADIO MASTER SWITCH

On the Miscellaneous panel there is the Radio Master switch that has two positions:

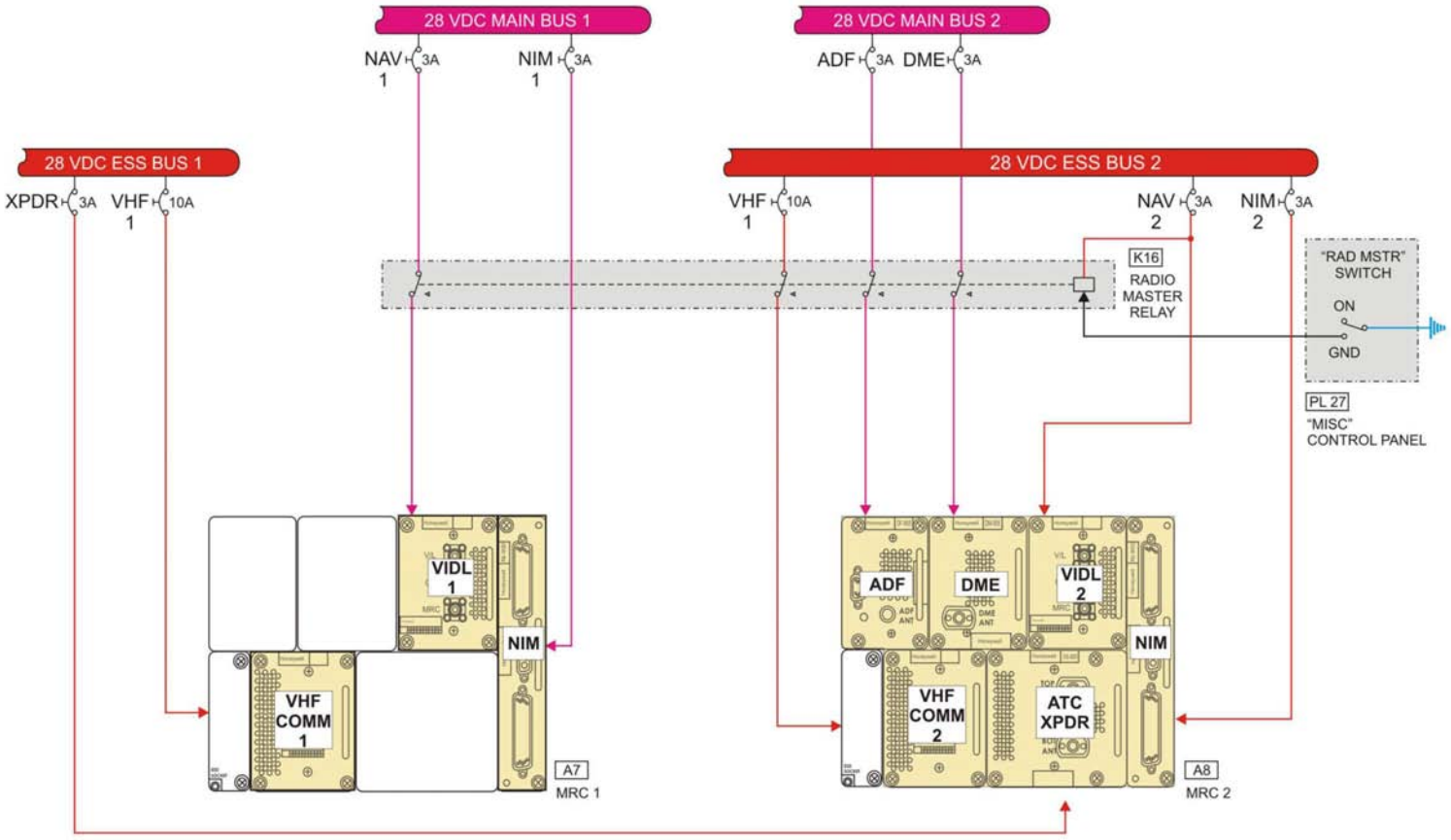
1	ON:	Normal Condition (All radios module are available)
2	GND:	Battery Start (COM1; NIM1; NAV2; ATC XPDR and NIM2 are available).





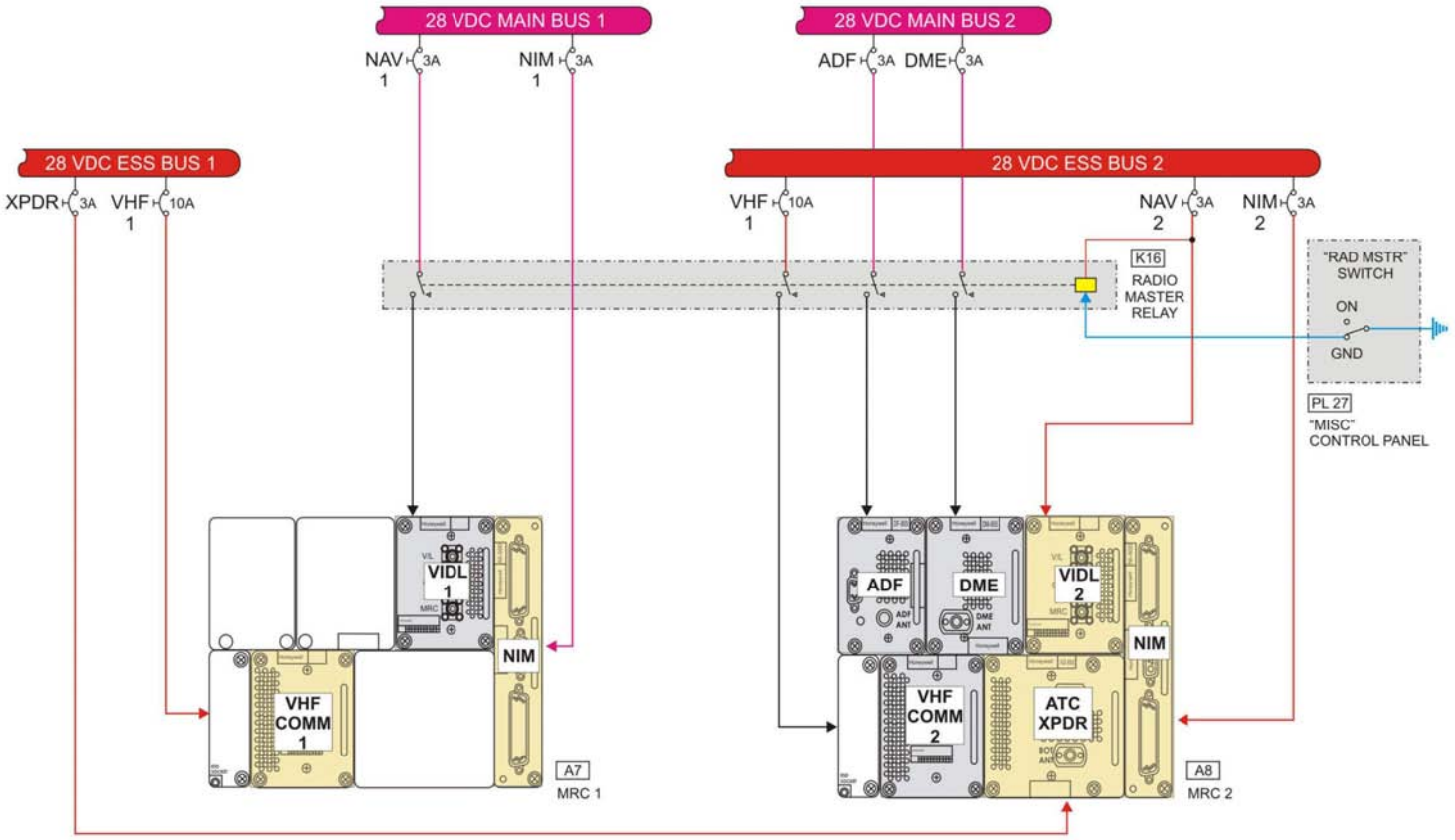
ICN-39-A-230000-G-A0126-02900-A-01-1

RADIO MASTER SWITCH



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RADIO MASTER SWITCH SCHEMATIC DIAGRAM (SHEET 1 OF 2)



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RADIO MASTER SWITCH SCHEMATIC DIAGRAM (SHEET 2 OF 2)

AUDIO INTEGRATING SYSTEM

The audio integrating system connects the communication (COM) and navigation (NAV) radios to the radio headsets and load speakers in the aircraft.
The audio integrating system includes the Airborne Audio System.
The airborne audio system is the central audio controller for the systems that follow:

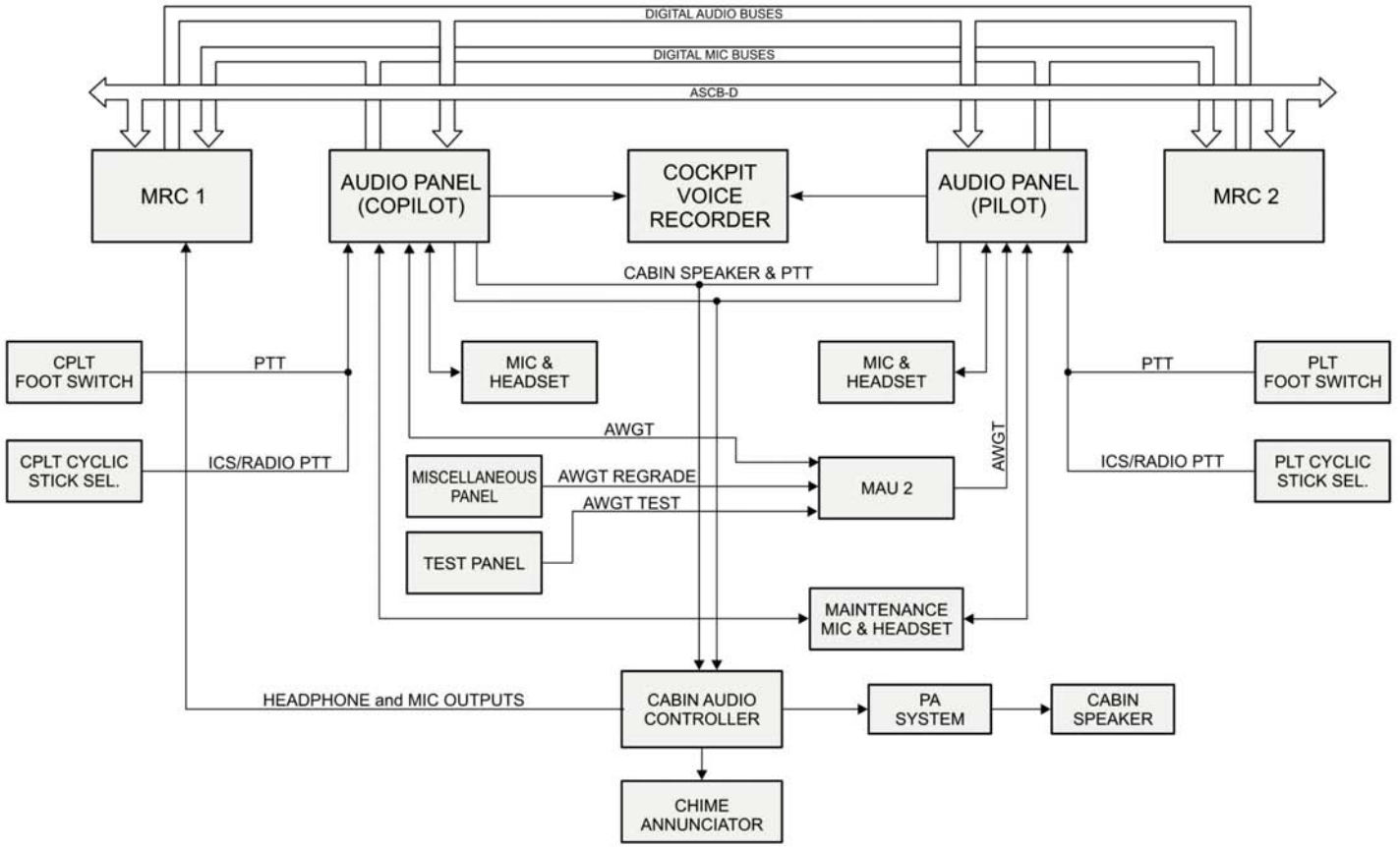
- Navigation (NAV)
- Communication (COM)
- Central Warning System (CWS)

The airborne audio system controls the audio functions of the headphones (HDPH), microphone (MIC), navigation (NAV) and communication (COM) radios.
Main components of the airborne audio system are two Audio Control Panels (ACPs) respectively in the left and right side of the cockpit center console, that are connected to the modular radio cabinets (MRC), including navigation/ communication (NAV/COM) modules, through the digital microphone (MIC) and audio buses. The system lets the pilot and copilot independently control the different audio functions in the aircraft.
Purpose of the airborne audio system is:

- connects the MICs to different radios
- sends audio from different sources to the headsets
- controls the functions of the interphone.

The primary interfaces in the airborne audio system are the two digital audio buses from each MRC to each ACP and the two digital MIC buses from each ACP to each MRC.
Radio audio signals are transmitted from the MRC through the digital audio buses to the ACPs.
Each MRC has one bus output while each ACP has two bus inputs. Each digital audio bus is a high-speed, unidirectional serial bus that multiplexes multiple audio channels to a single shilded-twisted wire pair. Audio signals are digitised in the MRCs for all internal radio modules and changed back to analog in the ACPs.
The transfer of data on the digital audio bus is controlled by the NIM in each MRC.
There are two digital microphone buses in the system to transmit audio signals from and ACP to the MRCs and to other ACPs. Each digital microphone bus is a high-speed, bi-directional serial bus that multiplexes multiple microphone channels to a single shilded-twisted wire. Microphone channels are supplied for for each ACP.
The system can support installation of a maximum of eight ACPs. The microphone signals are digitised in the ACPs and charged back to analog signals by the audio panels for the interphones and MRCs for radio transmissions. The transfer of data on the digital microphone buses is controlled by the ACP.
The analog signals go to the HDPH and cockpit voice recorder outputs
The airborne audio system has the components that follow:

- Audio Control Panel (ACP)
- Cabin Audio Controller (CAC)
- Headset Jack Outlets
- Push-to-Transmit Switch (PTT)



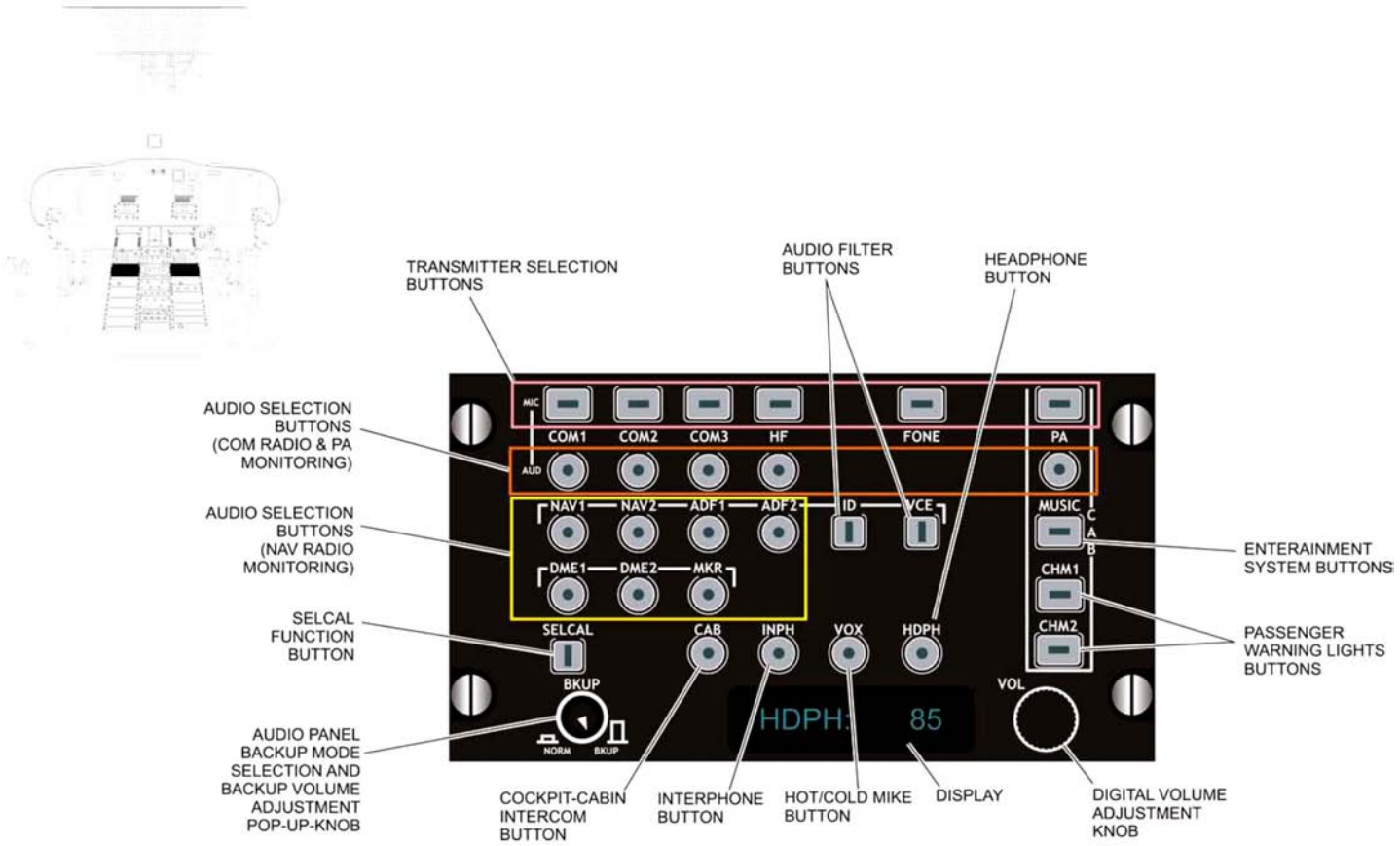
ICN-39-A-230000-G-A0126-00058-A-01-1

AUDIO SYSTEM SCHEMATIC

AUDIO CONTROL PANEL (ACP)

The audio control panel includes transmitter selections buttons, a liquid crystal display (LCD), volume control, audio inputs and audio selection buttons. The lamp on the buttons comes on to identify the active input channel.
The transmitter selection section is composed by rectangular button labeled “MIC” used to activate the COM system to be used. Each “MIC” button has a related circular button labeled “AUDIO” used to activate the only receiver section of the related COM system. The transmitter selections start the related audio in automatic. Only one MIC transmit selection can be activate at a time, but more than one AUDIO selection can be set for listening. The public address “PA” selection it is also connected to a third-party passenger address system.
The audio control panel has the selections and buttons that follows:

- COM..... Very High Frequency
- HF..... High Frequency (optional)
- FONE..... Satellite Communication (optional)
- PA..... Public Address
- NAV..... Navigation
- ADF..... Automatic Direction Finder
- DME..... Distance Measuring Equipment
- MKR..... Marker Beacon
- ID..... Identification Morse code for ADF and NAV Audio
- VCE..... Voice for ADF and NAV Audio
- SELACAL..... (Select Calling) (optional)
- CAB..... Cabin Intercommunication system
- INPH..... Pilot’s Intercommunication System
- VOX..... Volume Knob
- HDPH..... Headphone
- VOL..... Volume Knob
- BKUP..... Emergency Backup
- MUSIC..... Music (optional)
- CHM..... Chime (no smoking and seat belt signs)



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AUDIO CONTROL PANEL

There are two audio control panels (ACPs) mounted at the top left and right of the cockpit center console in the AW139/AB139, one for the pilot and one for the copilot.
The cockpit audio system permits switching microphones to various radios, distribution of audio to headsets and interphone functionality.
The primary interfaces in the audio system are the two digital audio buses from the MRC (one from each MRC) and the two digital microphone (MIC) buses. Both MIC buses are connected to each ACP and MRC in the system and contain identical data.
The ACP is the central point of access for controlling the aircraft audio system, supports connection of up to eight crewmembers audio panels on the digital audio and microphone buses.
It converts digital audio signals from the communication and navigation radios into analog signals that are audible in headphones and cockpit voice recorder (CVR) outputs.
The audio panel supports non--integrated radios such as the high frequency communications (HF COMM) radio.
Applying power to the avionics system powers up the COM, NAV, XPDR, DME, and ADF radios along with the audio panel and each MCDU simultaneously.
To increase mean time between failures and increase integration/control of functions, a single volume knob has been chosen for the ACP. Because of this design, selection of an audio channel is required before the volume can be adjusted or before it can be deselected.
One microphone transmit selection is possible at a time, but listening on more than one audio channel at a time can be done. The default volume adjustment selection that appears in the LCD window is always the headphone volume.
The following scenario is included to help describe the operation:
1. The pilot has selected COM1 to transmit and receive by pushing the MIC. This enables the microphone and audio (headphone) lines for COM1. The pilot adjusts the volume for COM1. After adjusting the volume (allowing 15 seconds to pass without further knob adjustments) it defaults to the headphone (HDPH) in the LCD window.
2. The pilot has tuned another frequency on COM2 and has requested and received a frequency change approval. The pilot selects the COM2 MIC button on the audio panel and is able to adjust theCOM2 volume.
3. Suppose the pilot wants to monitor transmissions on COM1. This is accomplished by selecting the audio (AUD) button of COM1. The COM1 volume can be adjusted. With this setting, the pilot can listen to the transmission on COM1 and listen and transmit on COM2.
As noted in the scenario above, the display in the ACP defaults to the headphone volume setting after 15 seconds.
There are two methods to select and readjust the volume on the other audio functions when the default HDPH is displayed.
The first method is to select the round AUD button for COM2 and hold it down for approximately two seconds and the volume of COM2 is displayed so that the pilot can adjust it.
The second method supposes the COM2 round AUD button is already selected (lit) and the pilot would like to readjust the volume. The pilot can push the COM2 audio button twice to display the volume of COM2 so the pilot can adjust the volume. This action is turning OFF the audio, and back on again. When the COM2 button is not selected, pushing the COM2 button once shows the volume.
The audio system gives a low-level analog audio output to support the connection of a CVR to record the following data:

- Voice communication transmitted from or received by the flight deck by way of the attached radios.
- Audio signals from each pilots' microphone.
- Voice communication of the flight crewmember using the interphone system
- Voice or NAV IDENT signals introduced into the headset or speaker
- Voice communication of flight crewmembers using the PA system.

AUDIO SELECTION BUTTONS

Four rows of round audio selection buttons, grouped side-by-side, are located on the ACP. Pushing an audio select button permits the pilot to receive audio signals from the selected radio. Multiple radios can be selected enabling the pilot to maintain listening on other radio frequencies while communicating on a selected primary frequency.
Centered in each button is a green annunciator light that alerts the pilot to the condition of the radio. Pushing a microphone selection button connects the pilot to the associated radio enabling voice communications and lights the annunciator light. Pushing the button a second time disconnects the radio disabling voice communications and the green annunciator light turns OFF.

Communication Buttons (First Row)

1 COM1 Button: ..... Number one VHF communications radio. Pushing the COM1 AUD button activates/deactivates COM1 audio.

2 COM2 Button: ..... Number two VHF communications radio. Pushing the COM2 AUD button activates/deactivates COM2 audio.

3 COM3 Button: ..... The COM3 button is the number three VHF communications radio. Pushing the COM3 audio activates and deactivates the COM3 audio.

4 HF Button: ..... Pushing the HF button activates and deactivates the HF audio. (Options)

NOTE A

a. The FONE button on the top row of the audio panel supports communication devices such as satellite communications (SATCOM) radio/cell phones. It does not have an audio selection button that is separate from the microphone. Control of tuning other modes for the FONE are through a dedicated controller and are not available with the PRIMUS EPIC system.

b. One microphone selection button can be active at a time. Pushing another button deactivates the first and activates the new selection.

c. The PRIMUS EPIC system gives an interface for a third, non-integrated COM radio. This radio can be VHF, FM or UHF radio. Control of tuning and other modes for this radio is through a dedicated controller.

Navigation Buttons (Second row)

5 NAV1 Button: ..... Pushing the NAV1 button activates and deactivates audio from the number one navigation radio.

6 NAV2 Button: ..... Pushing the NAV2 button activates and deactivates audio from the number two navigation radio.

7 ADF1 Button: ..... Pushing the ADF1 button activates and deactivates audio from the number one ADF.

8 ADF2 Button: ..... Pushing the ADF2 button activates and deactivates audio from the number two ADF.

9 ID Button: ..... When the ID button is pushed, the ADF and NAV audio filter attenuates the voice audio so the Morse Code IDENT can be prominently heard.

10 VCE Button: ..... When the VCE button is pushed, the ADF and NAV audio filter attenuates the IDENT audio so the voice audio Or Morse Code IDENT can be prominently heard.

Navigation Buttons (Third row)

11 DME1 Button: ..... Pushing the DME1 button activates/ deactivates audio from the number one DME source.

12 DME2 Button: ..... Pushing the DME2 button activates/ deactivates audio from the number two DME.

13 MKR Button: ..... Pushing the MKR button activates and deactivates audio from the Marker Beacons.

Internal System Audio Controls (Fourth Row)

14 SELCAL (Select Calling) Button (option):

- The system radios can be configured with select calling capability. When a SELCAL is received, the SELCAL button and the annunciator light for the proper radio flashes. The SELCAL function decodes all the VHF and HF COMM digital audio signals.

15 CAB (Cabin) Intercommunication System (ICS) Volume Control Button:

- Pushing the CAB button connects pilot to the cabin ICS. Pushing the PILOT button on the cabin audio controller flashes the CAB button annunciator light. The CAB button on the pilot-side ACP flashes and generates a single tone that is sent through the PA system and to the cabin audio controller. The acknowledgement of the cabin call by the cabin audio controller is indicated as a fast flash on the CAB button. When the CAB button is flashing fast, it establishes an audio connection between the cockpit ICS channel and cabin audio controller. This is indicated by a steady lighting of the CAB annunciator. The annunciator on the CAB button slowly flashes and a single audio tone sounds in the pilots' headset when a request for cabin ICS is initiated by the cabin audio system. When the pilot pushes the CAB button after a call from the cabin, it establishes an audio connection between the cockpit and the cabin. This is indicates by a steady light.

16 INPH Pilot Intercommunication System (ICS) Volume Control Button:

- The INPH button connects and disconnects the pilot to the ICS.

17 VOX Voice-Activated Squelch (VOX) System Button:

- Pushing the VOX button turns the VOX system and the associated annunciator light ON and OFF. With the VOX system ON, tuning the VOL Knob in the lower right corner of the audio panel increases or decreases the VOX system sensitivity.

18 HDPH (Headphone) Pilot Master Volume Control Button:

- Pushing the HDPH button turns the headphone master volume control ON and OFF. With the master volume control system ON, tuning the VOL knob in the lower right corner of the ACP increases or decreases the volume of all the active radios simultaneously.

Cabin Audio System

Cabin Audio System enables communication between the cockpit and cabin. There are two options given for the electrical interface. However, the two cabin communications are consistent. The cabin

communication can be implemented using the Honeywell cabin audio controller or a generic cabin audio system interfaced by way of a conventional analog audio and call discrete.
Four buttons are grouped vertically in a white outlined box on the right side of the control head. The word CAB printed vertically on the right side of the outlining box identifies them as the cabin audio control buttons.
All of these buttons have the green annunciator light in the center. The cabin audio system control buttons are described in the following paragraphs.

19 PA Button: ..... Pushing the PA button connects and disconnects the pilot microphone to the passenger compartment public address system. Pushing the button turns OFF any other previously connected microphone.

20 Music Button: ..... Pushing the MUSIC button connects the cabin crew or passengers to the cabin entertainment system.

21 CHM1 and CHM2 Buttons: … when the CHM1 or CHM2 button is selected, the audio system turns ON the corresponding button annunciator and activates cabin visual annunciators, such as seat belts. These two buttons toggle ON and OFF. The chime buttons operate when CHM1 is pushed on the pilot-side ACP, the CHM1 button on both the ACPs lights. The same is true for CHM2. when one pilot turns a chime OFF, the annunciators goes out on both panels.

22 BKUP Button ..... When the BKUP button is in the ON position, the headset normally connected to that audio panel is connected to the opposite side ACP to give backup intercom system (ICS). Pushing the PTT button connects the pilot microphone to the On-side communications radio.

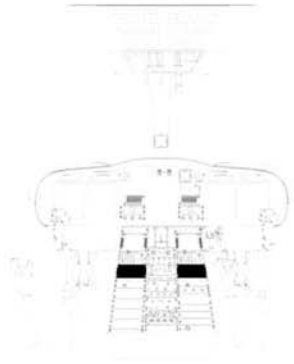
23 Frequency Window: ..... the frequency window, found just to the left of the volume control knob, shows the selected radio and the current setting. It shows some messages.

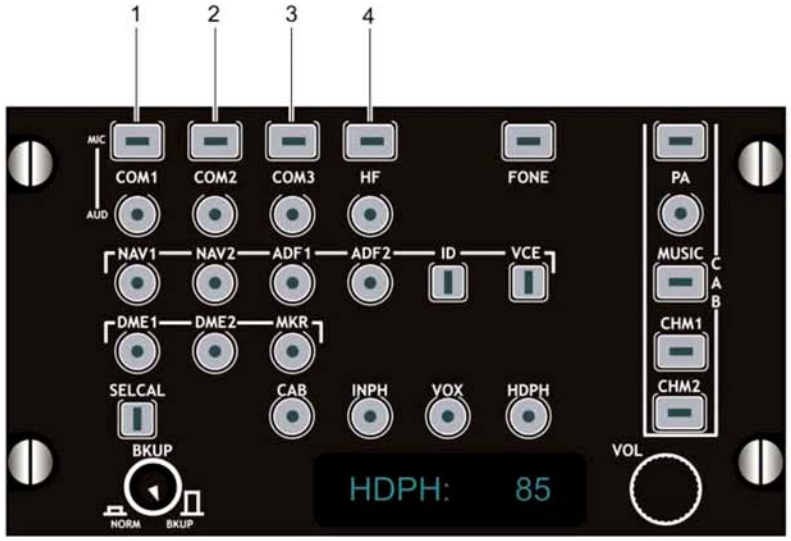
24 VOL (Volume Knob): ..... A volume control knob is positioned in the bottom right corner of the ACP next to the control window.
Change the audio volume of a radio using the following steps:

- Push the button associated with the desired radio to do the following:
- Annunciate the selected radio and current volume setting in the display window.
- Activate audio
- Light the green annunciator light on the button
- Change the volume of the selected radio by turning the volume control knob clokwise to increase volume and counterclowise to decrease volume

NOTE B

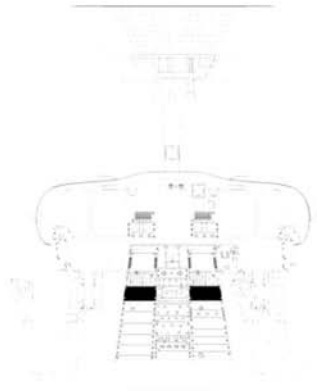
The volume control knob controls the volume of the radio displayed in the control window. The exception is selecting the headphone (HDPH) button. When HDPH is displayed in the control window, the volume control knob adjust the volume of all selected radios simultaneously.

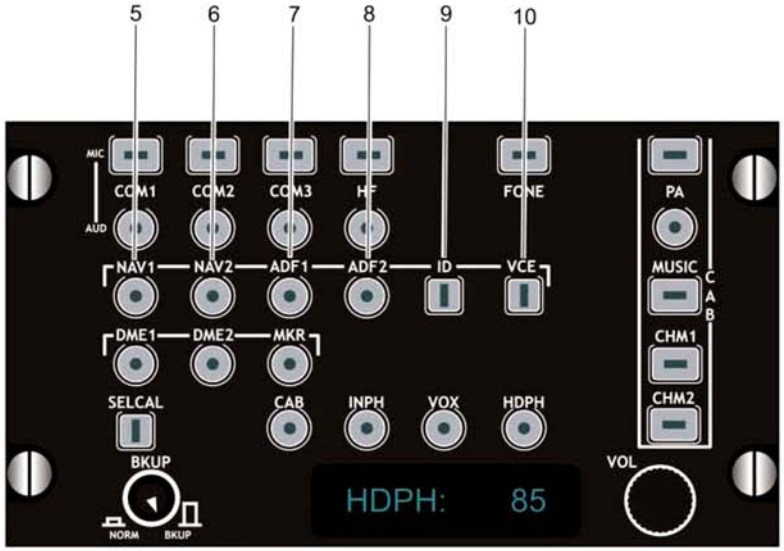




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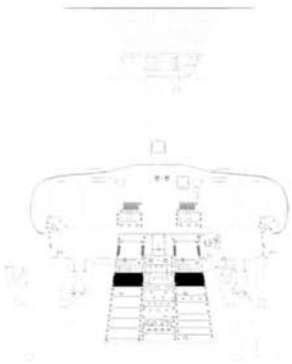
AUDIO CONTROL PANEL-FIRST ROW (COMMUNICATION BUTTONS)

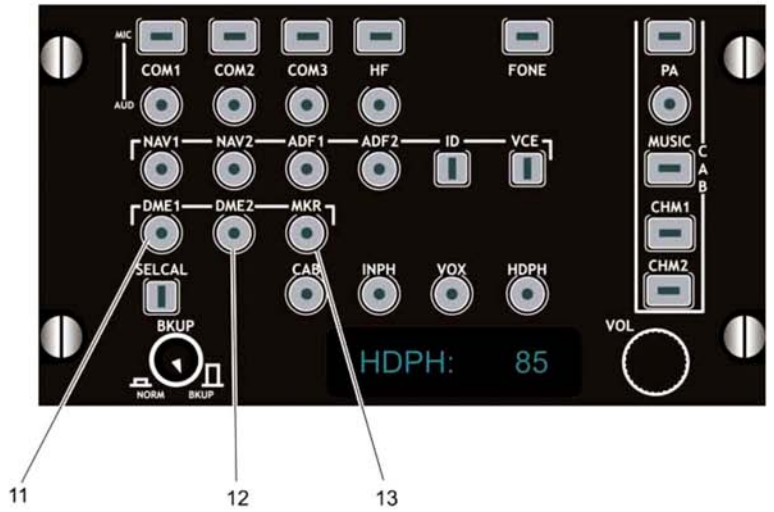




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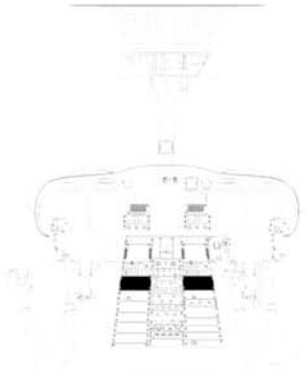
AUDIO CONTROL PANEL – SECOND ROW (NAVIGATION BUTTONS)

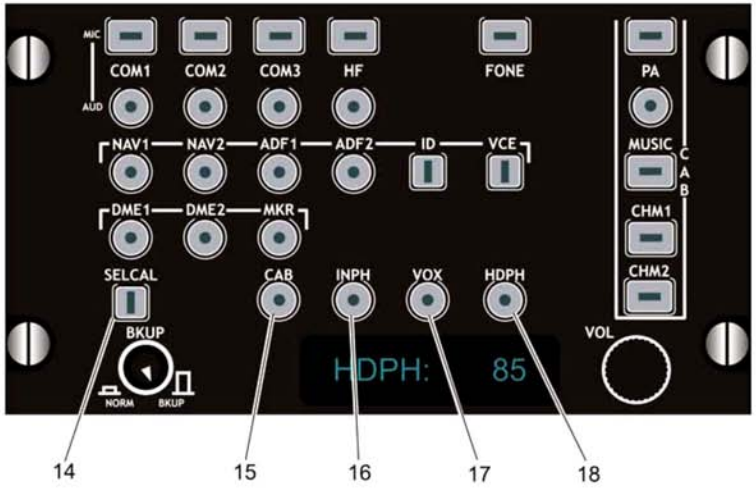




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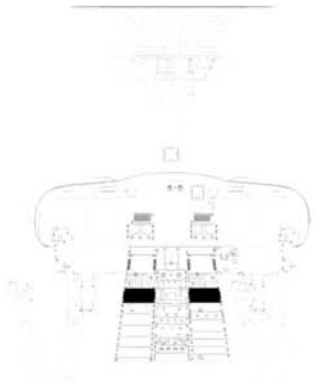
AUDIO CONTROL PANEL-THIRD ROW (NAVIGATION BUTTONS)

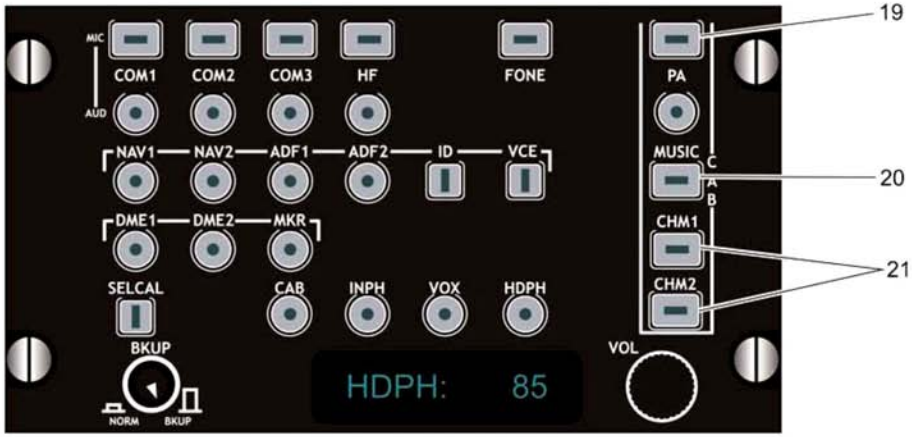




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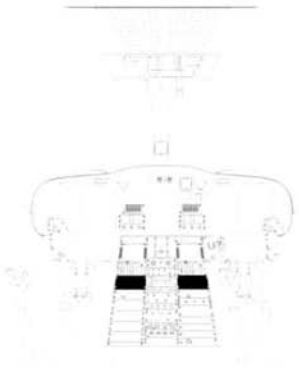
AUDIO CONTROL PANEL-FOURTH ROW (INTERNAL SYSTEM AUDIO CONTROL BUTTONS)

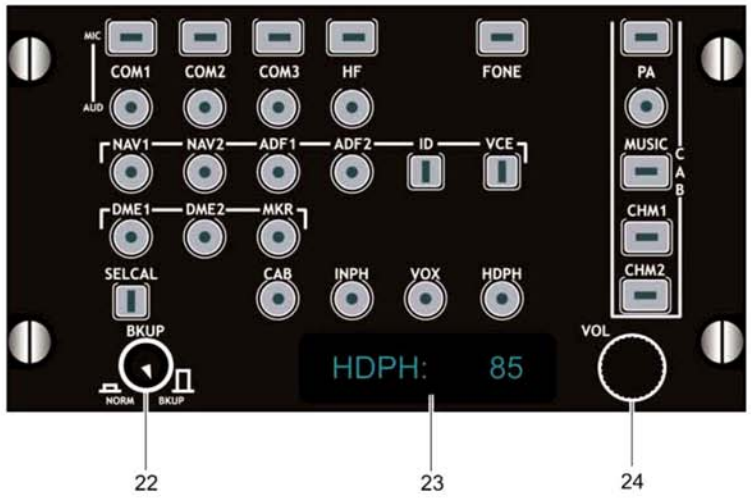




ICN-39-A-230000-G-A0126-02906-A-01-1

AUDIO CONTROL PANEL (CABIN AUDIO SYSTEM BUTTONS)





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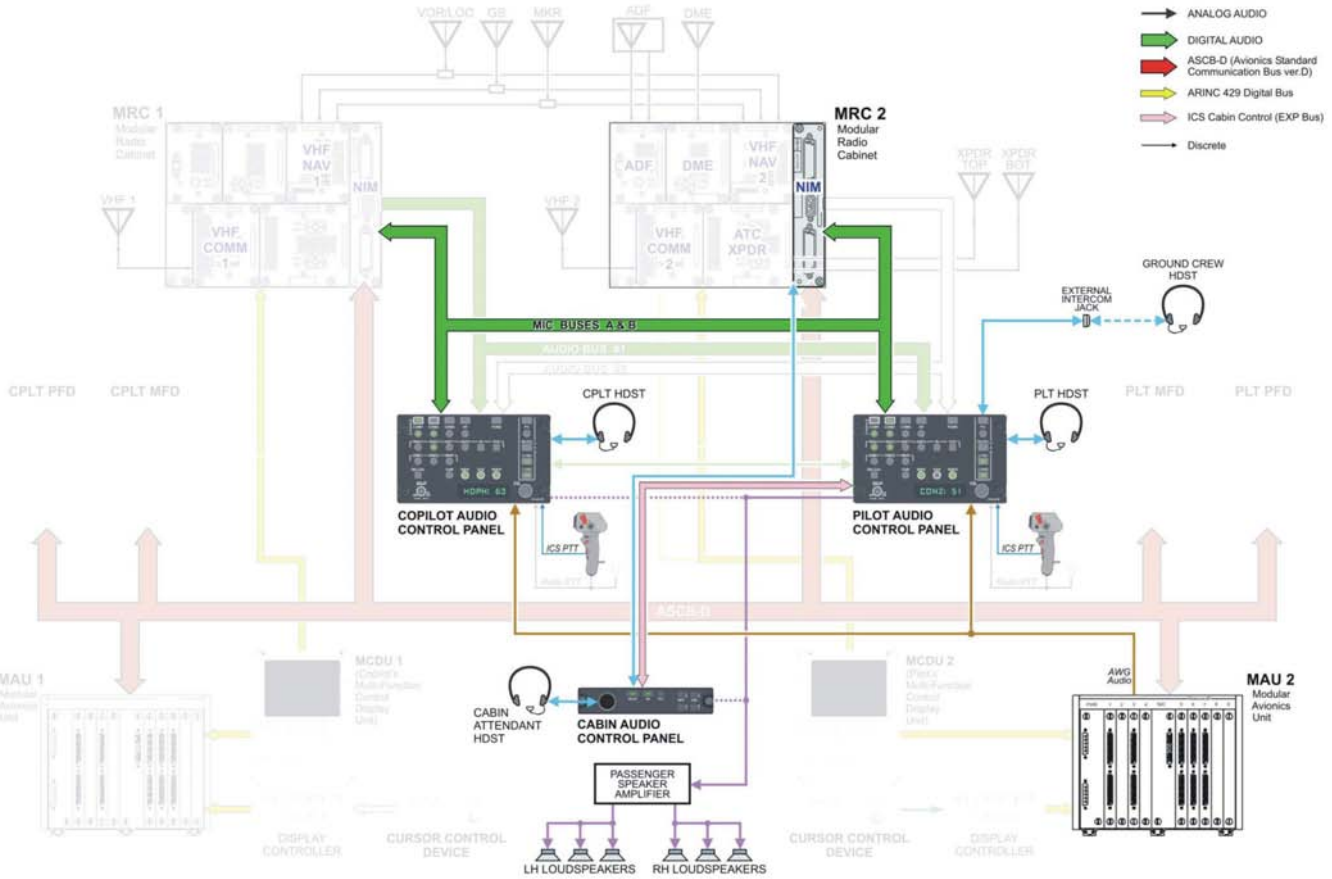
AUDIO CONTROL PANEL – SECOND ROW (NAVIGATION BUTTONS)

GENERIC CABIN ICS AUDIO INTERFACE

Audio system enables interface and operations for a generic cabin ICS audio controller. The audio interface permits paging and cabin communication with the cockpit crew. The following paragraphs contain the interface operational functions for the generic cabin ICS controller.
The ACP gives the following set of discrete inputs and outputs to facilitate two-way paging between the cockpit ACP and the cabin audio system:

- A Call In pin is given to permit the generic cabin controller to page the cockpit crew.
- A Call Out is given to permit the cockpit crew to page personnel using the cabin audio system or to acknowledge a cabin page.
- A Cabin Acknowledgement to the ACP is given to permit the generic cabin controller to acknowledge a cockpit crew page.

When a call is initiated from the cabin, a Call In is received by the ACP.
When the Call In signal is received, the CAB button annunciator on the cockpit audio control panel slowly flashes and plays an audio tone.
When the cockpit crew is ready to communicate with the cabin, they accept the incoming call by pushing the CAB button.
When the CAB button is pushed when the annunciator is flashing, the audio interface with the cabin audio system is enabled and the CAB annunciator shows a steady annunciator.
The cockpit crew can initiate a call by pushing the CAB button.
When the CAB button is pushed, an audio tone plays through the right cockpit speaker PA and the CAB annunciator slowly flashes.
The cabin audio system can acknowledge a call from the cockpit.
When the CAB button is flashing slowly and the acknowledgement is set, and an audio tone plays in the pilots' headset and initiates a fast flash of the CAB button.
When the CAB button is pushed, the audio interface with the cabin enables and a steady light is displayed on the CAB annunciator.



ICN-39-A-230000-G-A0126-00066-A-01-1

MODULAR RADIO SYSTEM BLOCK DIAGRAM – INTERCOM & PA

CABIN AUDIO CONTROLLER (CAC)

The cabin audio controller is mounted in the cabin and it is intended for use by a single cabin operator.
The cabin audio controller includes a headset jack and enables interphone communication with the flight crew and public address functions for communication with other passengers.
The buttons on the cabin audio controller head are used to control specific functions of the audio system. Pushing the buttons activates the respective function. Pushing the button again deactivates the function. The cabin audio controller buttons and their functions are described in the following paragraphs.

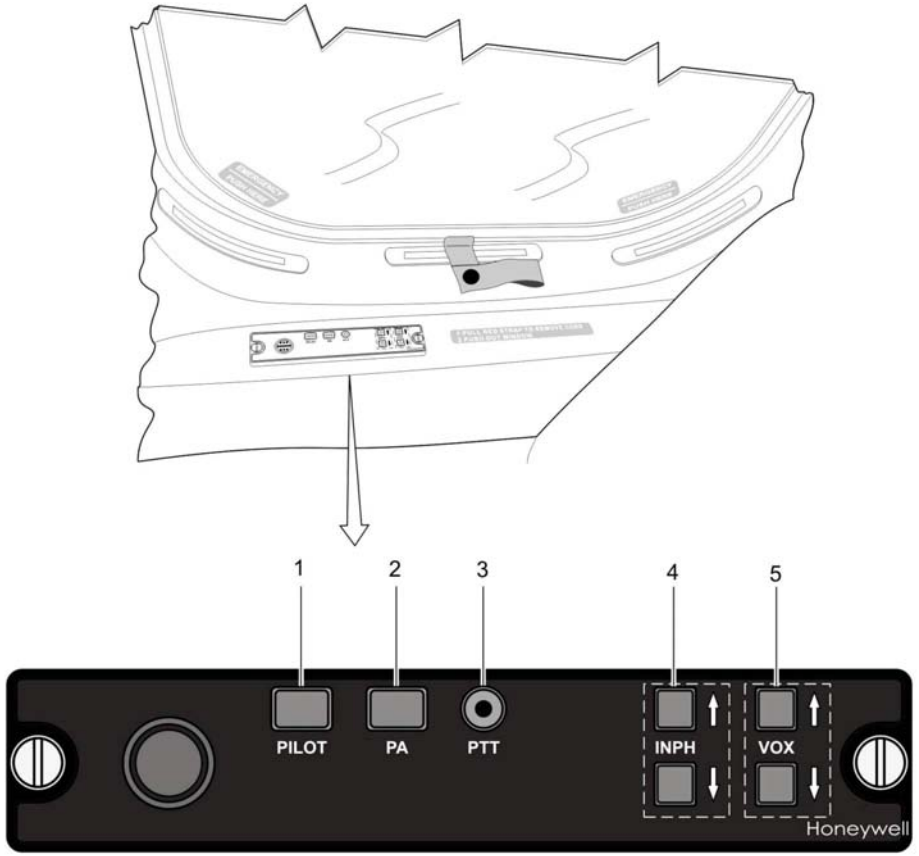
1 Pilot Button: ..... the PILOT button activates the cabin intercommunication system (ICS). It supplies cabin crew or passengers a discrete means of contacting the pilot by flashing the CAB button on that ACP and a chime is sounded in the headset. The CAB button on the ACP continues to flash and the chime sounds until is selected by the pilot or deselected by the cabin operator. The acknowledgement and connection from the cockpit crew results in a steady lighting of the PILOT button.
When the PILOT annunciator is ON, the cabin operator can listen and talk on the cockpit ICS.

2 PA (Public Address) Button: ..... pushing the PA button connects cabin crew to the public address system. The PA audio is available through the cabin speaker output.

3 PTT (Push to Talk) ..... pushing the PTT button enables the pilot to speak over the public address system.

4 INPH (Interphone) Buttons: ..... the INPH buttons are used to control headset audio volume control. Pushing the button beside the up arrow increases headset volume, pushing the button beside the down arrow decreases headset volume.

5 VOX (Voice Activated Squelch) Buttons: ..... the buttons are used to control VOX volume control. Pushing the button beside the up arrow increases headset volume, pushing the button beside the down arrow decreases headset volume.



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CABIN AUDIO CONTROLLER (CAC)

MAINTENANCE COMMUNICATIONS

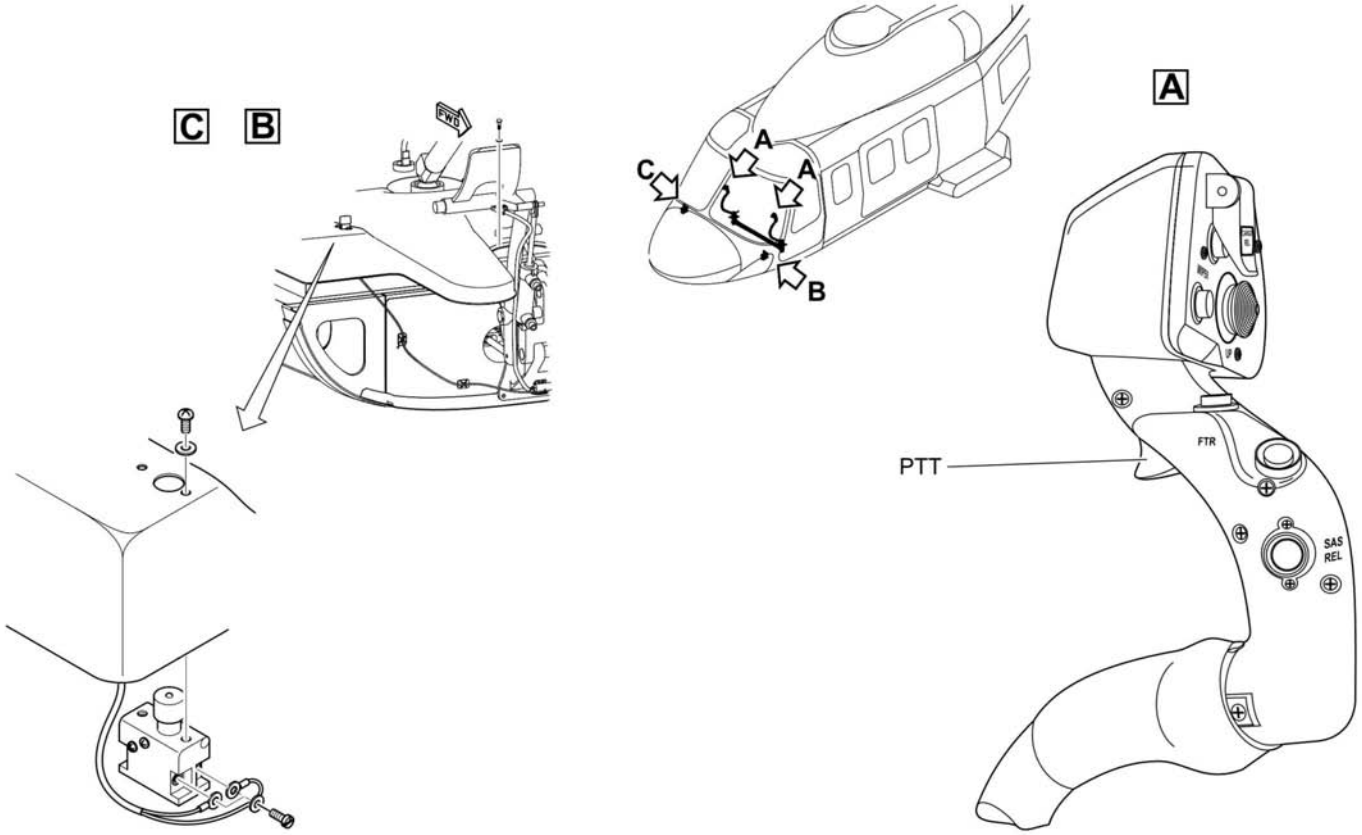
The cockpit audio system supplies an interphone port that can be used by maintenance or ground support personnel. The audio panel does not have a dedicated control for the maintenance port headset volume. Use of headsets with an integrated volume control enables maintenance personnel to adjust the headset volume up to the limit defined by the configuration setting. The maintenance ports are connected to the pilots ICS at all times.
Pushing the INPH button enables the maintenance port lines. The maintenance port one is dedicated for use with the backup ICS mode. External headsets must not be connected to maintenance port one.

PUSH TO TRANSMIT (PTT) SWITCH

The pilot and copilot each have a PTT switch installed on the cyclic and a foot activated PTT switch. The switches connect the pilot or copilot MIC to the relevant audio control panel.
The cyclic control PTT switch has three positions:

- OFF:..... in rest condition.
- INTERCOM:..... middle press for the interphone system (ICS).
- TRANSMIT:..... full press for the radio transmission.

The foot PTT switch provides the only function of “RADIO TRANSMISSION”.
A PTT switch is located on the cabin audio controller. It is used by the cabin operator to talk to the pilot if the voice operated transmits “VOX” sensitivity is set too high.

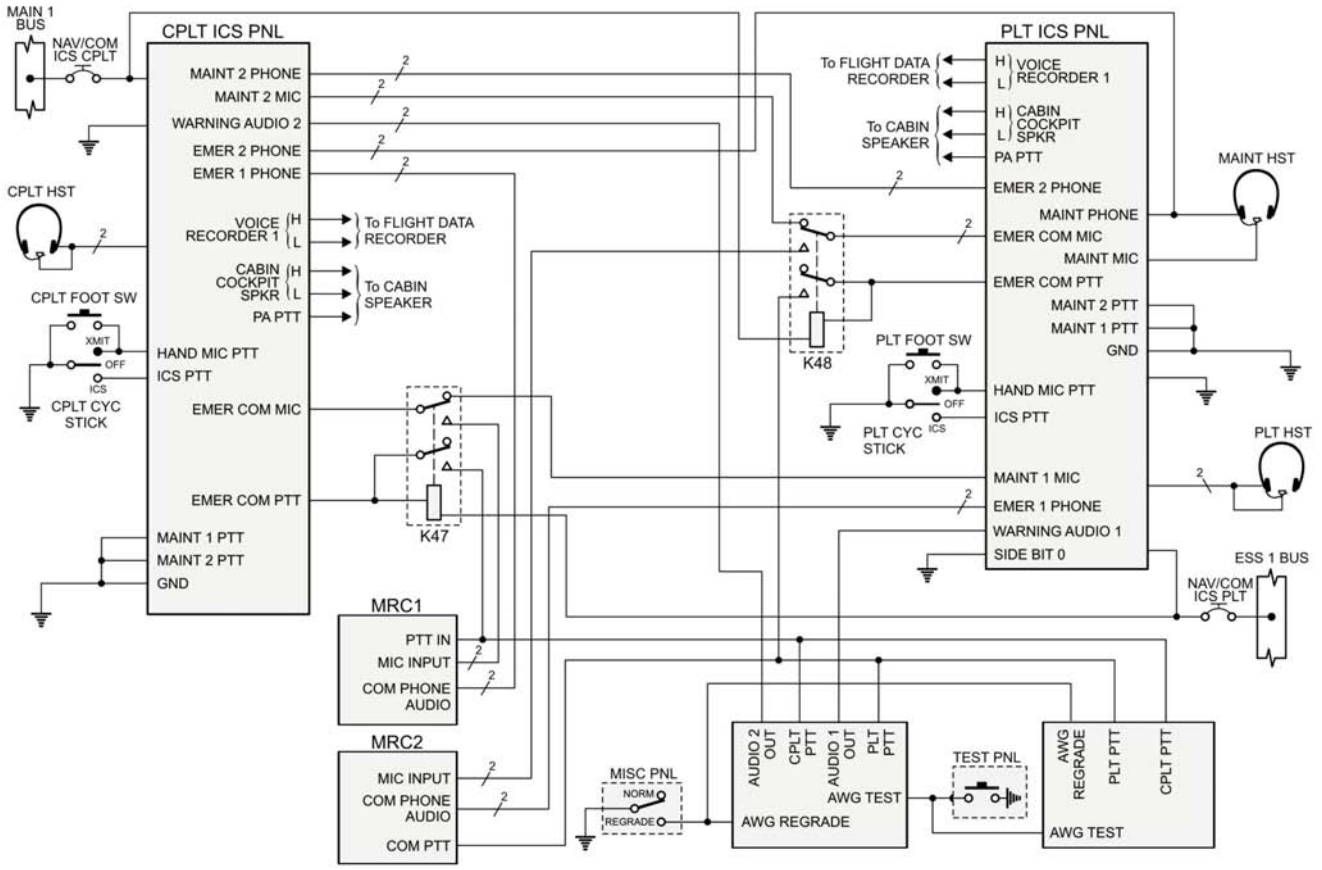


ICN-39-A-230000-G-A0126-00068-A-01-1

SPEECH COMMUNICATION COMPONENTS

BACKUP ICS MODE

Each ACP has a backup mode that can be selected when one of the audio panels fail. The backup mode permits the pilot headset to be connected directly to the on-side VHF COMM and off-side ACP (ICS), through the relays K47 and or K48.
No power is required to the on-side ACP. Switching the pilots MIC between the COM radio and the off- side maintenance port is performed by an external relay (K47 and or K48) controlled by the PTT button. The audio panel backup mode controls the ICS and COM audio volume through the use of a single volume control knob. The control knob is located on the lower left corner of the audio panel and labeled BKUP.
The ICS and COM audio volumes cannot be adjusted separately.



ICN-39-A-230000-G-A0126-00069-A-01-1

INTERPHONE ELECTRICAL SCHEMATIC

WARNING AUDIO INPUTS

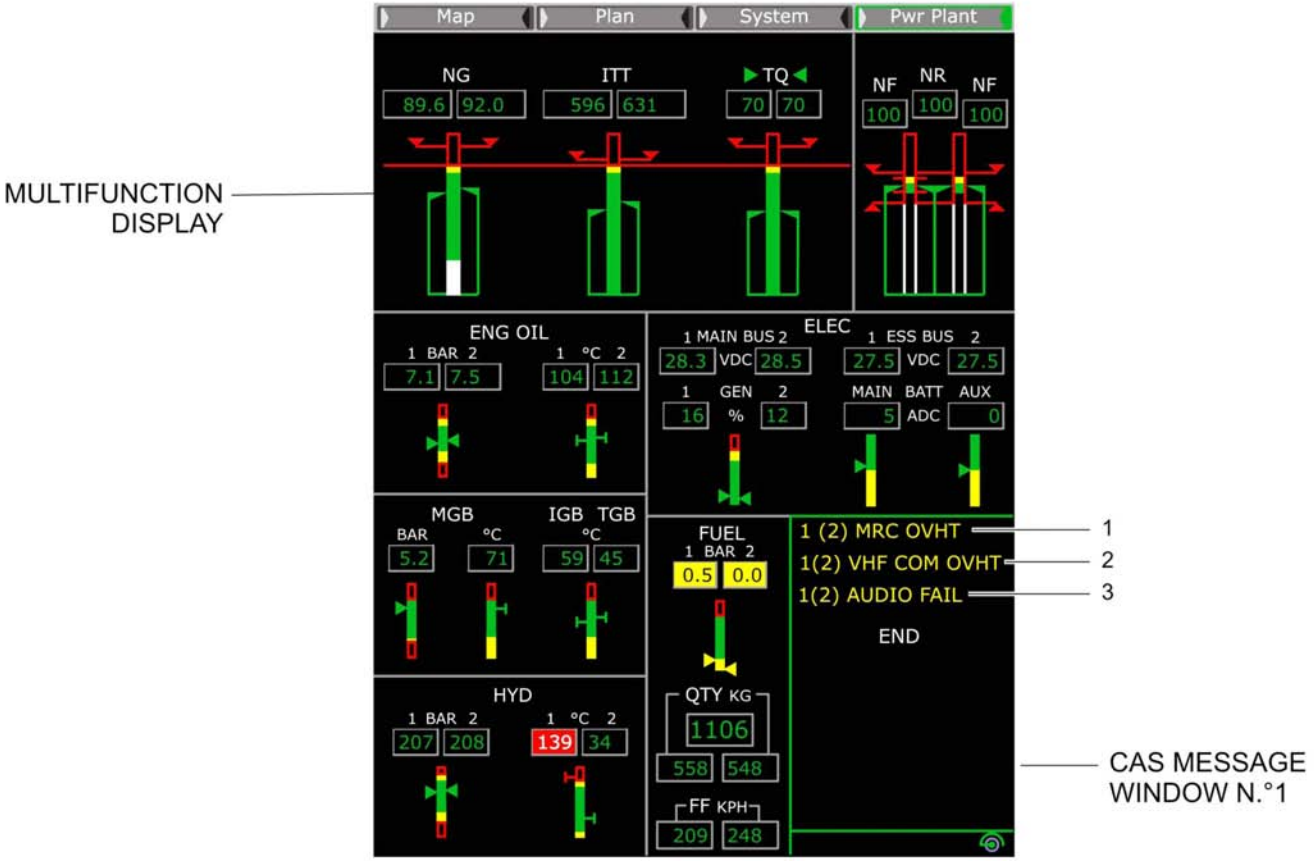
Each Pilot and Copilot audio control panel has a dedicated audio input which receives signals from the aural warning function in the central warning system. This input is non muting (can not be interrupted) and is adjustable for the volume. The set volume level is kept in the nonvolatile memory. These warning sounds identify possibly dangerous aircraft operations or flight conditions.

CAPTIONS

CAUTION

Cautions in the CAS messages windows are displayed in the event of radio communication systems failure.
Failure annunciations:

1	1 (2) MRC OVHT:	this amber message is displayed when the 1(2) MRC Over Temperature condition is detected.
2	1 (2) VHF COM OVHT:	this amber message is displayed when the VHF1 (2) COM system Over Temperature condition is detected.
3	1 (2) AUDIO FAIL:	This amber message is displayed in case of: • CPLT (PLT) audio panel in fail condition. • CPLT (PLT) audio panel in over temperature condition. • Microphone connected to the CPLT (PLT) audio panel in stuck condition. • Failure condition detected for the MIC or AUDIO DATA BUSES.



ICN-39-A-230000-G-A0126-00070-A-01-1

COMMUNICATION SYSTEM INDICATING

LEADING PARTICULARS (TR-865 VDR MODULE)

- Power Requirements
  - DC Voltage Normal Operational..... 27.5 VDC nominal
  - Range..... 22.0 VDC minimum, 30.3 VDC maximum
- Circuit Breaker ratings:..... 28 VDC circuit breaker 10 A
- Frequency Range
  - Normal range..... 118.000 to 136.975 MHz
  - Extended range (if applicable)..... 137.000 to 151.975 MHz
- Channel Spacing..... 8.33 or 25 KHz
- Frequency Stability..... ± 0.0005 %
- Transmitter Power Output..... 16 to 25 W (20 W nominal)
  15 W minimum mode 2 (if applicable)
- Duty cycle..... 20% maximum, 1 min transmit, 4 min receive
- Harmonic Content..... - 70 dB from 50KHz to 1215 MHz
  - 90 dB from 1015 MHz thru 1045 MHz
- Spurious Output..... 0 to 2500 Hz □ 6 dB
  3200 Hz 45 dB
  5000 Hz 60 dB
  8330 Hz 70 dB
- Carrier Noise Level..... 50 dB below demodulated output at 85% modulation

SECTION 10 - SPEECH COMMUNICATION

NUMBER 1 VHF DATA RADIO (VDR) SYSTEM

The Number 1 VHF Data Radio (VDR) system includes these primary components:

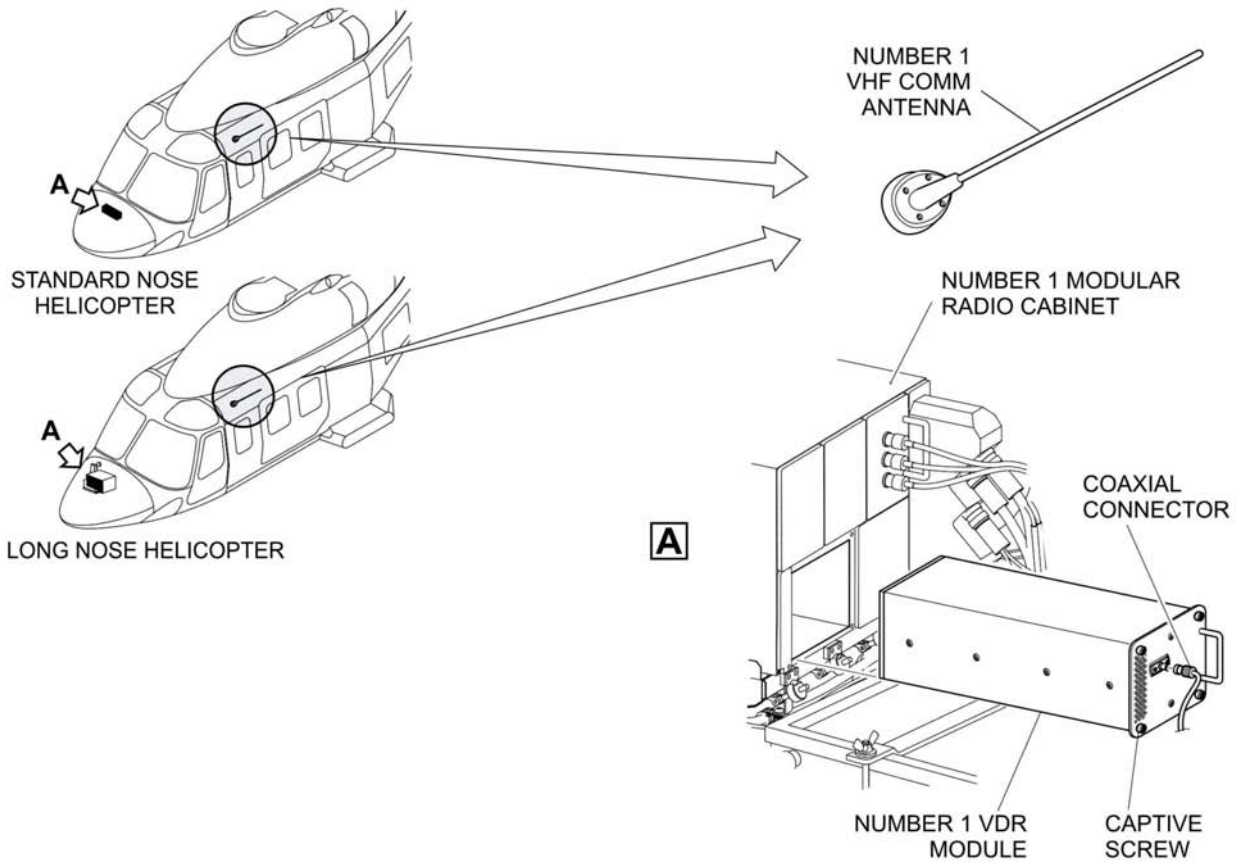
- The Number 1 VDR module
- The VHF COM1 antenna

NUMBER 1 VDR MODULE

The Number 1 VDR module is contained in the Number 1 Modular Radio Cabinet (MRC1), which is installed on the left side of the helicopter nose.
The Number 1 VDR module includes four circuit card assemblies, a front mounting plate and backplane connector.
Four captive screws hold the Number 1 VDR module in its correct position.

VHF COM1 ANTENNA

The Number 1 VHF COM antenna is installed on the left side of the fuselage top. Four screws attach the Number 1 VHF COM antenna to its base. The electrical connectors A7 EP1 and E3 P1 connect the Number 1 VHF COM antenna (E3) to the Number 1 VDR module.



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NUMBER 1 VDR SYSTEM

NUMBER 2 VHF DATA RADIO (VDR) SYSTEM

The Number 2 VHF Data Radio (VDR) system includes these primary components:

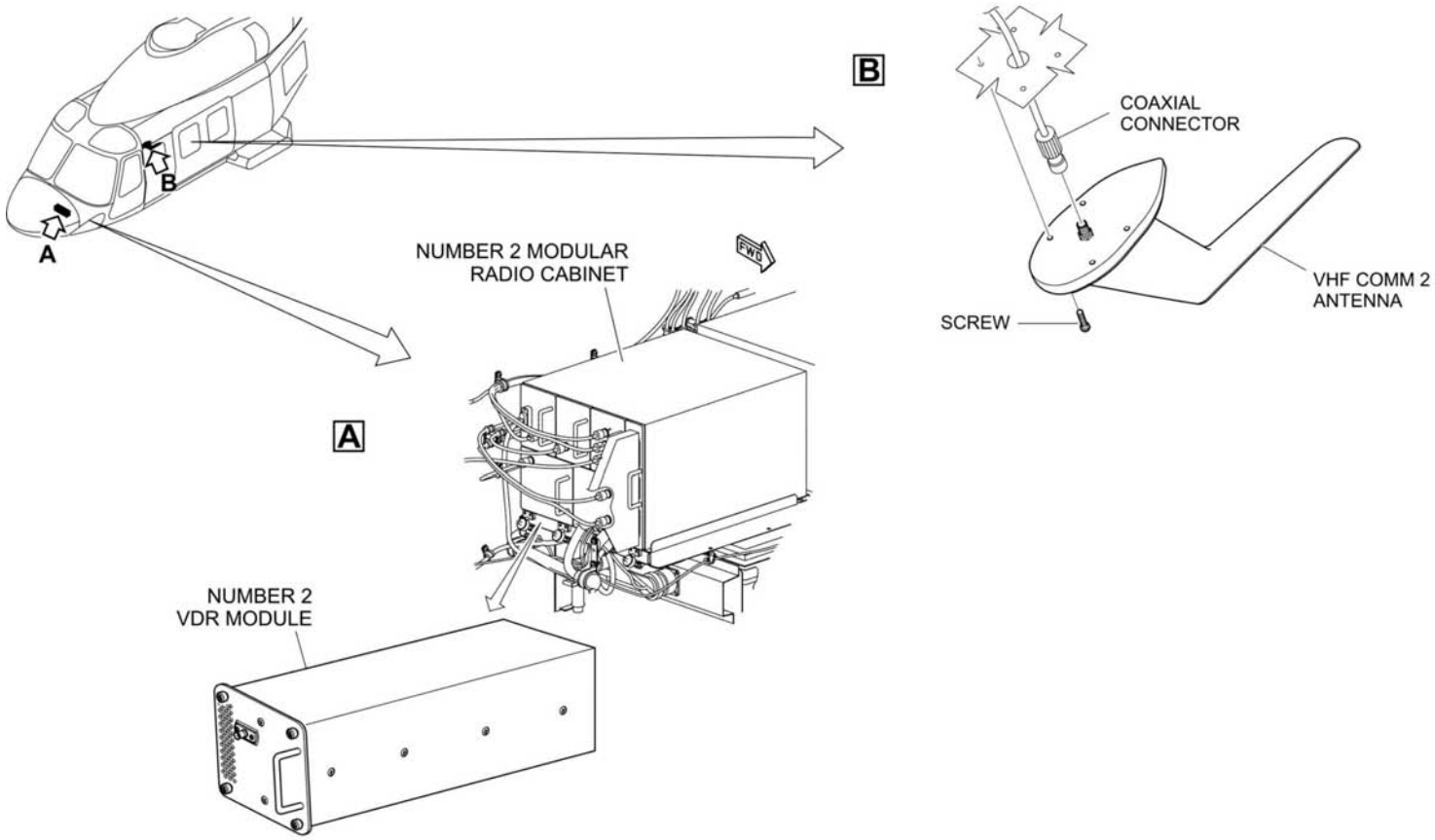
- The Number 2 VDR module
- The Number 2 VHF COM antenna (E4).

NUMBER 2 VDR MODULE

The Number 2 VDR module is contained in the Number 2 Modular Radio Cabinet (MRC2), which is installed on the right side of the helicopter nose.
The Number 2 VDR module includes four circuit card assemblies, a front mounting plate and a backplane connector. The electrical connectors A8-3P1 and E4 P1 connect the Number 2 VDR module to the Number 2 VHF COM antenna (E4).
Four captive screws attach the Number 2 VDR module.

NUMBER 2 VHF COM ANTENNA

The VHF COM2 antenna is installed on the bottom of the center fuselage, adjacent to the right sponson.
Four screws attach the VHF COM2 antenna to the structure.



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NUMBER 2 VDR SYSTEM

VDR SYSTEM OPERATION

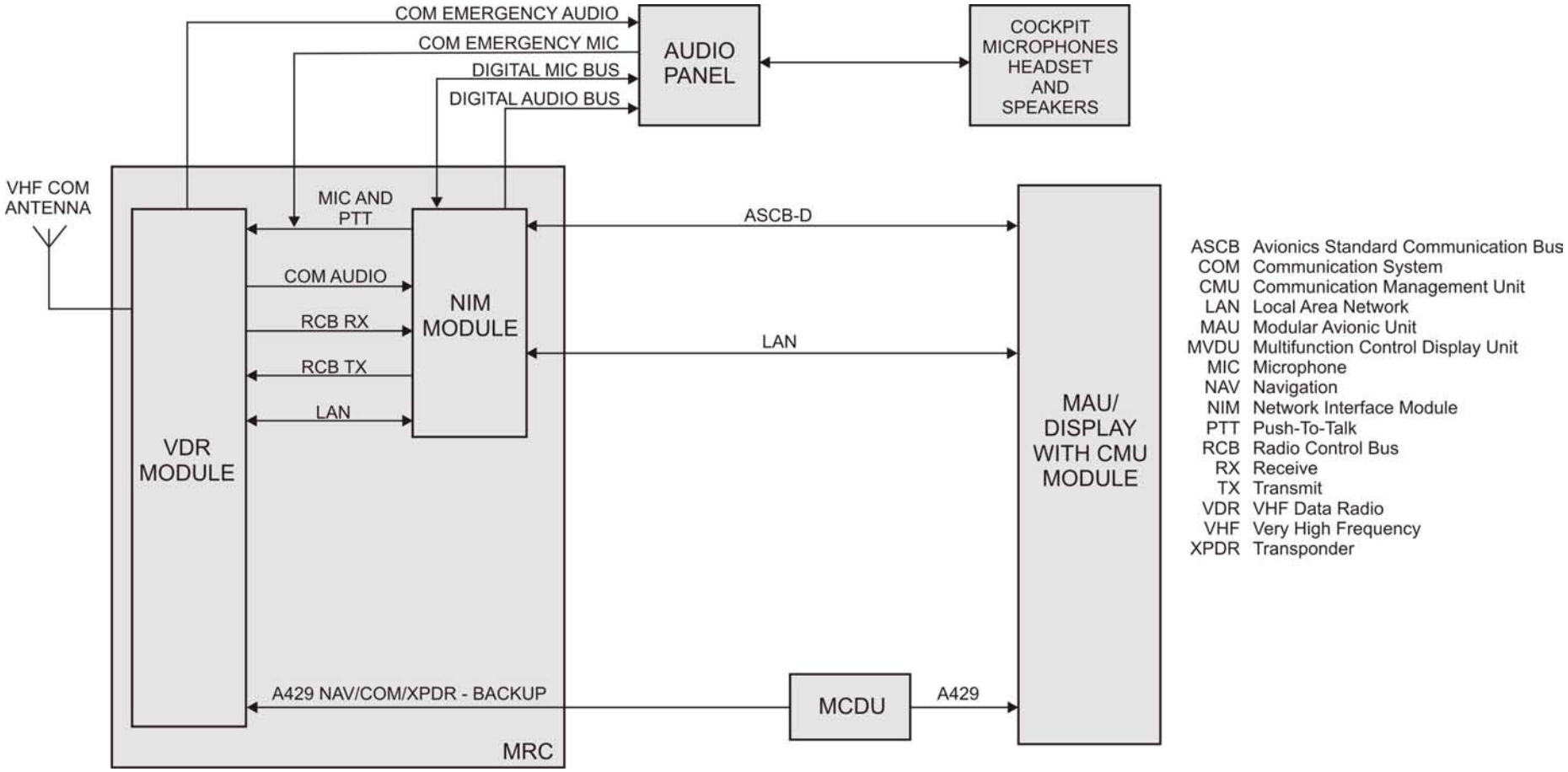
The VDR system operates in the range of 118.000 to 136.975 MHz. The system transmits and receives ACARS data at a rate of 2.4 kb/s, and D8PSK data modes at a rate of 31.5 kb/s. The VDR module uses +28 V dc.
The VDR module also supplies the fault history and the status data to the maintenance system.
The VDR module is a two-way VHF transmitter-receiver that includes the Circuit Card Assemblies (CCA) that follow:

- VHF receiver
- VHF transmitter
- I/O
- CPU/power supply

The VHF receiver receives and demodulates the voice signals and the data. The VHF transmitter modulates and transmits the voice signals and the data. The discrete, audio and analog signals go through the I/O CCA. The power supply CCA supplies the internal voltages necessary for the operation of the VDR module.
The CPU has the functions that follow:

- Digital bus I/O
- Frequency mode control
- Built-in Test (BIT)
- Maintenance
- AM envelope detection
- Filtering
- Squelch control
- AM modulation
- Audio signal processing
- ARINC 429 interfaces
- Commercial Standard Digital Bus (CSDB) interfaces
- Auxiliary tuning interfaces.

The VDR module is set to a voice mode or a VHF Data Link (VDL) mode for usual operation.



ICN-39-A-231000-G-A0126-00221-A-01-1

VDR OPERATION SCHEMATIC DIAGRAM

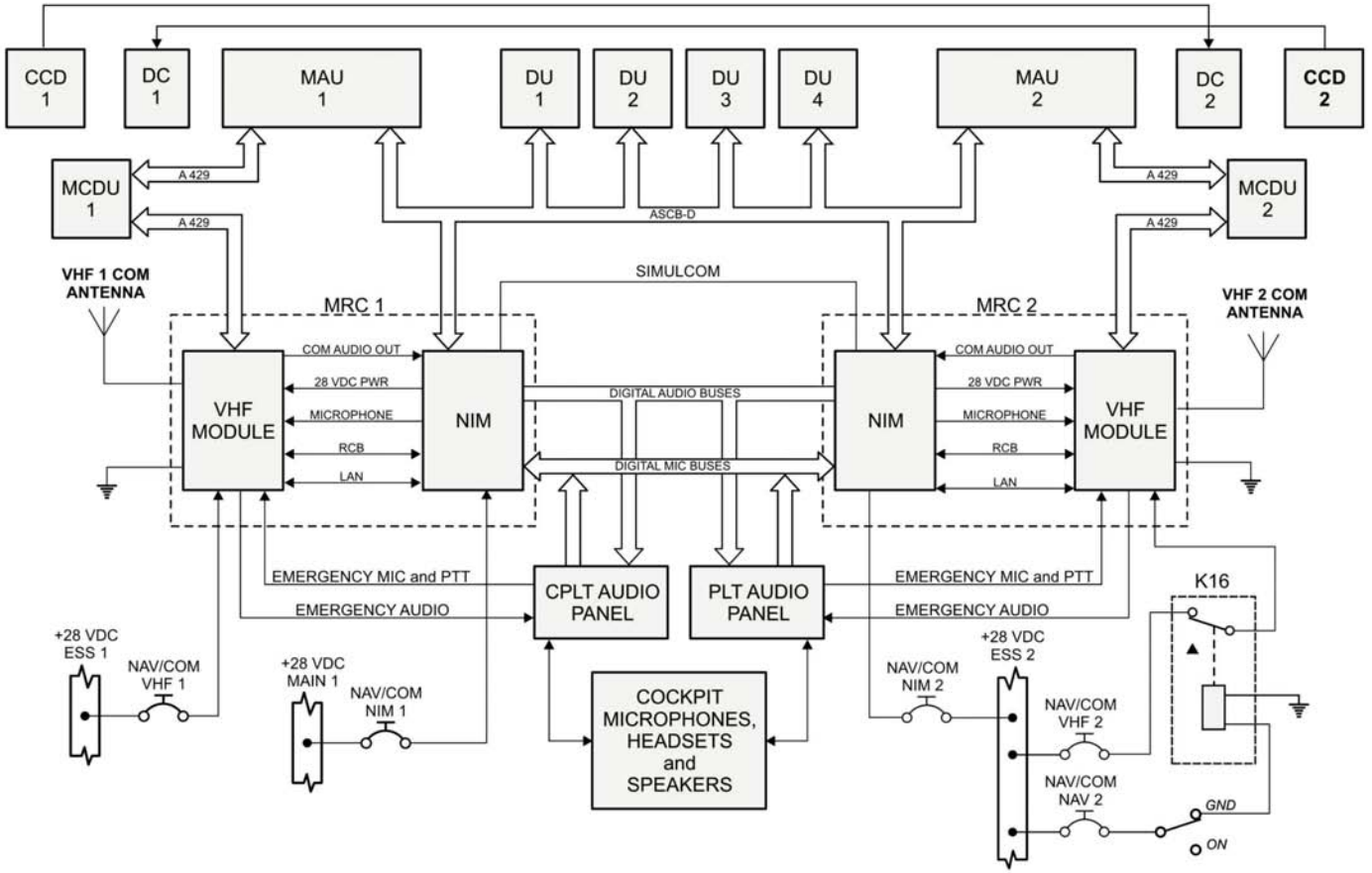
In the voice mode, the VDR module receives the AM microphone signals that modulate the transmitter when the push-to-talk signal is on. In the other modes the VDR module receives and demodulates the AM voice audio. The signals are then sent through the NIM (Network Interface Module) to the audio system. In the data modes, the VDR module modulates and transmits the data from the CMU. The VDR module receives the data from the ground station, and demodulates them. Then the data are sent to the CMU. The VDR module supplies the fault history and status data to the maintenance system.
The VDR module can start the emergency mode from each of the other modes of operation. The emergency mode immediately supplies voice communication on the emergency frequency of 121.5 MHz. When the emergency mode is started, all the tuning data come from the input bus that started the mode.
The VDR module can be energized independently. The power-on/off control-signal goes through the NIM to the VDR module without change. This low-power control-signal controls the power supply of the VDR module.
The audio circuit supplies the noise-squelch function and the carrier-level squelch function. These functions operate together with the multicarrier networks. The audio circuit also supplies the outputs for SEL CAL. SEL CAL sends audio signal codes (different tones) on a COM channel to let the ground station speak with only one aircraft. The audio circuit supplies an output of 100 mW into a 600 ohm load. It also has an isolated side-tone output that lets the receiver and transmitter transmit their audio outputs to the related analog audio systems. The audio circuit supplies the transmitter side-tone when it senses the VDR module output. This function supplies a full check of the modulator and transmitter of the VDR module.
The transmitter has a time-out function. This function causes the VDR module to automatically go back to the receive mode after it transmits continuously for 2 min. The function also helps prevent interference or intermittent operation when more than one VDR module uses the same frequency.

MAINTENANCE LOG AND INTERNAL MONITOR CIRCUIT

The VDR module has other standard functions that include a non-volatile maintenance log and an internal monitor circuit. The maintenance log records the possible data errors and makes them available to the maintenance personnel. The internal monitor circuit monitors the circuit performance. The 1 VHF COM OVHT caution message is shown in the CAS message window of the MFD. This occurs when the circuit senses too high a temperature in the Number 1 VDR module.

RADIO TUNING FUNCTION

The radio tuning function is done with the MCDU or the PFD and CCD. The primary tuning is done through the ASCB-D. There is also an emergency tuning mode. This mode lets the radio be tuned through a direct ARINC 429 link from the MCDU to the radio.



ICN-39-A-231000-G-A0126-00222-A-01-1

VHF ELECTRICAL SCHEMATIC DIAGRAM

SECTION 50 - AUDIO INTEGRATING AND VOICE COMMAND SYSTEMS

NUMBER 1 SYSTEM DESCRIPTION

The Number 1 airborne audio system includes these primary components:

- The Number 1 audio panel
- The Number 1 foot switch
- The relay K47
- The relay K48.

NUMBER 1 AUDIO PANEL

The Number 1 audio panel is installed in the left side of the interseat console. Four fasteners hold the Number 1 audio panel in correct position. In the rear side of the Number 1 audio panel there are installed two electrical connectors PL8P1 and PL8P2 that connect the panel to:

- The relay K47
- The relay K48
- The 1 Number 1 foot switch (S23)
- The Number 1 Modular Radio Cabinet (MRC1) (A7)
- The Number 2 Modular Radio Cabinet (MRC2) (A8)
- The headset (HT1).

The audio control panel has transmitter selections buttons, a liquid crystal display (LCD), the volume control, audio inputs, and the audio selection buttons.
The buttons are of the backlight type, and the knobs have a halo-type lighting that makes it easier to find them at night.
The audio panel has the buttons that follow:

- The Very High Frequency (VHF) COM
- The HF
- The Phone
- The HOIST
- The Public Address (PA)
- The no smoking (NO SMK)
- The seatbelt (SEATBLT)
- The authorization (AUTH)
- The NAV
- The Automatic Direction Finder (ADF)
- The Direction Finder (DF)
- The Distance Measuring Equipment (DME)
- The Marker (MKR)
- The Identification (ID) button
- The Voice (VCE) button
- The SELCAL button
- The cabin (CAB) button
- The HDPH button
- The Intercom Interphones (INPH) button
- The normal (NORM) and emergency backup (BKUP) volume

- The MIC (rectangle) button with the related audio AUD button (circle).

NUMBER 1 FOOT SWITCH

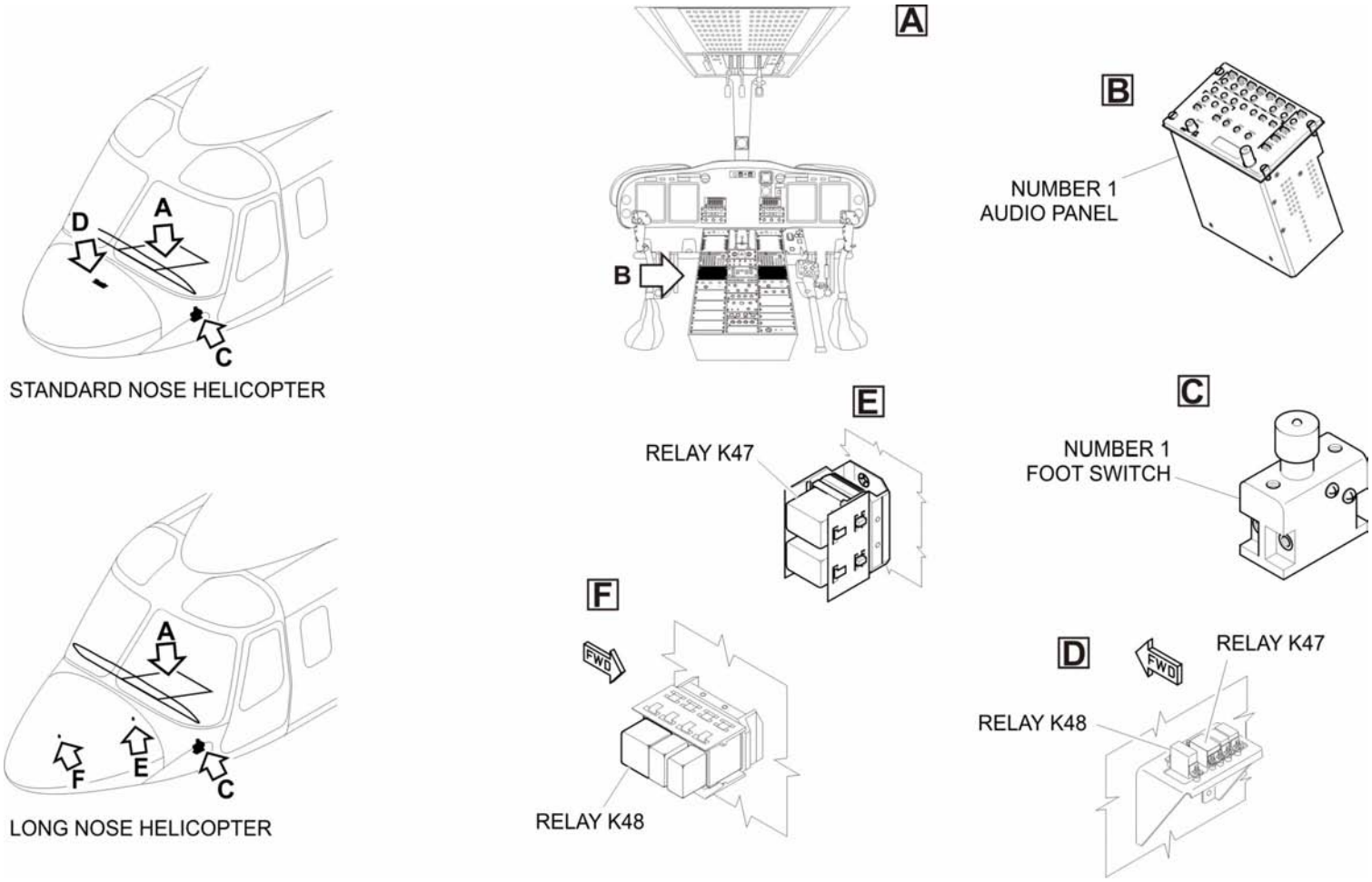
The Number 1 foot switch (S23) is installed in the floor at the left of the Copilot side. Two screws attach the Number 1 foot switch to the structure. The Number 1 foot switch has two electrical wires that connect its to the Number 1 audio panel.

RELAY K47

The relay K47 is installed on a support behind the instrument panel (standard nose helicopter) / in the left side of the nose bay (long nose helicopter). Four nuts and the four washers attach the relay K47 to the support. The relay K47 is connected to the Number 1 audio PL8 and to the MRC1 A7.

RELAY K48

The relay K48 is installed in a socket behind the instrument panel (standard nose helicopter) / in the right side of the nose bay (long nose helicopter). The relay K48 is connected to the Number 1 audio PL8, to the Number 2 audio panel PL24 and to the MRC2 A8.



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NUMBER 1 AIRBORNE AUDIO SYSTEM-PRIMARY COMPONENTS

NUMBER 2 SYSTEM DESCRIPTION

The Number 2 airborne audio system includes these primary components:

- The Number 2 audio panel
- The Number 2 foot switch

NUMBER 2 AUDIO PANEL

The Number 2 audio panel is installed in the right side of the interseat console. Four fasteners hold the Number 2 audio panel in the correct position. In the rear side of the Number 2 audio panel there are installed two electrical connectors PL24P1 and PL24P2 that connect the panel to:

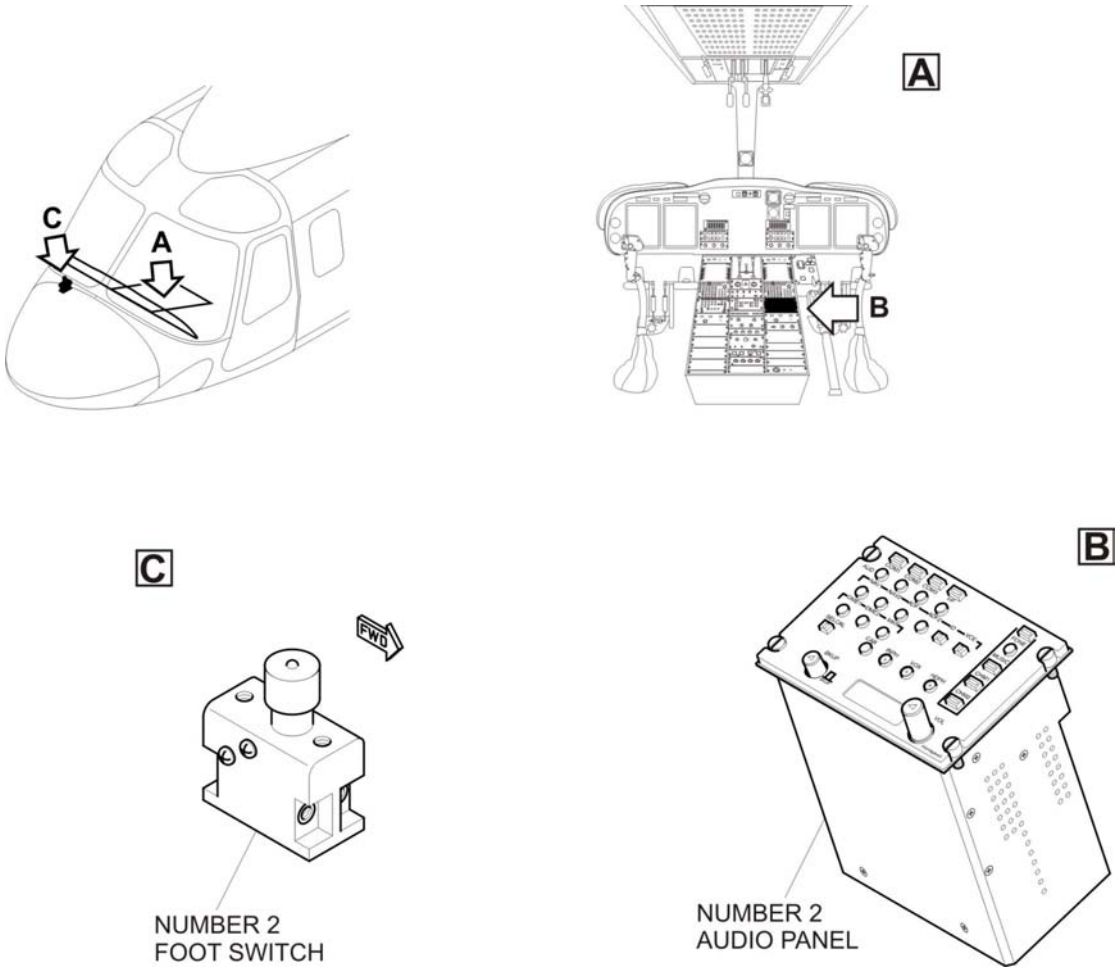
- The circuit breaker CB44
- The test panel (PL13)
- The relay K48
- The Number 2 foot switch (S24)
- The Number 2 Modular Radio Cabinet (MRC2) (A8)
- The headset (HT2)
- The Number 1 audio panel (PL8).

The audio control panel has transmitter selections buttons, a liquid crystal display (LCD), the volume control, the audio inputs and the audio selection buttons.
The buttons are of the backlight type, and the knobs have a halo-type lighting that makes it easier to find them at night.
The audio panel has the buttons that follow:

- The Very High Frequency (VHF) COM
- The HF
- The Phone
- The HOIST
- The Public Address (PA) routing
- The no smoking (NO SMK)
- The seatbelt (SEATBLT)
- The authorization (AUTH)
- The NAV
- The Automatic Direction Finder (ADF)
- The Direction Finder (DF)
- The Distance Measuring Equipment (DME)
- The Marker (MKR)
- The Identification (ID) and Voice (VCE) buttons that enable Morse Code and voice, or both for ADF and NAV audio
- The SELCAL
- The cabin (CAB), and HDPH and Intercom Interphones (INPH)
- The normal (NORM) or emergency backup (BKUP) volume
- The MIC (rectangle) button with related audio AUD button (circle).

NUMBER 2 FOOT SWITCH

The Number 2 foot switch (S24) is installed in the floor at the right of the Pilot side. Two screws attach the Number 2 foot switch to the structure. The Number 2 foot switch has two electrical wires that connect it to the Number 2 audio panel.



ICN-39-A-235000-G-A0126-00224-A-01-1

NUMBER 2 AIRBORNE AUDIO SYSTEM-PRIMARY COMPONENTS

AIRBORNE AUDIO SYSTEM OPERATION

The airborne audio system has two audio panels that are connect to the modular radio cabinets (MRCs), the navigation/communication (NAV/COM) unit and the high frequency communication (HF COM) unit. The airborne audio system lets the pilot and copilot control the audio functions of the helicopter independently.
The airborne audio system connects the MICs to helicopter radios and transmits audio signals from the related sources to the headsets. Also, it controls the interphone.
The primary interfaces of the audio system are the two digital audio buses from the MRCs (one from each MRC) and the two digital MIC buses.
The digital audio buses from each MRC contain multiplexed digital audio from the radios that are internal and external to the MRC. The two MIC buses connect to each audio panel and each MRC in the system and contain the same data.
The audio panels de-multiplex audio data, set the levels and change the set digital audio to analog signals. The analog signals go to the HDPH, speaker and cockpit voice recorder outputs. The MIC buses contain multiplexed digital audio samples. The MIC buses control data that is transmitted between the two audio panels and from the audio panels to the MRC.

COMPONENTS FUNCTION

AUDIO PANEL

The audio panel has transmitter selections buttons, a liquid crystal display (LCD), volume control, audio inputs, and audio selection buttons.
A transmitter selection is a MIC button (rectangle) that can have a related audio button (circle). The public address (PA) selection is connected to a third-party passenger address system.
Only one MIC transmit selection can be on at a time, but more than one audio selection can be set for listening. The transmitter selections start the related audio. For example, when the communication MIC is set, the communication audio starts.
The audio panel also gives basic telephone functions for the pick-up, hang-up, hold, transfer a call, and for conference calls. The selection of the telephone numbers is through the multifunction control display unit (MCDU).
Selective calling (SEL CAL) is an option. The SEL CAL function is contained in the audio panel, and the indication shows on the display system.
The Number 1 audio panel has the functions that follow:

- Supplies integrated control of all audio functions in the aircraft
- Receives digital audio data from the VHF NAV and VHF COM systems through a digital audio bus
- Changes digital data to analog data and sends it to the flight compartment loudspeakers and radio handsets
- Receives VHF NAV and VHF COM emergency audio signals and aircraft warning audio signals through analog audio buses
- Supplies an audio output for the cockpit voice recorder system.

The MRC has redundant circuits for radio audio and data. The MRC contains nonvolatile memory that is used to store radio and audio configuration data.
The configuration data is input through the audio panel for audio configuration.
Two independent audio circuits are contained in the MRC network interface module (NIM). Each audio function receives all audio signals from the radio modules contained in, or connected to, the MRC.
The audio data is changed to a digital audio bus output. Each digital audio bus output is connected to all audio panels. Radio systems, such as the NAV/COM or HF radios, input their analog audio into the NIM for transmission onto the digital audio bus.
The NAV/COM has an interfaces to the MRCs through the digital audio control bus and to the modular avionics units (MAU) through the Aeronautical Research, Inc. (ARINC) 429 bus.

AUDIO PANEL CONTROLS

1 MIC Buttons ..... push these buttons to connect the related input-channel to the MIC and arm the audio function of that channel.

2 AUDIO Buttons ..... push the necessary button to disengage the audio function of the related channel. Push the button again to engage the audio function.

3 SEL CAL ..... push this button to connect to the SEL CAL system. The function transmits a warning that can be heard and an engine-instruments and crew-alerting system (EICAS) message. The warning and the message are transmitted when one of the aircraft HF and VHF radios receive a set code of four letters. The system can monitor three VHF and two HF radios at the same time. It is not necessary that the pilot and copilot monitor the communication frequencies continuously.

4 BKUP (Norm or Bkup) ..... push this button to connect the MIC directly to the VHF COM transmitter. The headset connects to the VHF NAV/COM receiver and the HDPH volume control. All the MIC (transmit) buttons are released and all other functions and modes of the audio panel are off.

5 HOIST ..... Some helicopter operations make it necessary for a flight crew member to be on a cordless headset when the helicopter is on the ground or the hoist cable is used. Push this button to talk to the flight crew member on the cordless headset.

6 AUTH ..... this button lets the pilots control access to the radios in the cabin.

7 NO SMK, SEAT BELT ..... push these buttons to transmit a chime signal to the cabin speakers. When the NO SMK or SEAT BELT button is engaged, a discrete signal is sent which causes a visual indication to come into view in the cabin.

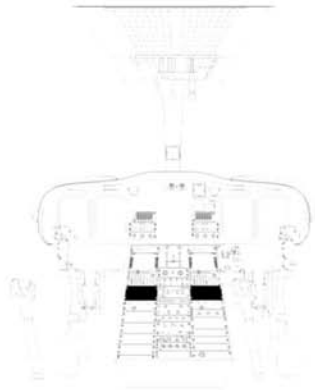
8 CAB ..... push this button to set and control the volume for the cabin intercom system.

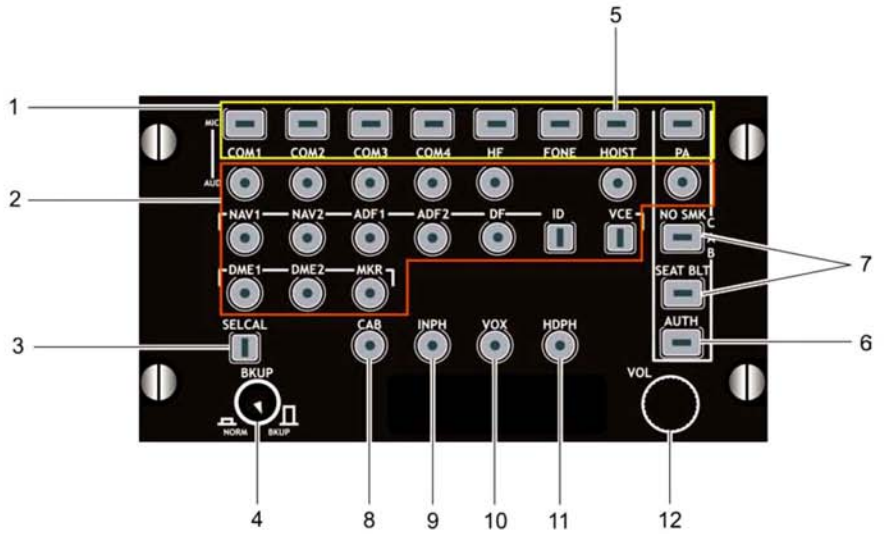
9 INPH ..... push this button to set and control the volume from the pilot's intercom system.

10 VOX ..... push this button to control the sensitivity of the VOX system.

11 HDPH ..... push this button to control the master volume of the pilot's headset.

12 VOL ..... Turn this knob to control the volume from the sources that the buttons control.





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AUDIO PANELS CONTROLS

LIQUID CRYSTAL DISPLAY (LCD)

The LCD shows status of the audio panel. The data that shows on the display include volume, blocked MIC indication and transmit indication.

BACKUP VOLUME

The NORM mode is set when the volume control knob is pushed in. The BKUP mode is set when the volume control knob is pulled out.
The backup mode is set if a failure of the audio control-panel power or of the digital audio bus.
The backup mode:

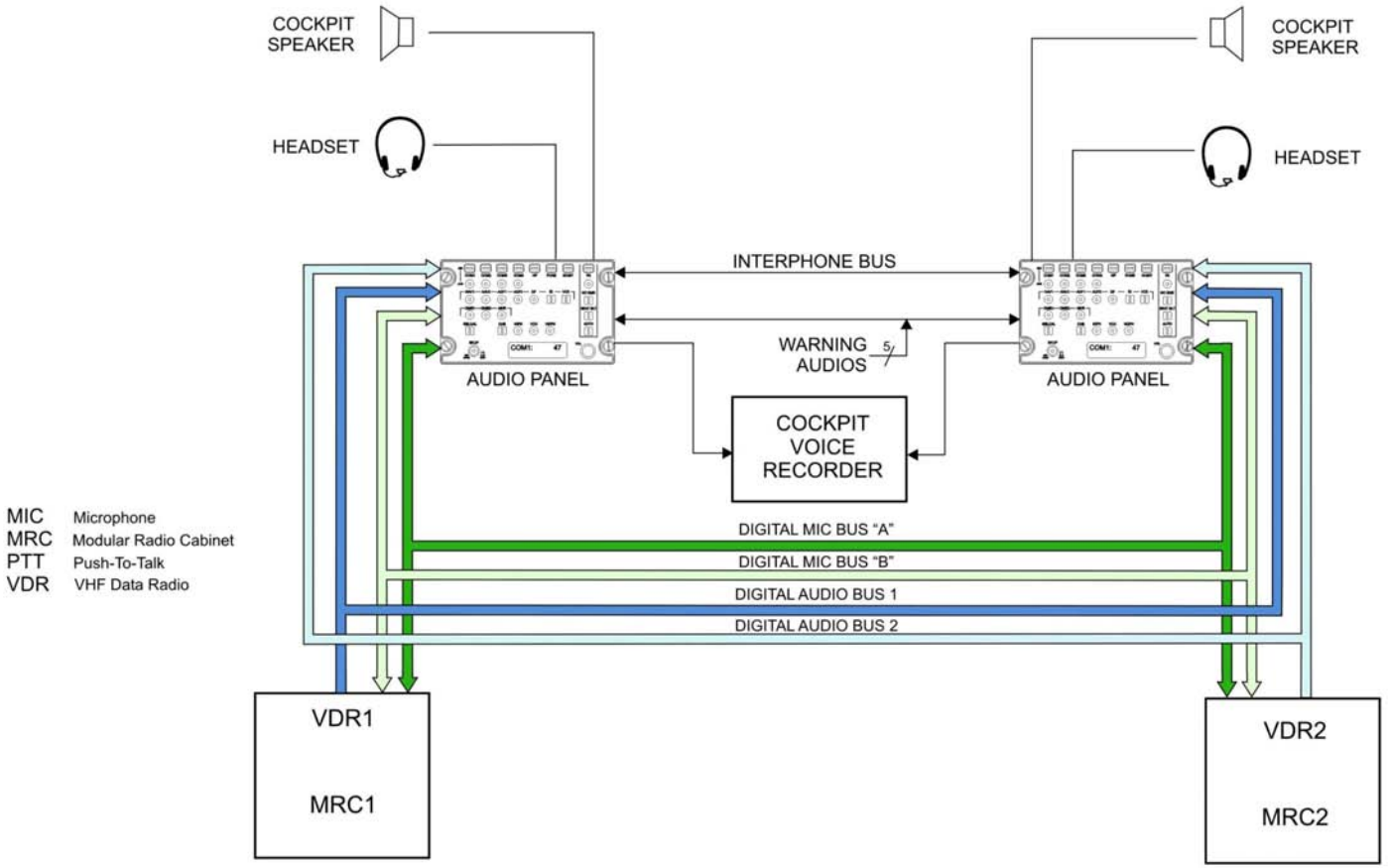
- Connects the emergency MIC to the VHF COM transmitter
- Connects the headset to the VHF NAV or VHF COM receiver and HDPH volume control
- Releases all the transmit buttons
- Disarms all the other functions and modes of the audio panel.

VOLUME CONTROL

The volume control adjusts the speaker volume and sidetone volume.
The default volume adjustment that shows in the LCD window is always the HDPH volume.
The default volume adjustment sets after a time-out of 18 seconds. During emergency operation, the speaker is not available.

WARNING AUDIO INPUTS

The warning audio inputs are transmitted to the two audio panel. All the inputs cannot be stopped and each input is independently adjustable.



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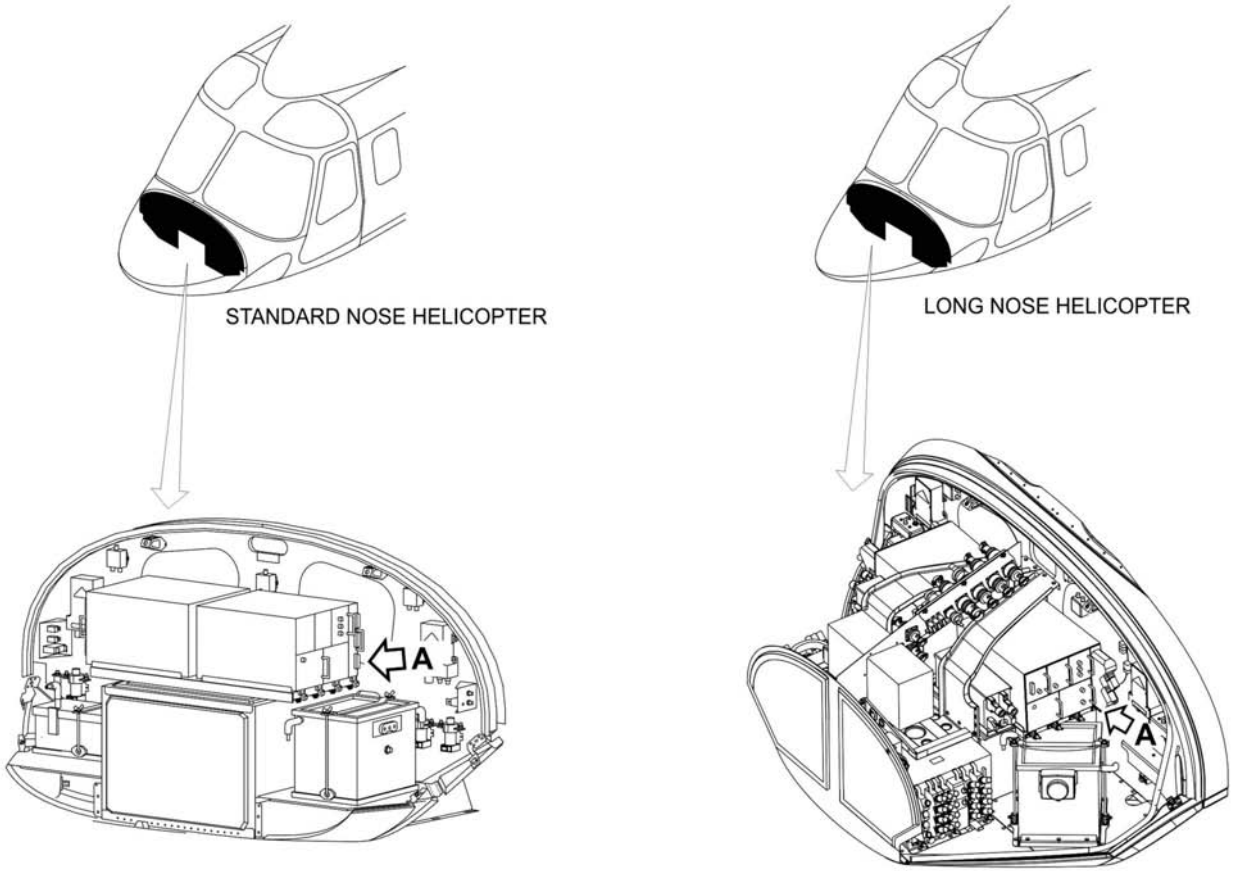
AUDIO SYSTEM SCHEMATIC DIAGRAM

SECTION 80 - INTEGRATED AUTOMATIC TUNING

MODULAR RADIO CABINET INSTALLATION

The Modular radio Cabinet (MRC) installation has the components that follow:

- The Number I Modular radio Cabinet (MRC1)
- The Number 1 Network Interface (NI1) module
- The Number 1 rear fan cover
- The Number 2 modular radio cabinet (MRC2)
- The Number 2 network interface (NI2) module
- The Number 2 rear fan cover.



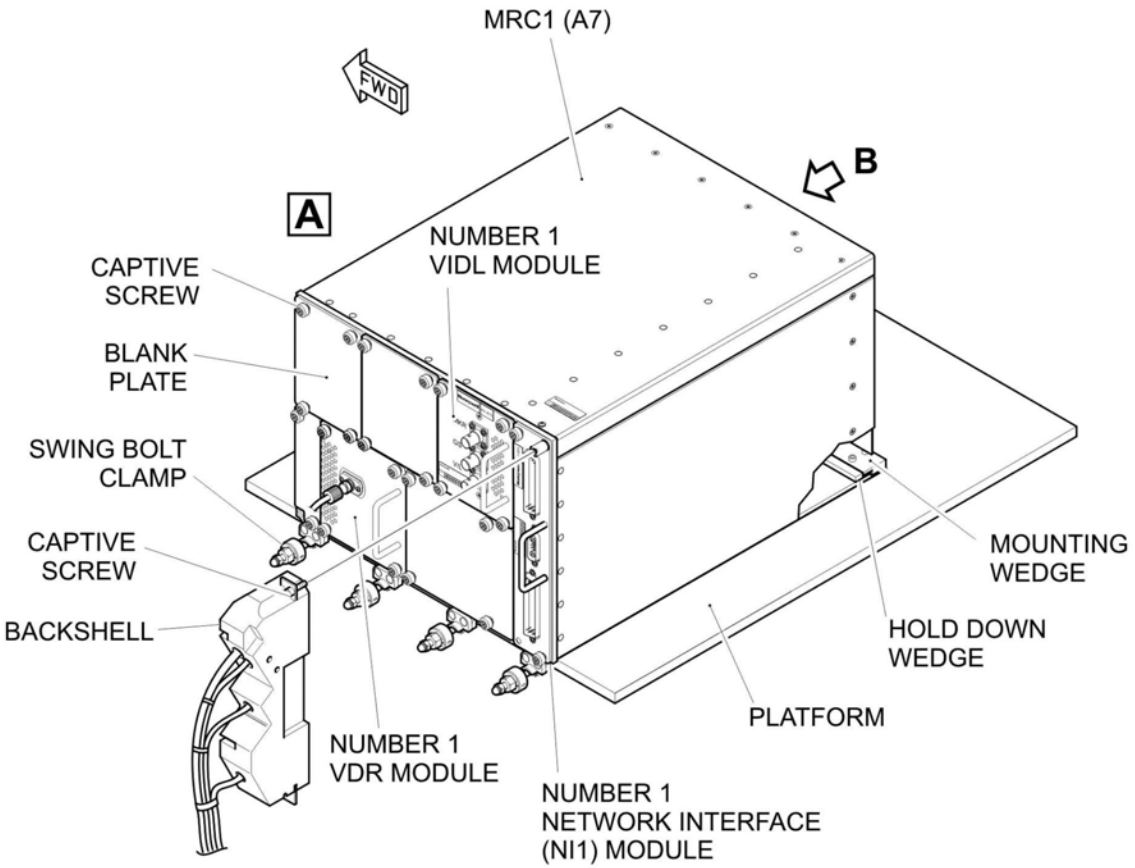
ICN-39-A-238000-G-A0126-00228-A-01-1

MODULAR RADIO CABINET 1 INSTALLATION (SHEET 1 OF 3)

NUMBER 1 MODULAR RADIO CABINET (MRC1)

The MRC1 is installed in the left side of the nose bay, between the STA 1500.00 and 1070.00. Four swing-bolt clamps hold the MRC1 in its position on the mounting platform. The mounting wedge holds the MRC1 to the structure of the mounting platform.
The MRC1 is a cabinet that contains multiple Line Replaceable Modules (LRMs) that do radio-communication functions. Each module has a self-contained power supply, Radio Frequency (RF) receivers/transmitters, signal processing circuits and all the other circuits necessary for its operation.
The radio modules also include a metal-enclosed circuit card, a front mounting plate, the backplane connectors and the front connectors.
The MRC1 also contains a Network interface module (NIM1).
The radio modules connect to the NIMs through the MRC backplane.
The MRC1 communicates to the rest of the PRIMUS EPIC system through the Avionics-standard Communication Bus, Version D (ASCB-D), and the local area networks (LAN).
A blank plate is installed on each empty slot of the MRC1.
The modules installed in the MRC1 are as follows:

- The NI1 module
- The Number 1 VDR module
- The Number 1 VIDL module



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MODULAR RADIO CABINET 1 INSTALLATION (SHEET 2 OF 3)

NUMBER 1 NETWORK INTERFACE (NI1) MODULE

The NI1 module is installed in the right side of the MRC1. Two captive screws attach the NI1 module to the MRC1. The NI1 module has a backshell. Two captive screws attach the backshell to the NI1 module.
A handle is installed on the NI1 module front to make its installation and removal easier.

NUMBER 1 VDR MODULE

The Number 1 VDR module is installed in the right bottom part of the MRC1. Four captive screws attach the Number 1 VDR module to the MRC1. A handle is installed on from of the Number 1 VDR module to make the module installation and removal easier.

NUMBER 1 VIDL MODULE

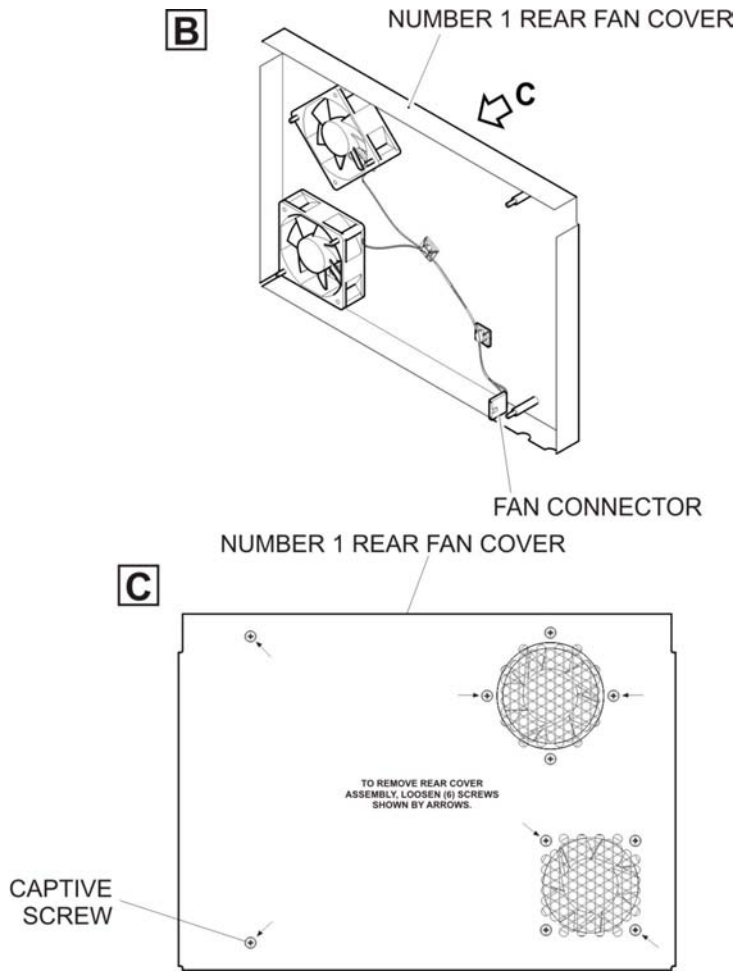
The Number 1 VIDL module is installed in the MRC1 between the NI1 and Number 1 VDR module. Four captive screws attach the Number 1 VIDL module to the MRC1. Three electrical connectors (A7-4P1, A7-4P2 and A7-4P3) connect the Number 1 VIDL module to:

- The Antenna (MRK) (E6)
- The Number 2 VOR/LOC antenna (E14)
- The Number 1 VOR/LOC antenna (E5).

A handle is installed on the front of the Number 1 VIDL module to make the module installation and removal easier.

THE NUMBER 1 REAR FAN COVER

The Number 1 rear-fan cover is installed on the rear side of the MRC1. Six captive screws hold the Number 1 rear-fan cover in its position on the MRC1. Two fans attached to the Number 1 rear cover of the MRC1 supply cool air to decrease the temperature of the NI1 module and radio modules. The NI1 module controls the operation of the fan.



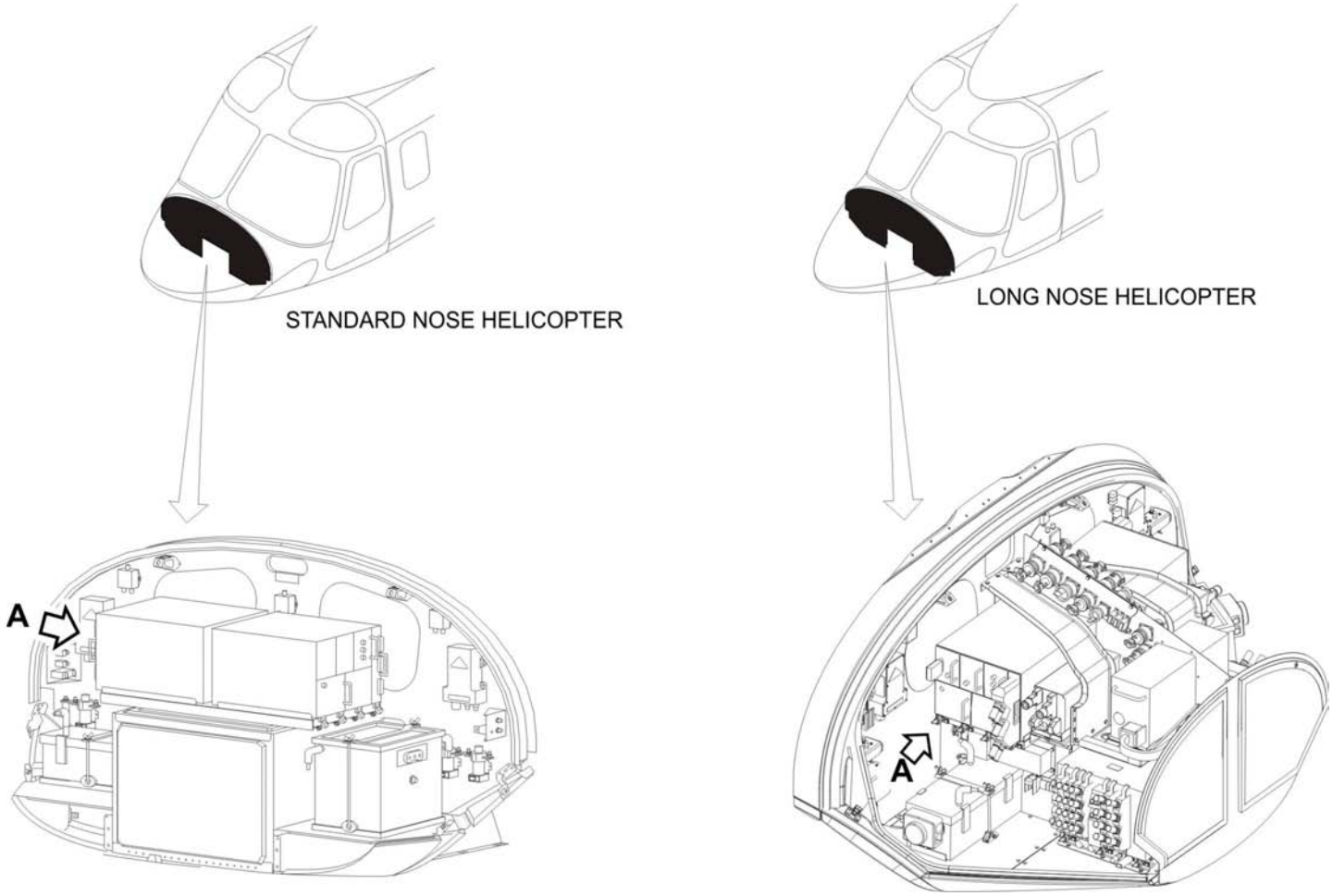
ICN-39-A-238000-G-A0126-00230-A-01-1

MODULAR RADO CABINET 1 INSTALLATION (SHEET 3 OF 3)

NUMBER 2 MODULAR RADIO CABINET (MRC2)

The MRC2 (A8) is installed in the right side of the nose bay between the STA 1500.00 and 1070.00. Four swing-bolt clamps hold MRC2 in its position on the platform. The mounting wedge holds the MRC1 to the structure of the mounting platform.
The MRC2 is a cabinet that contains five LRMs that do radio-communication functions. Each module has a self-contained power supply, Radio Frequency (RF) receivers/transmitters, the signal processing circuits, and all other circuits necessary for its operation.
The radio modules (LRM) have metal-enclosed circuit cards, a front mounting plate, the backplane connectors and the front connectors.
The MRC2 communicates to the PRIMUS EPIC system through the Avionics-standard Communication Bus, version D (ASCB-D), and the local area networks (LAN).
The modules installed in the MRC2 are as follows:

- The Number 2 Network Interface (NI2) module
- The Number 1 Automatic Direction Finder (ADF) module
- The Distance Measuring Equipment (DME) module
- The Number 2 VOR/ILS data link (VIDL) module
- The Number 2 VHF data radio (VDR) module
- The Air traffic control transponder (ATC XPDR) module.



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MODULAR RADIO CABINET 2 INSTALLATION (SHEET 1 OF 3)

NUMBER 2 NETWORK INTERFACE (NI2) MODULE

The NI2 module is installed in the right side of the MRC2. Two captive screws hold the NI2 module to the MRC2 module on the MRC1. The NI1 module has a backshell, two captive screws attach The NI1 module to the MRC1. A handle is installed on the NI1 module front to make its installation and removal easier.

NUMBER 1 AUTOMATIC DIRECTION FINDER (ADF) MODULE

The ADF module is installed in the top left part of the MRC2. Four captive screws attach the ADF module to the MRC2. The electrical connector A8-1P1 connects the ADF module to the ADF antenna (E8) and the electrical connector A8-1P2 connects the ADF module to the MRC2. A handle is installed on the front of the ADF module to make installation and removal easier.

DISTANCE MEASURING EQUIPMENT (DME) MODULE

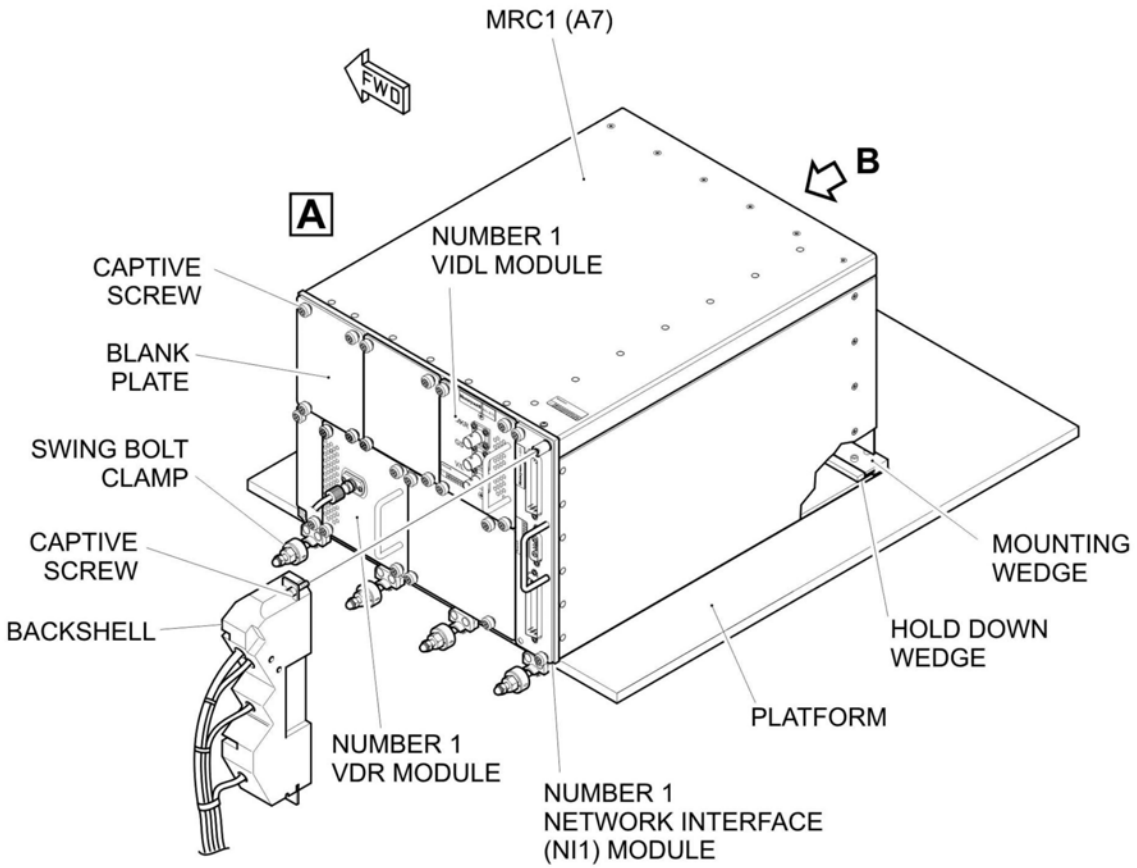
The DME module is installed on the right side of the MRC2 adjacent to the ADF module. Four captive screws attach the DME module to the MRC2. The electrical connectors A8-2P1 connects the DME module to the DME antenna (E9). A handle is installed on the front of the DME module to make the module installation and removal easier.

NUMBER 2 VOR/ILS DATA LINK (VIDL) MODULE

The Number 2 VIDL module is installed in the top side of the MRC2 between the DME and the NI2 modules. Four captive screws attach the Number 2 VIDL module to the MRC2. Three electrical connectors (A8-4P1, A8-4P2 and A8-4P3) connect the Number 2 VIDL module to the MKR antenna (E6) and to the MRC2 (A8). A handle is installed on the front of the Number 2 VIDL module to make the module installation and removal easier.

AIR TRAFFIC CONTROL TRANSPONDER (ATC XPDR) MODULE

The ATC XPDR module is installed in the bottom part of the MRC2 between the VDR and the NI2 modules. Four captive screws attach the ATC XPDR module to the MRC2. Three electrical connectors A8-4P1, A8-4P2 and A8-4P3 connect the ATC XPDR module to the top XPDR antenna (E13), to the bottom XPDR (E10) and to the MRC2 A8. A handle is installed on the front of the ATC XPDR module to make the module installation and removal easier.



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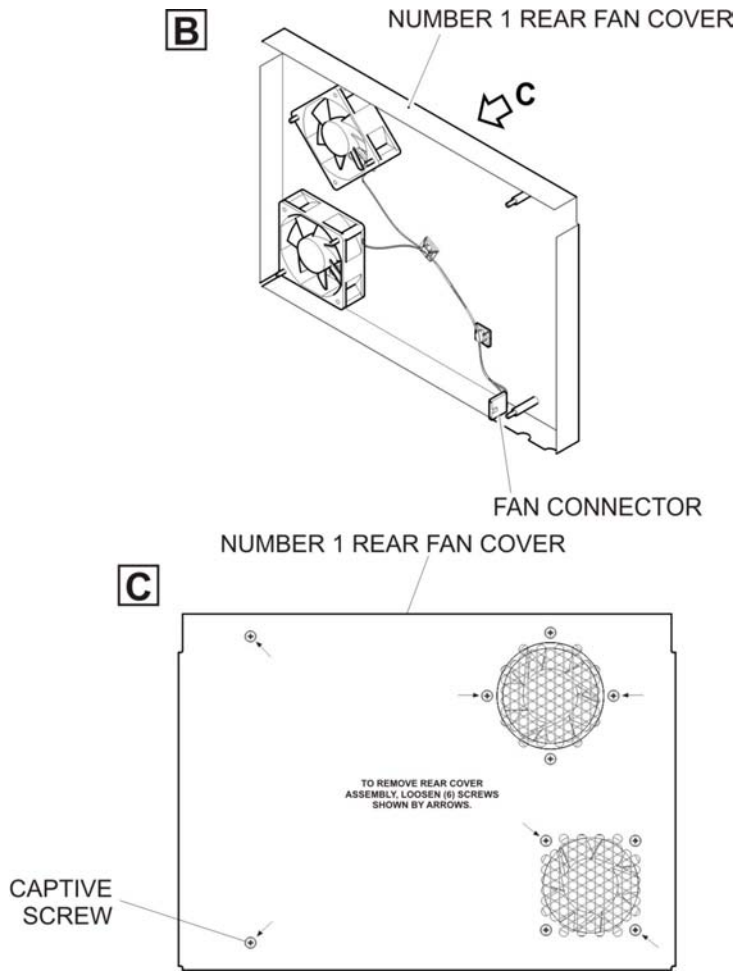
MODULAR RADIO CABINET 2 INSTALLATION (SHEET 2 OF 3)

NUMBER 2 VHF DATA RADIO (VDR) MODULE

The Number 2 VDR module is installed in the bottom left part of the MRC2. Four captive screws attach the Number 2 VDR module to the MRC2. The electrical connector E3-P1 connects the Number 2 VDR module to the Number 2 VHF-COM antenna (E6). A handle is installed on the front of the Number 2 VDR module to make the module installation and removal easier.

NUMBER 2 REAR FAN COVER

The Number 2 rear-fan cover is installed on the rear side of the MRC2. Six captive screws hold the Number 2 rear-fan cover in its position on the MRC2. Two fans attached to the Number 2 rear cover of the MRC supply cool air to decrease the temperature of the NI2 module and radio modules. The NI2 module controls the operation of the fans



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MODULAR RADIO CABINET 2 INSTALLATION (SHEET 3 OF 3)

MODULAR RADIO CABINET INSTALLATION

The radio modules installed in each MRC supply the system that follow:

- The Very High Frequency (VHF) Omnidirectional Radio Range/ Instrument Landing System (VOR/ILS)
- The Automatic Direction Finder (ADF)
- The Distance Measuring Equipment (DME)
- The VHF Communication system (COM)
- The VHF Navigation (NAV)
- The Diversity Mode S Transponder (XPDR).

The modular radio system also has an interfaces to the digital audio system. The cockpit controls are the Primary Flight Displays (PFD), the Multifunction Control Display Units (MCDU), the Cursor Control Devices (CCD), and the audio panels.
The modular radio system supplies COM and NAV data to the flight crew. The helicopter has two modular radio cabinets (MRC) identified MRC 1 and MRC 2. MRC 1 is installed on the left side and MRC 2 installed on the right side of the helicopter.
Each MRC has an Aircraft Personality Module (APM) for storage of installation options. The strap options are kept in A non-volatile memory (NVM) that can be uploaded through the central maintenance computer (CMC). Because the APM is part of the MRC motherboard, option selections are kept in the NVM after a radio module or NIM is replaced.
Each MRC communicates to the rest of the PRIMUS EPIC system through the avionics standard communication bus, version D (ASCB-D) and the local area networks (LAN).

SYSTEM OPERATION

The modular radio system does the navigation and communication functions related to the systems that follow:

- VOR
- ADF
- DME
- Instrument Landing System (ILS)
- Very High Frequency Communications (VHF COM)
- Air Traffic Control Transponder (ATC XPDR)

The NIM is the interface between the radio modules and the other components of the modular radio system.
The primary function of the NIM is a communication interface between the MRC radio modules and the LRUs that are connected on the ASCB-D. This interface is for radio tuning, radio data, and CMC functions.
The NIM moves the data between the ASCB-D and each of the radio modules installed in the MRC. Each radio module transmits to and receives from the NIM through isolated serial receive (RX) and transmit (TX) data buses (Radio Communication Buses-RCB).

NUMBER 1 NETWORK INTERFACE (NI1) MODULE

The NI1 module is a Line Replaceable Module (LRM) in each MRC that includes four main section:

- The NIC CPU section
- The radio CPU section
- The low voltage power supply section
- The temperature monitoring section

Through these sections the NIM does the primary functions that follow:

- Interface between the MIC radio modules and the helicopter buses and LAN
- Processing and transmission of radio commands and signals

- Power supply to the radio modules
- Temperature monitoring.

The CPU section has a processor that can run the Honeywell Digital Engine Operating System (DEOS). Also, it is an interface between the radio modules in each MRC and the other LRUs in the PRIMUS EPIC system.
The interface with the PRIMUS EPIC system is through the ASCB-D and the LAN. The NIM collects the data from the ASCB-D and LAN and supplies them to the radio modules through the radio communication bus (RCB). The NIM also collects the data from the radio modules through the RCB, and supplies them to the ASCB-D and LAN.
The transmitted and received data include:

- The control parameters and status data to and from the radio modules
- The frequency, channel, code, and mode control data to the radios.
- The NAV and status data to the avionics system.

Another function of the NIM is to monitor the data that are transmitted across the digital buses and make sure that the buses operate correctly.
The digital audio bus from each MRC connects to each audio panel in the cockpit. This lets the pilot and copilot control the two sides of the audio independently.
The radio CPU section supplies an RCB (ARIN 429) interface and digital audio. It processes the commands from the ASCB-D, for example the radio tuning commands, and formats all radio module data for transmission on the ASCB-D.
The NIM also does the analog-to-digital conversion of the radio audio signals transmitted on the digital audio bus. And it receives the analog audio signals from the internal and external radios. The NIM digitizes and multiplexes the data on a single digital audio bus for transmission to each audio panel. The digital audio bus operates at a 1-MHz bit rate.
The NIM has a sensor that monitors its temperature and receives the temperature data of the other radio modules in the MRC through the RCB. The NIM uses the temperature data to control the two MRC cooling fans, and thus keep the MRC temperature within the limits.

The NIM connects internally to the radio modules through the MIC backplane and a RS-422 serial connection.

OPERATION WITH OTHER SYSTEM

The modular radio system operates with the other systems and components that follow:

- The NV-875 VOR/ILS Data Link (VIDL) Receiver Module
- The DF-855 ADF Module
- The DM-855 DME Module
- The TR-865A VDR Module
- The XS-856A Air Traffic Control (ATC) XPDR Module
- The VOR Antenna, one is necessary for each VIDL module
- The Glideslope (GS) Antenna, one is necessary for each VIDL module
- The Marker (MKR) Beacon Antenna, one is necessary for each VIDL module
- The ADF Antenna, one is necessary for each ADF module
- The DME Antenna, one is necessary for each DME module
- The VHF COM Antenna
- The ATC XPDR Antenna (A second ATC XPDR antenna can be installed for each module for diversity).
- The Modular Avionics Units (MAU) with LRMs
- The PFDs
- The MCDUs
- The CCDs
- The audio panels

- The microphones
- The headsets.

VIDL MODULE

The VIDL module contains a VOR/localizer (LOC) receiver, GlideSlope (GS) receiver, and marker (MKR) beacon receiver. The VIDL module is part of the VIDL system.

ADF MODULE

The ADF module contains an ADF receiver that supplies en-route, terminal navigational and area guidance. The ADF module is part of the ADF system.

DME MODULE

The DME module contains a DME receiver that supplies en-route, terminal navigational and area guidance. The DME module is part of the DME system.

VDR MODULE

The VDR is a two-way VHF COM transceiver (transmitter and receiver) that supplies:

- The voice communications
- The low-speed minimum shift keying (MSK) ACARS Data
- The high-speed data modes related to differential 8-phase shift keying (D8PSK) modulation techniques.
- The VDR module is part of the VDR system.

XPDR MODULE

The XPDR module contains a transceiver that does the functions that follow: air traffic control radar beacon system (ATCRBS), Mode S and diversity XPDR. The XPDR module is part of the ATC XPDR system.

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CHAPTER 24

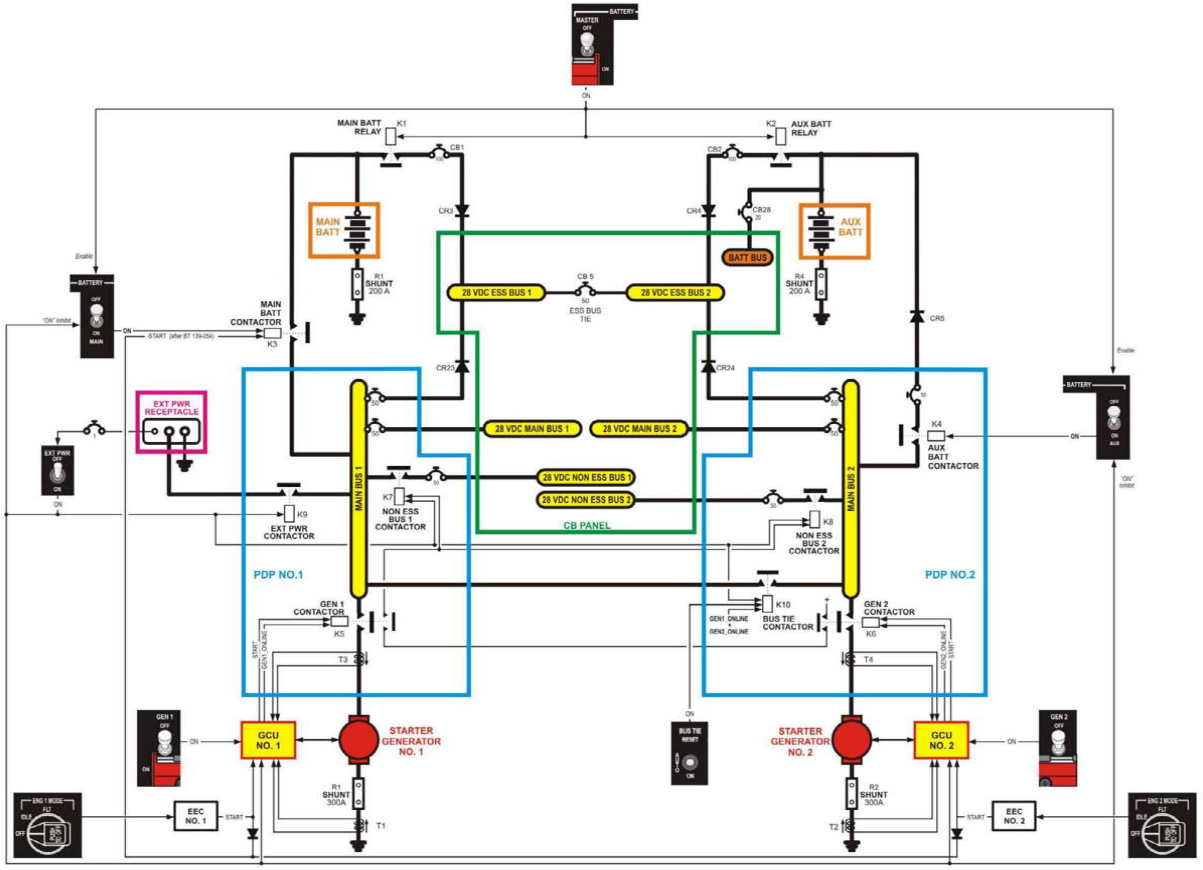
ELECTRICAL POWER

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SECTION 00 - ELECTRICAL POWER

DESCRIPTION

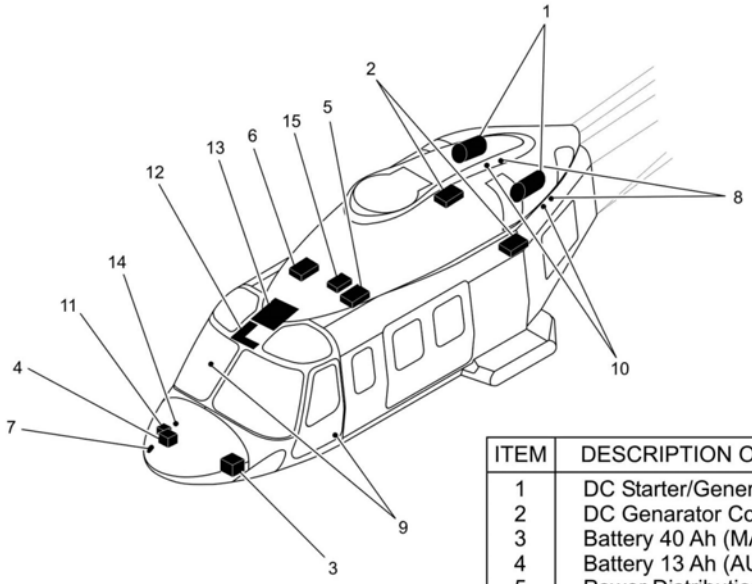
The DC Electrical Power system generates and distributes electrical power to the rotorcraft systems via power distribution panels (PDP).
The DC Electrical Power system main items are two starter-generators, two batteries and an external power supply.
The DC power generators are driven directly from the turbo shaft engine; the generators are mounted on each engine.
The DC power is generated by two 300 A DC starter-generators and controlled by Generator Control Units (GCUs).
The 28 VDC is distributed via the POWER DISTRIBUTION PANEL N.1 and N. 2 to the DC circuit breaker panel located in the overhead panel of the cockpit.
Priority of the loads is determined by the bus to which they are connected.
In order of importance, there are the Battery Bus, the Essential Buses, the Main Buses and the Non-Essential Buses.
Two batteries, provide the self-contained electrical reserve and they are rated at 40 Ah standard or 44 Ah optional (MAIN BATT), and 13 Ah standard or 27 / 28 Ah optional (AUX BATT).
When external power is not available, the MAIN BATT provides power to the starter-generator while the AUX BATT supports the essential bus loads during the start sequence.
During starting with EXT PWR (external power) the MAIN BATT assists the EXT PWR.
The two batteries are also used to provide the power to rotorcraft essential electrical loads in the event of a total electrical generation failure during flight.
The batteries are automatically disconnected when external power is applied to inhibit charging.
The DC generation system controls are located in the overhead console.
The DC electrical power system indications are provided on the display units.



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DC ELECTRICAL POWER SYSTEM-BLOCK DIAGRAM

(Printed A3 format at the end of data module)



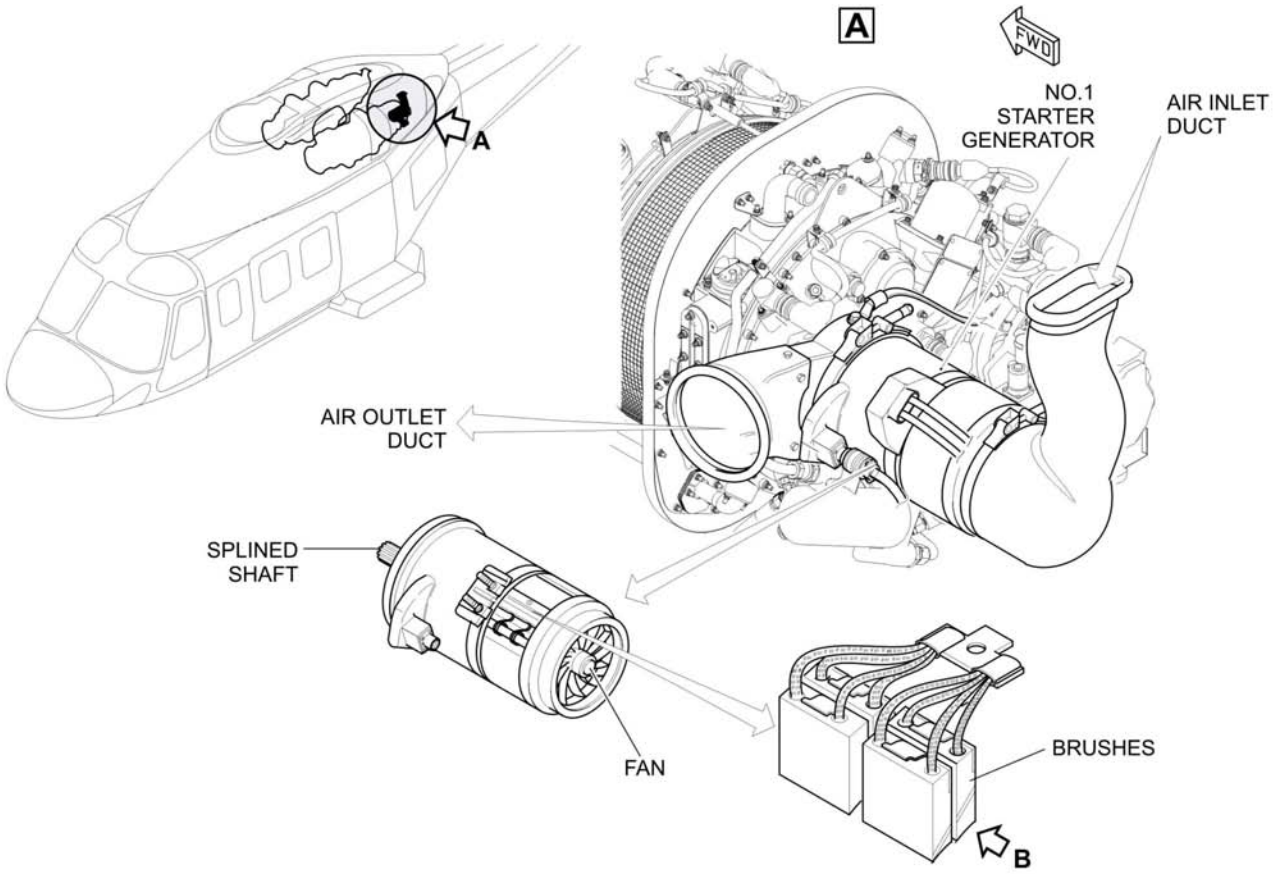
ITEM	DESCRIPTION OF ITEM	LOCATION	QTY
1	DC Starter/Generator (300A)	Engine Bay	2
2	DC Genarator Control Unit	Aft Avionic Bay	2
3	Battery 40 Ah (MAIN Battery)	RH Nose Compartment	1
4	Battery 13 Ah (AUX Battery)	LH Nose Compartment	1
5	Power Distribution Panel #1	LH Cabin Roof	1
6	Power Distribution Panel #2	RH Cabin Roof	1
7	External PWR Receptacle	RH Nose	1
8	DC Current Transformer	LH - RH avionic bay and PDP1 - PDP2	4
9	200 ADC Current Shunt (Battery)	Cockpit Underfloor	2
10	300 ADC Current Shunt (DC Gen.)	LH and RH Avionic bay	2
11	Control Panel	Cockpit	1
12	Circuit Breaker Panel	Cockpit	1
13	Diode	RH Nose Compartment	1
14	Diode Module	Cabin Roof	2

ICN-39-A-240000-G-A0126-00294-A-01-1

DC ELECTRICAL POWER COMPONENTS LOCATION

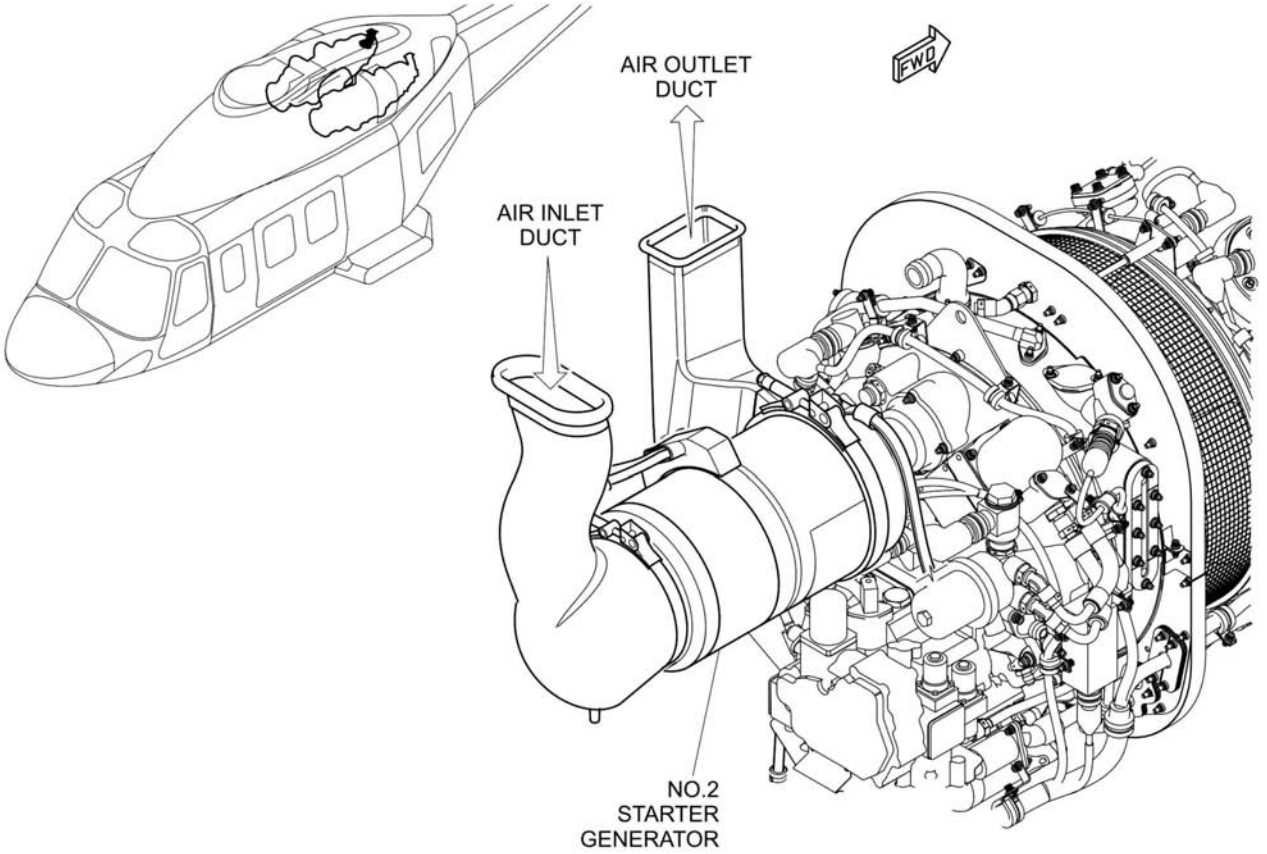
DC STARTER GENERATOR

Two DC starter-generators are electromagnetic rotating machines, capable of converting the engine input, mechanical torque, into output DC electrical power when operating as a generator.
The generator is regulated by the relevant GCU at 28 VDC and is capable of producing a max continuous output of 300 A.
In starter mode, the machine is capable of converting input DC electrical power into a mechanical output torque.
The starter-generator is constructed of a cylindrical shaped housing within which are mounted the stator and rotor assemblies.
The housing is constructed of a high iron content material in order to serve as part of the magnetic circuit for the starter assembly.
The rotor assembly consists of a rotor winding that interfaces with the fixed input terminals on the machine housing through a brush/commutator assembly.
The starter-generator is brush-type: four brush block assembly, each containing four brushes to input or output current to from the rotor windings.
The starter generator is coupled to the engine accessory gearbox via the splined shaft which incorporates a shear section.
The bearings are pre lubricated with grease at installation and require no further service until overload.
The starter-generator incorporates one fan that is drove by the generator rotor shaft, provides a forced air flow through the air inlet duct and towards the input spline, exhausted trough the air outlet duct.
This air flow also serves to cool the bearings supporting the rotor and maintain acceptable operating temperatures.
The generator temperature is monitored by a bi-metallic temperature switch embedded in the stator assembly.
In the event that the switch detects an over temperature condition, the bi-metallic element closes a set of contacts and provides loop closure to indicate the over temperature status.
The starter-generator also incorporates a speed sensor on the drive shaft, whose signal is routed to the GCU which interrupts the start cycle at a determined engine rpm in engine manual mode.



ICN-39-A-240000-G-A0126-00295-A-01-1

NO.1 STARTER-GENERATOR LOCATION



ICN-39-A-240000-G-A0126-00296-A-01-1

NO.2 STARTER-GENERATOR LOCATION

GENERATOR CONTROL UNIT (GCU)

Two GCU (GCU1 and GCU2) are used to control, protect and monitor the relevant starter generators.
Control functions include voltage regulation (generator mode), field weakening (starter mode) and line contactor control.
Protection functions include over voltage, under voltage, reverse current, over current (short circuit).
Monitoring functions include starter generator speed and external power presence.
The GCU's are installed in the after Avionic bays
Each GCU terminate the Manual start cycle of the relevant engine when the starter speed has exceeded the equivalent of 49 to 50 % Ng.

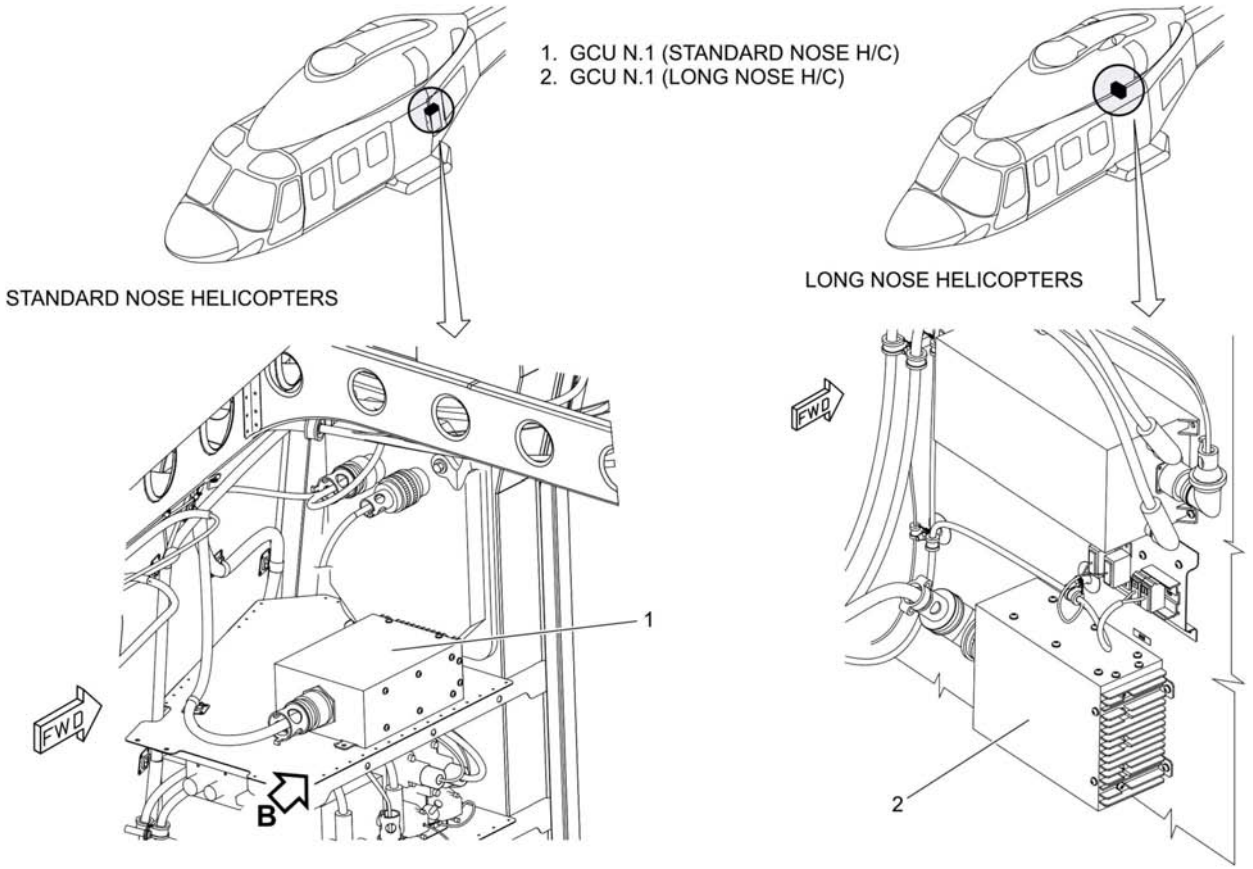
CONTROL FUNCTIONS

STARTER MODE

The GCU in combination with the starter-generator is designed to permit operation of the starter-generator as a starter motor. Upon receiving a start command from the engine control panel, the GCU activates the field winding and closes the line contactor to apply 28 V DC power to the starter-generator. The GCU applies full field at the initiation of the start to allow the starter-generator to build sufficient torque to accelerate the engine turbine, however, as the speed of the generator increases, the back EMF (Electro Motive Force) increases causing the machine to become a generator and as such present a load on the engine turbine. To avoid this effect and allow the starter generator to apply a positive torque throughout the start cycle, GCU reduces the field current to limit the back EMF to a level that allows rated current to flow into the starter-generator as the speed increases.
During Manual Starting the GCU monitors the speed signal produced by the Starter-generator and terminates the start cycle when the speeds signal indicates that the engine has reached 49 to 50 % Ng. At the termination of the start, the GCU opens the line contactor and de-energizes the starter-generator field winding.

GENERATOR MODE

Once the generator voltage is established, the GCU internal field build up circuit is interrupted by a field control relay and then the regulating field output is switched into the circuit and supplied by the voltage regulator. The voltage regulator is a closed loop control circuit that adjusts the duty cycle of the exciter field voltage to provide a constant voltage at the Point of Regulation (POR) under varying load conditions.
When the generator output attains voltage level within 0.5 V of the bus voltage, the GCU commands the line contactor to close connecting the generator output to the bus. In the event that the buses have been manually tied from the flight deck control, a signal is routed via the bus tie contactor to each GCU in the DC electrical power system to load share the generator output within 10% of rated load between 25% and 100% rated load.



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GENERATOR CONTROL UNIT 1 – LOCATION

PROTECTION FUNCTIONS

OVERVOLTAGE

The GCU provides the starter-generator to be disconnected from the load bus and to be de-energized before the voltage exceeds the limits.

UNDERVOLTAGE

The GCU provides for de-energizing and disconnecting the start-generator from the load bus when the voltage falls below 5 V. This is co-ordinated with the over current function.

FEEDER FAULT

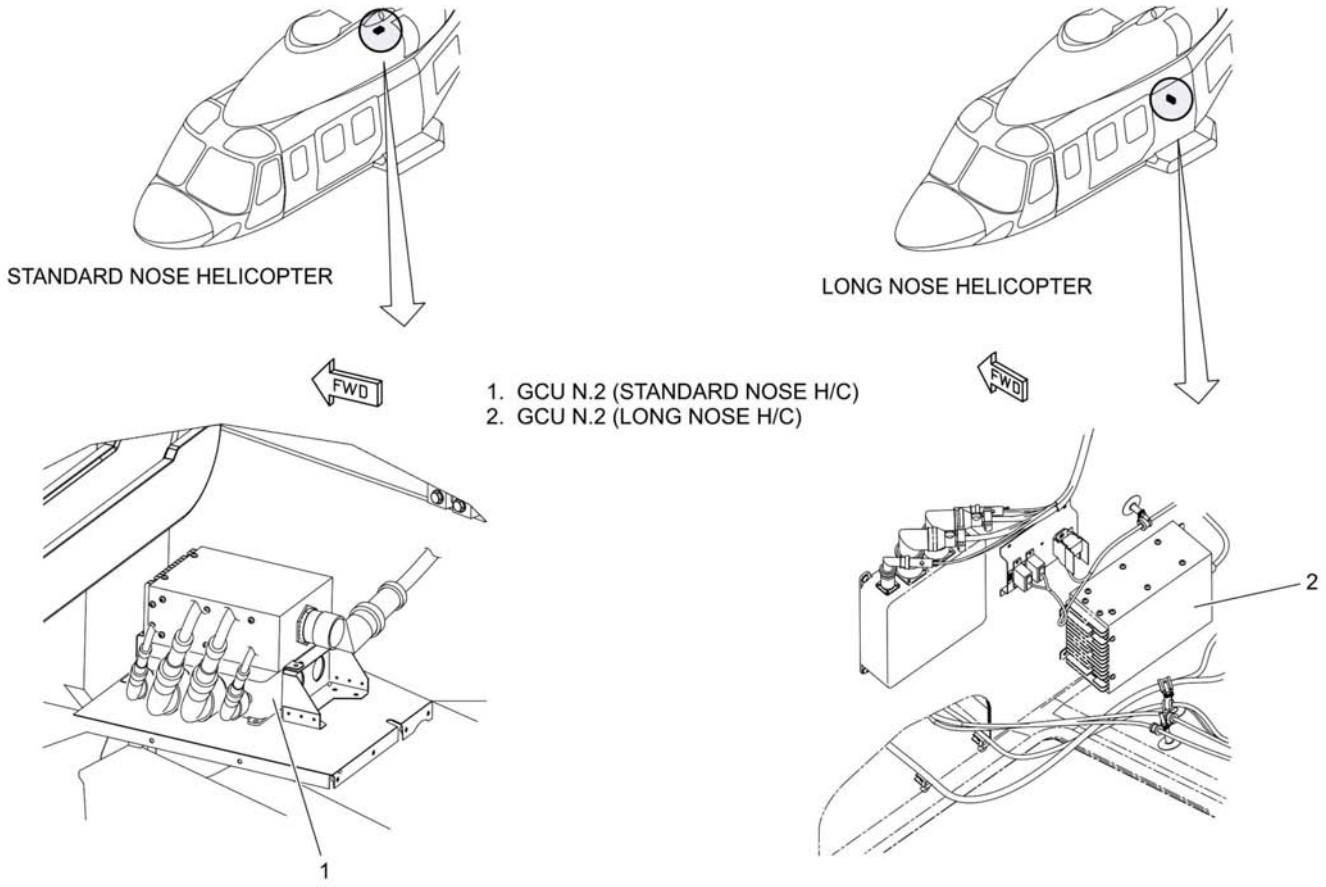
The GCU provides for de-energizing and disconnecting the start-generator from the load bus when the differential feeder current exceeds 125 A nominally.

BUS FAULT

The GCU detects the occurrence of an over current condition in excess of 550 Amps during Generator Mode operation, and provides a signal to be used to sequence fault clearance.

REVERSE CURRENT

The GCU provides for de-energizing and disconnecting the start-generator from the load bus when it detects a reverse current which exceeds 10% of rated load.



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GENERATOR CONTROL UNIT 2-LOCATION

MONITORING FUNCTIONS

START COMMAND INPUT

The GCU incorporates a pin on the interface connector used to receive the start command from the engine control system and configure the GCU functions to perform engine starting.

START TERMINATE SIGNAL

The GCU provides an output from the Interface connector. The signal is used by the Starting panel to unlatch the starting functions and terminate the starting sequence. The signal is issued when the starter-generator has attained 50-55% speed in engine manual mode.

EXTERNAL POWER INHIBIT

In order to prevent the paralleling of DC External Power and DC Generator outputs the GCU incorporates a pin on the interface connector to receive a signal indicating that DC External Power is active on the rotorcraft. If the GCU receives such a signal when the generator is not connected to the bus, the GCU will inhibit closing of the line contactor while the external power signal is present. If the generator is already connected and an external power signal is received, the GCU will de-energize the line contactor, disconnecting the generator output from the bus.

POWER READY SIGNAL

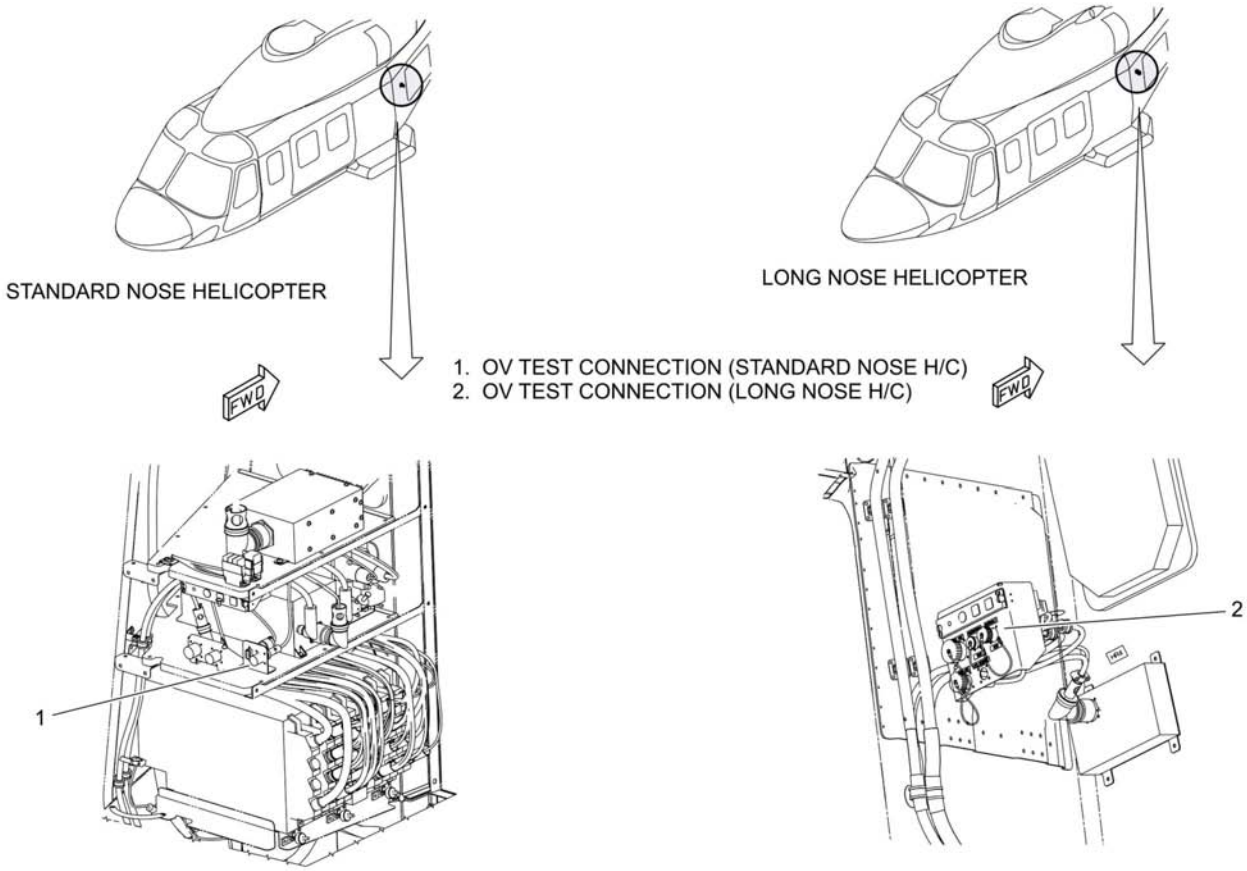
The GCU provides an output from the interface connector that indicates the condition of the GCU and starter-generator.

POWER NOT-READY SIGNAL

The GCU provides an output from the interface connector, that during start, indicates the generator is not available or in the start mode.

OV TEST

The GCU incorporates a pin on the interface connector to accept an external signal that is used to test the functioning of the GCU over voltage protection. The external signal is provided by Ground Support Equipment. The function is exercised during scheduled ground maintenance.

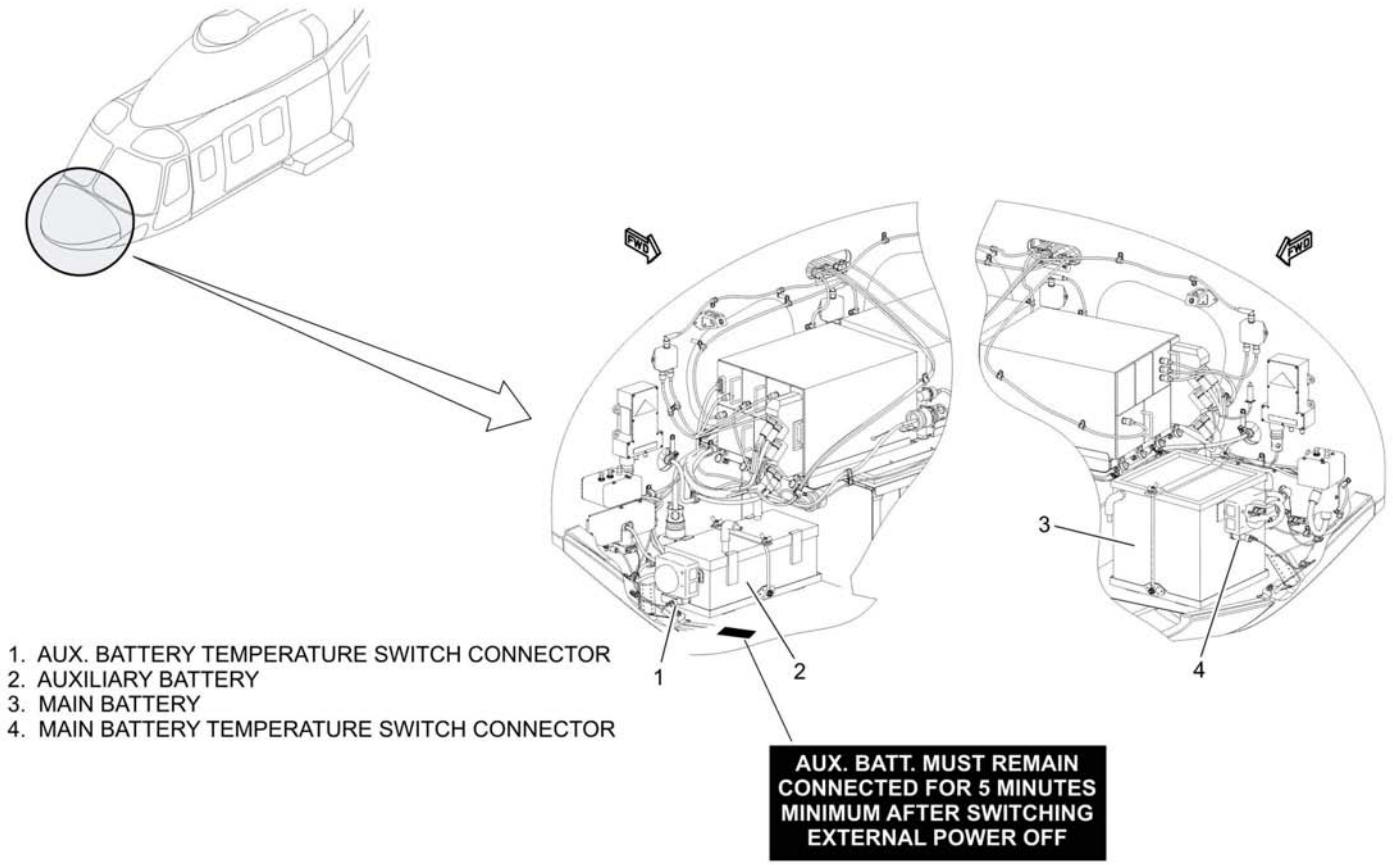


ICN-39-A-240000-G-A0126-00299-A-01-1

OV TEST CONNECTION LOCATION

MAIN AND AUX BATTERIES

The Main (40 Ah), installed in the left side of the nose compartment and Auxiliary (13 Ah) , installed in the right side of the nose compartment, batteries are an assembly of individual cells (20 total) contained by steel case within which each individual cell is connected in series to achieved a nominal terminal voltage of 24 V.
Cells interconnection is achieved using solid copper bus bars that are bolted to the individual cell positive and negative terminals.
The battery incorporates means to mount the battery to the airframe through secure tabs that form part of the battery case cover.
The case is essentially a steel box with the cover removable in order that individual components within the battery can be serviced.
The battery case incorporate vent ports with the provision to attach suitable tubing in order to route some outgasing of hydrogen to a containment vessel provided by the airframe and overboard.
The battery (NiCad) is based upon nickel and cadmium plates with a potassium hydroxide electrolyte.
This type of battery is common in aircraft and offers a high power with low maintenance.
A characteristic of NiCad battery is its relatively flat voltage curve under discharge conditions, thus supplying to aircraft services a constant potential through the majority of the discharge cycle.
Since the batteries on the AW139 are also used as a self contained power supply for engine starting, the high current capability and superior low temperature performance are important assets.
Inside the battery a temperature sensor is placed.
This bi-metallic temperature switch indicates when the battery exceeds a preset temperature limit.



1. AUX. BATTERY TEMPERATURE SWITCH CONNECTOR
2. AUXILIARY BATTERY
3. MAIN BATTERY
4. MAIN BATTERY TEMPERATURE SWITCH CONNECTOR

ICN-39-A-240000-G-A0126-00300-A-01-1

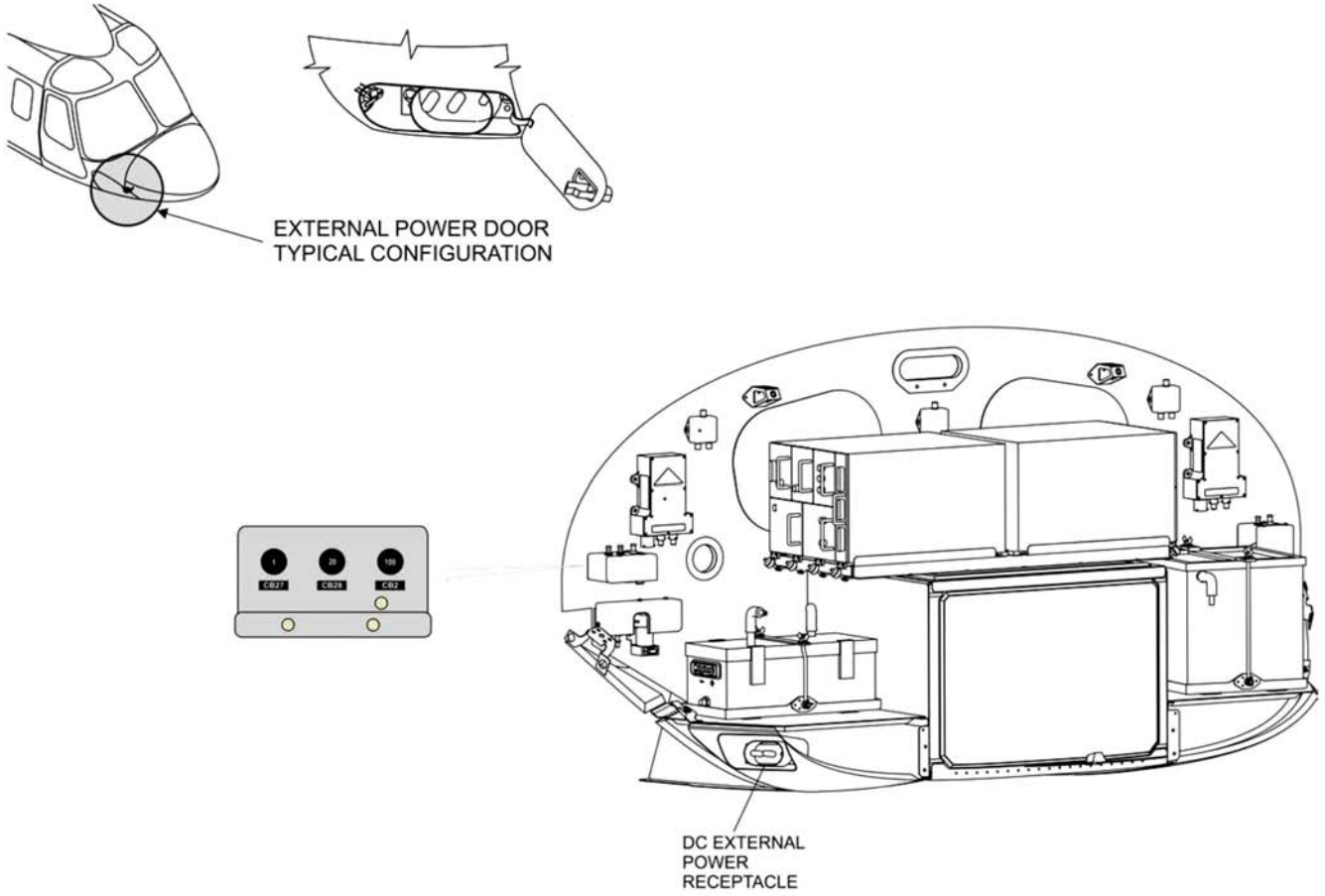
MAIN AND AUXILIARY BATTERY LOCATION

EXTERNAL POWER

The external power connector is of the tripolar type and the receptacle uses a large positive pin, a large ground pin and a small positive control signal input pin.
The large positive pin is connected to the external power contactor which, when energized, supplies external DC power to the aircraft bus bars.
The small positive control signal input pin, connected to the external power contactor coil, energizes the contactor if the external DC source parameter is within the range when the EXT PWR is set to ON.
When the external power over voltage (29.7 VDC) is detected, the O/V trip relay energizes removing power to the external power contactor that opens.
The external power is so disconnected from the aircraft.
This condition causes the trip of circuit breaker CB27 (located near the AUX battery).
To reset, the EXT PWR has to be lowered to its proper voltage and the CB27 reset.

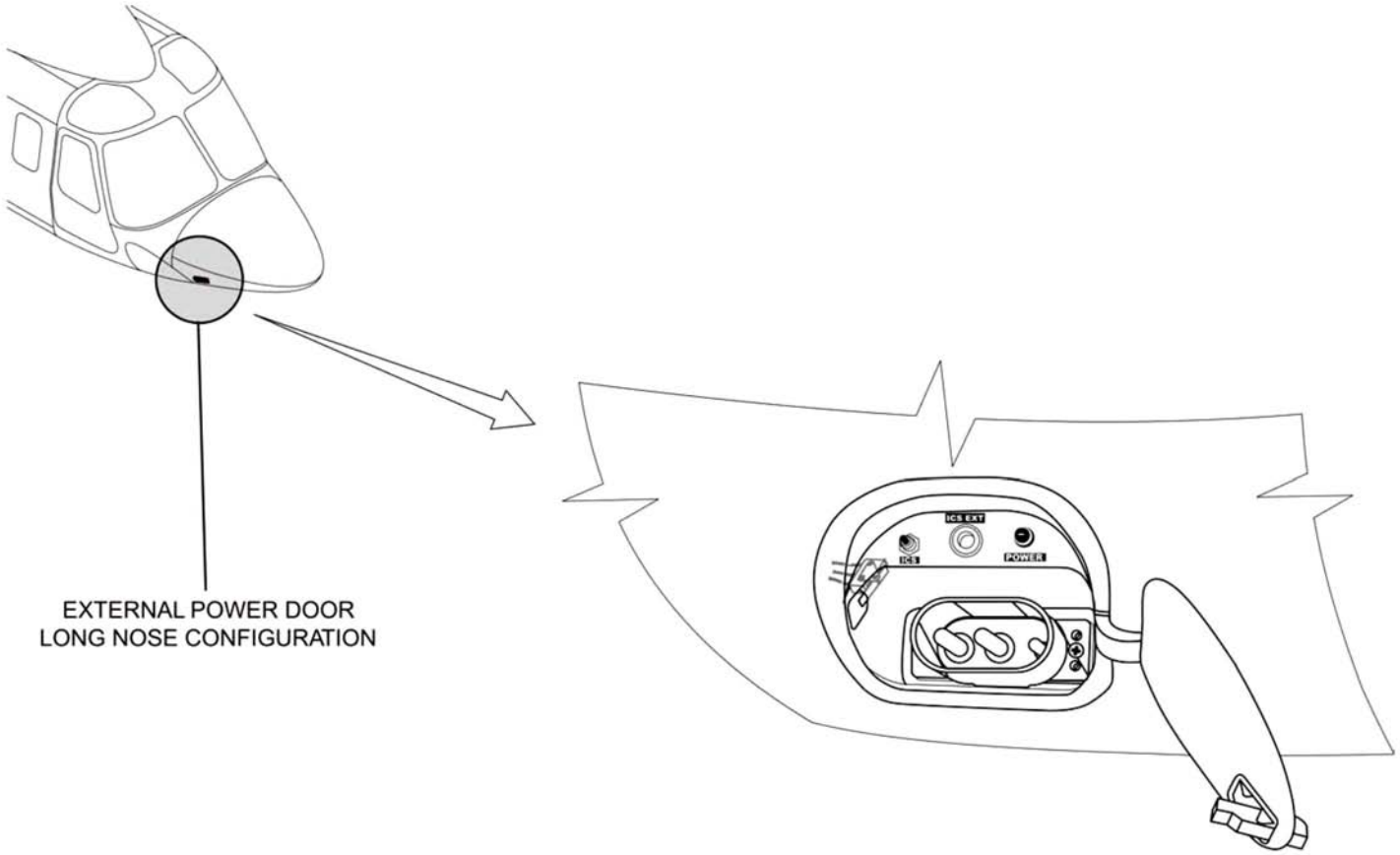
NOTE A

The 29.7 VDC is equivalent to 30.5V nominal at receptacle input pin allowing for line voltage drops.



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EXT PWR RECEPTALCE & CB27 [1J]



ICN-39-A-240000-G-A0126-02911-A-01-1

EXT PWR RECEPTACLE & CB27 [1L]

DC ELECTRICAL LOAD DISTRIBUTION

Primary Components

The DC electrical load distribution includes these primary components:

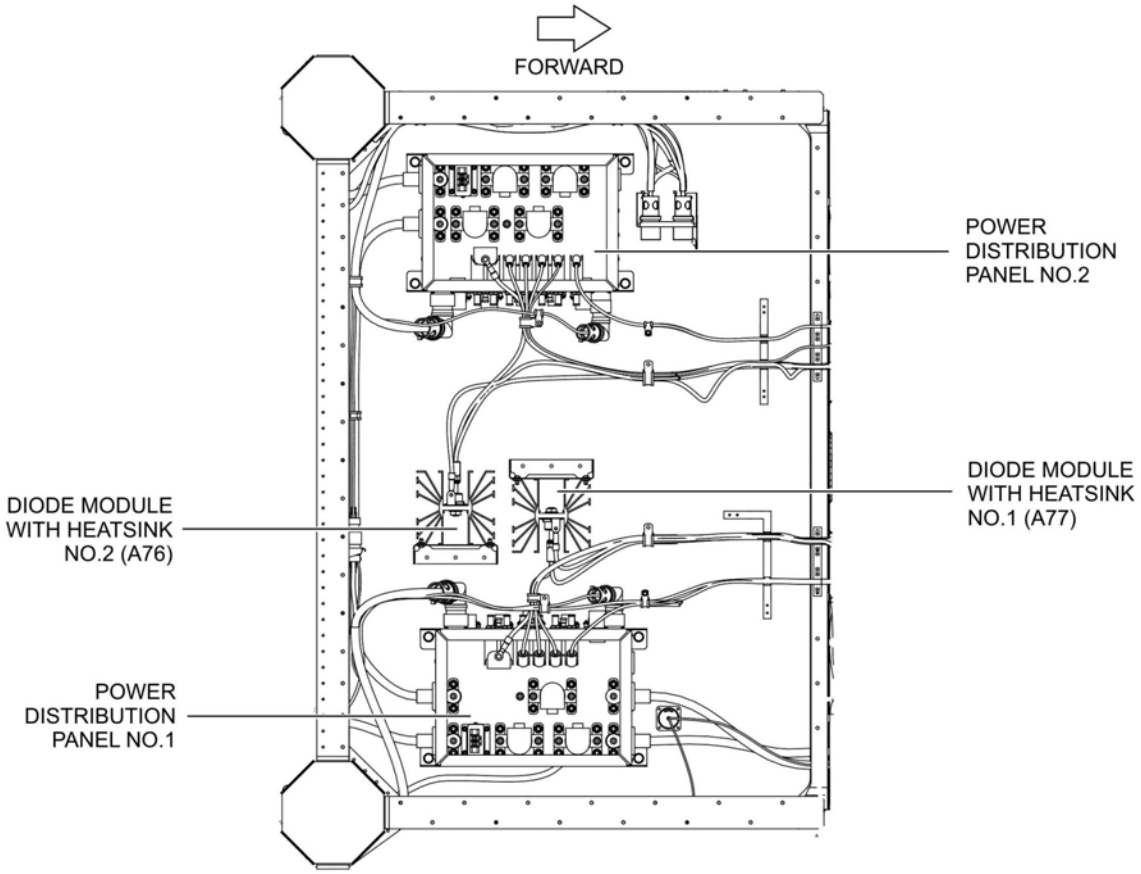
- Number 1 Power Distribution Panel (PDP1) (A3)
- Number 2 Power Distribution Panel (PDP2) (A4)
- Circuit breaker panel

NUMBER 1 POWER DISTRIBUTION PANEL

The Number 1 power-distribution panel (A3) is mounted on the roof of the cabin, on the left side.
The PDP1 is an electronic box. It has an aluminium chassis and a cover. Ten screws attach the cover to the box body. The box also has two louvers, one on the front and one on the rear side. These louvers let the heat in the box be released, thus the internal temperature of the box does not become too high.
PDP1 Contains heavy power switching components for the DC Power Subsystem on the No. 1 Generator side and including the power contactors and the majority of passive protective devices for the subsystem, part of the bus re-configuration logic for the system, the external power protection relay and bus protection control logic.

NUMBER 2 POWER DISTRIBUTION PANEL

The installation and the construction of the Number 2 power-distribution panel (A4) is almost the same as that of the PDP1. The PDP2 is mounted on the roof of the cabin, on the right side.
PDP2 contains heavy power switching components for the DC Power Subsystem on the No. 2 Generator side, including the power contactors, rectifiers, and diodes, majority of passive protective devices for the subsystem and part of the bus re-configuration logic for the system.



ICN-39-A-240000-G-A0126-00303-A-01-1

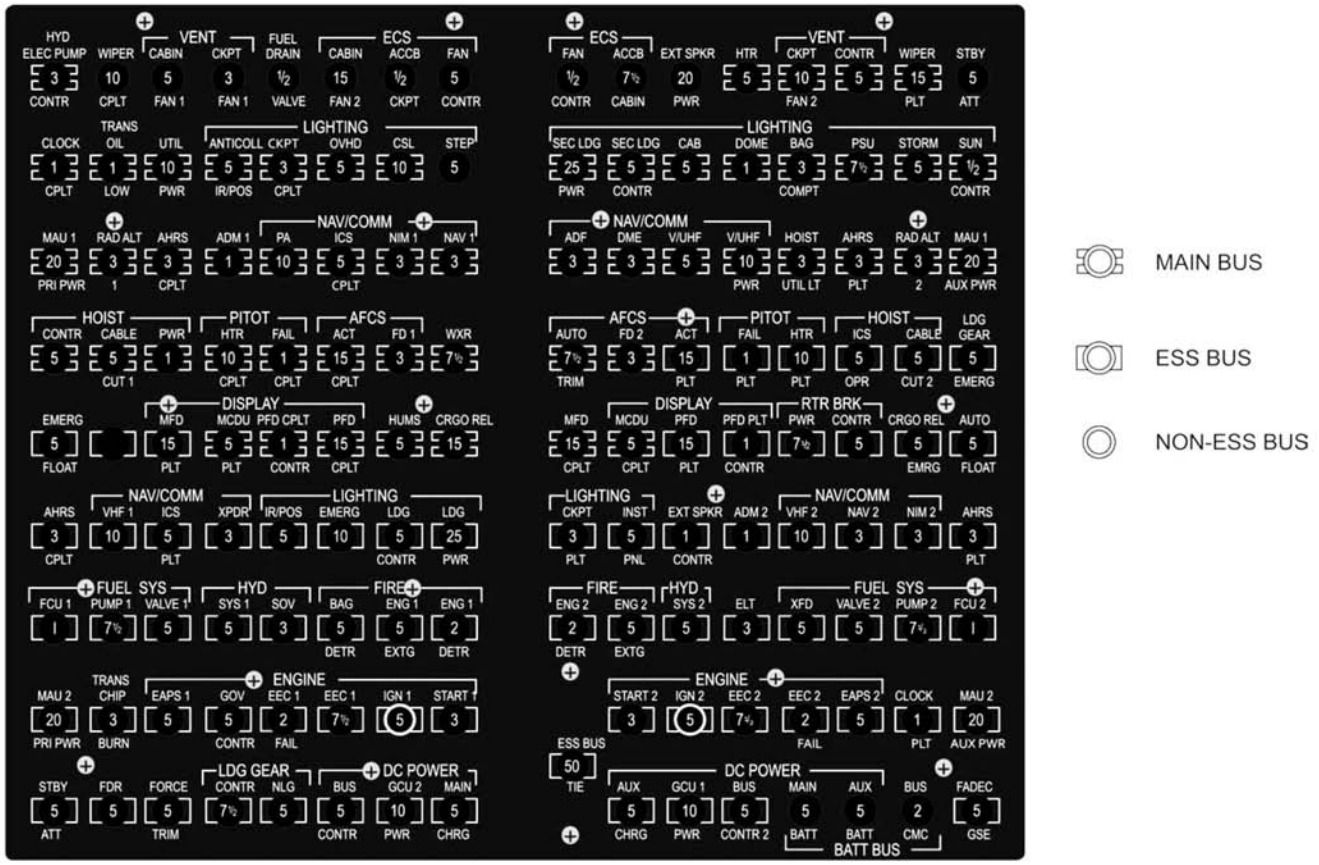
POWER DISTRIBUTION PANELS

CIRCUIT BREAKER PANEL

The circuit breaker panel is located in the overhead console and provides the connections of the rotorcraft loads to the power distribution system.
The CB panel is divided in two groups of circuit breakers:

- LH group includes the CBs connected to Essential, Main and Non-Essential bus bars number 1.
- RH group includes the CBs connected to Essential, Main and Non-Essential bus bars no.2 and Battery Bus.

The ESS BUS TIE circuit breaker (rated 50A) keeps ESS BUS 1 and ESS BUS 2 interconnected.



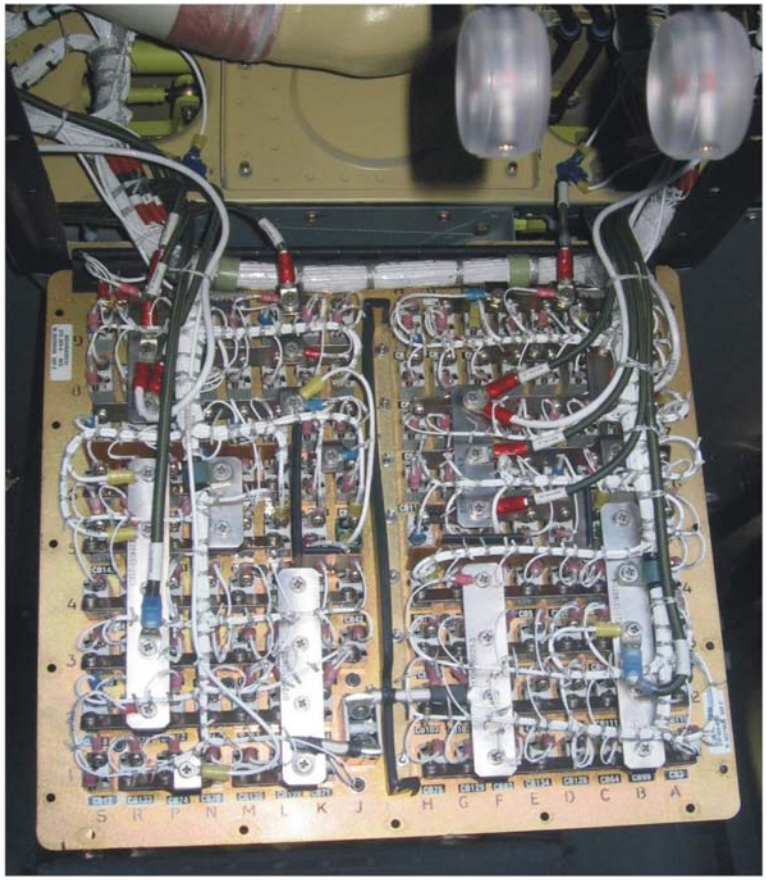
MAIN BUS

ESS BUS

NON-ESS BUS

ICN-39-A-240000-G-A0126-02912-A-01-1

CIRCUIT BREAKERS PANEL



ICN-39-A-240000-G-A0126-00305-A-01-1

CIRCUIT BREAKERS PANEL

DC CURRENT TRANSFORMER

Four current transformer are used to detect the differential changes in DC current and routes the signals to the GCU which de-energizes the line contactor when the differential feeder or the over current condition is present. The transformers are of the annular type. Their output is supplied to the power cable and gives a monitoring signal. The power cable goes through the current transformer. They are installed in the best position to let the system monitor the current.

The Number 1 DC current-transformer (T3) is installed in the Number 1 Power Distribution Panel (PDP1).

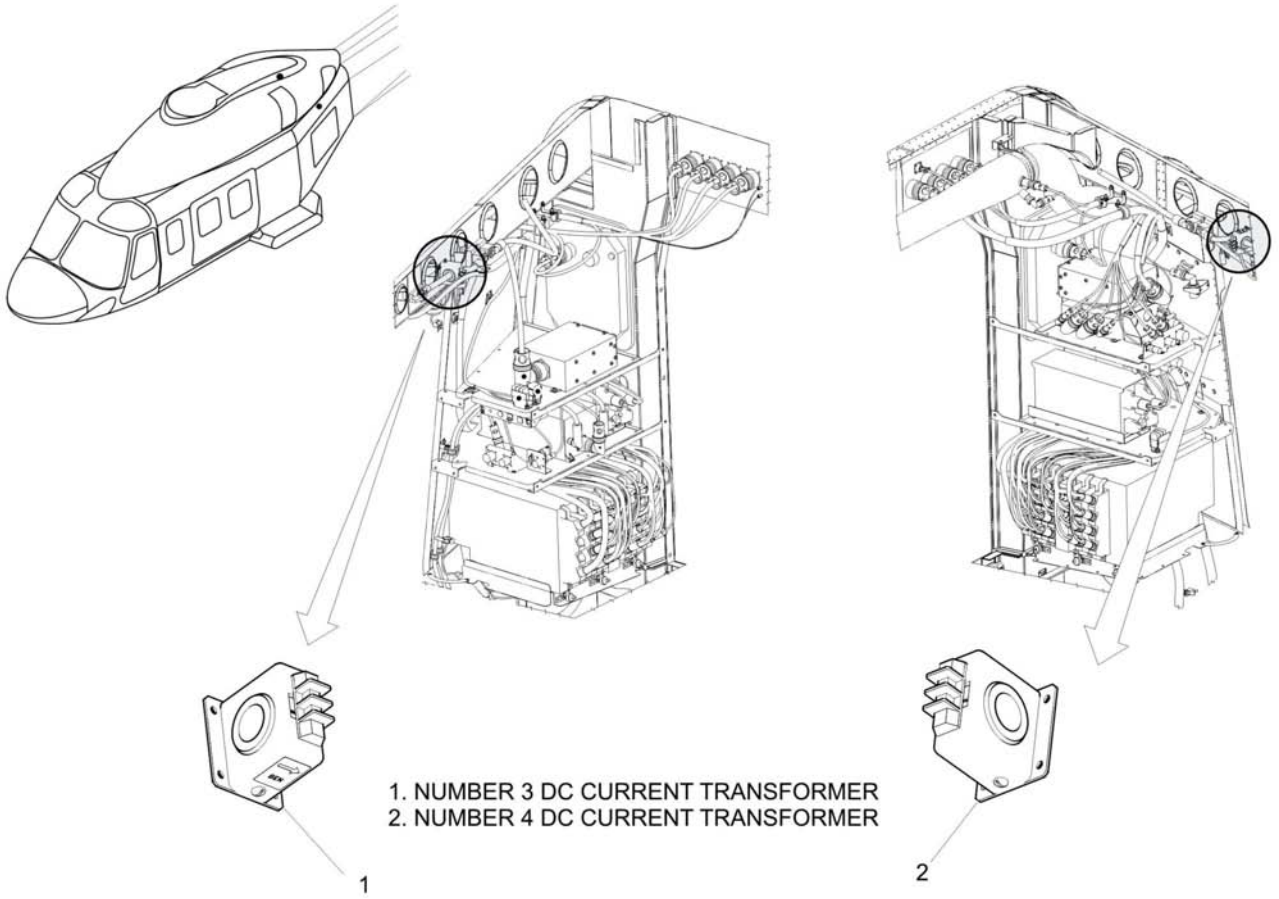
The Number 3 DC current-transformer (T1) is installed near the related shunts above the left avionic equipment bay.

Both signal are sent to the Number 1 generator-control unit that uses this signal to control the generator and the generator line contactor.

The Number 2 DC current-transformer (T4) is installed in the Number 2 Power Distribution Panel (PDP2).

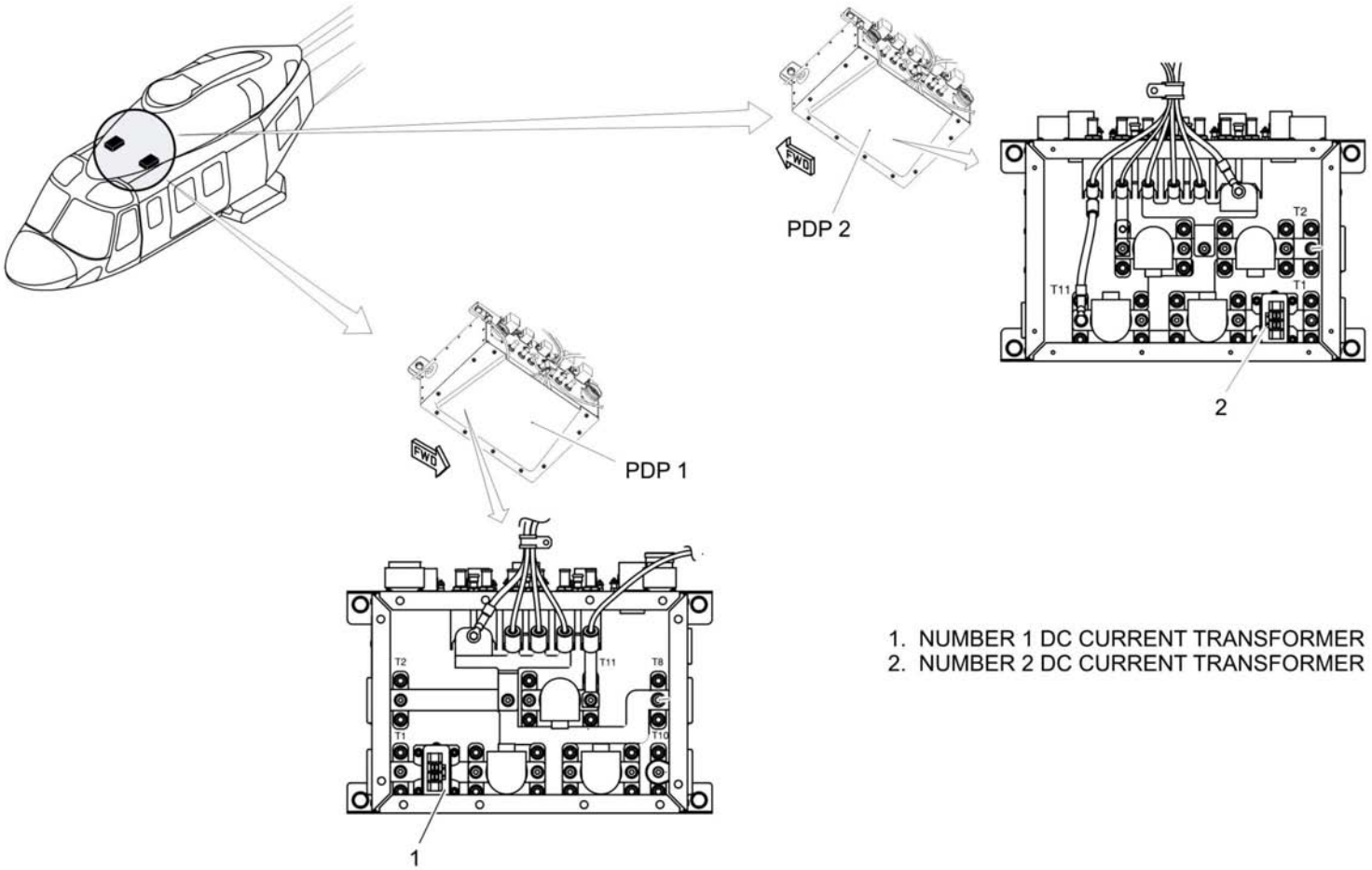
The Number 4 DC current-transformer (T2) is installed near the related shunts above the right avionic equipment bay.

Both signals are sent to the Number 2 generator-control unit that uses this signal to control the generator and the generator line contactor.



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DC CURRENT TRANSFORMER (SHEET 1 OF 2)



1. NUMBER 1 DC CURRENT TRANSFORMER
2. NUMBER 2 DC CURRENT TRANSFORMER

ICN-39-A-240000-G-A0126-00307-A-01-1

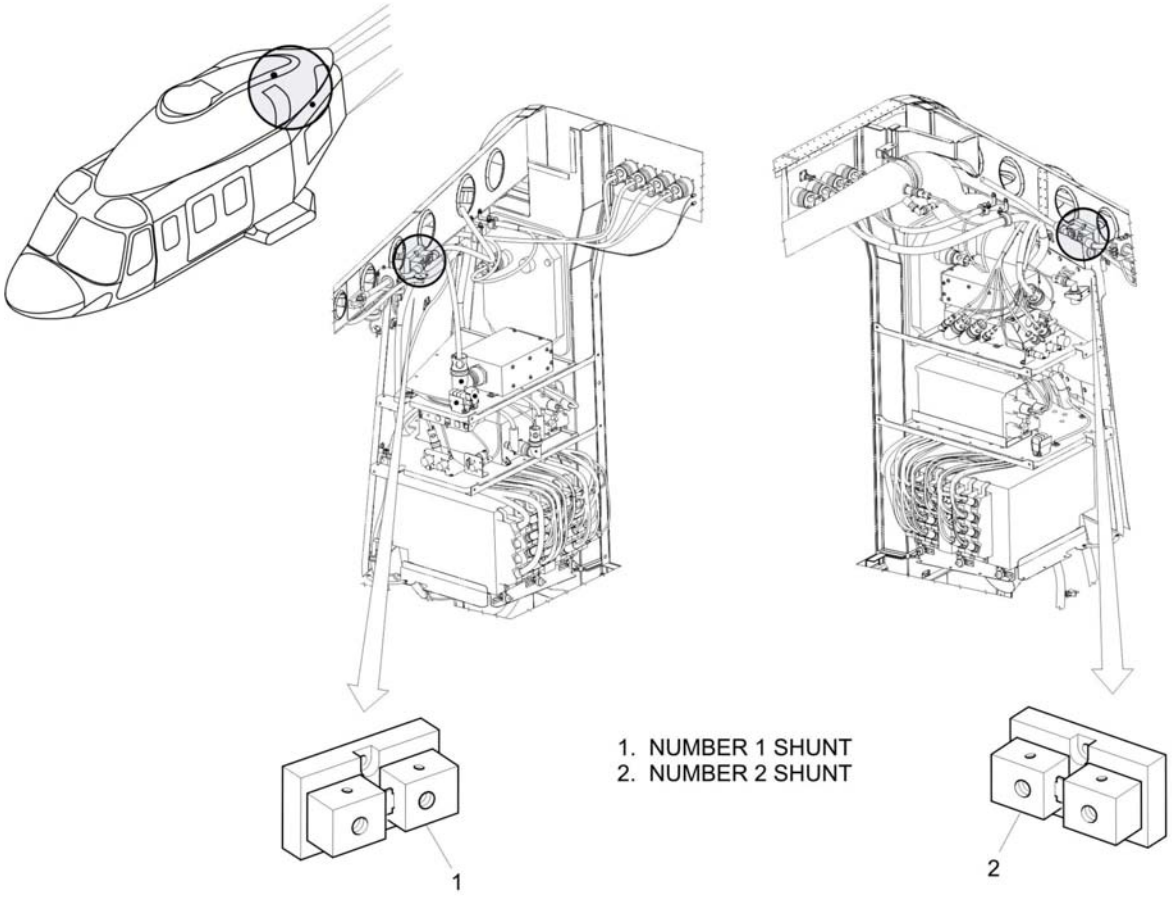
DC CURRENT TRANSFORMER (SHEET 2 OF 2)

SHUNT

Four shunts are used to pick up current from electrical power sources (starter-generator 1 and 2, main and auxiliary batteries) and display the relevant electrical load on indicators.

NUMBER 1 AND NUMBER 2 SHUNTS

The number 1 shunt (R3) measures the DC current output of the Number 1 generator and interface with the Number 1 Modular Avionic Unit (MAU1) to displays the indication of the DC ammeter on the multifunctional displays (MFDs).
The Number 2 shunt (R2) measures the DC current output of the Number 2 generator and interface with the number 2 Modular Avionic Unit (MAU2) to displays the indication of the DC ammeter on the multifunctional displays (MFDs).
They are a resistor shunt type with a 300A rating.



ICN-39-A-240000-G-A0126-00308-A-01-1

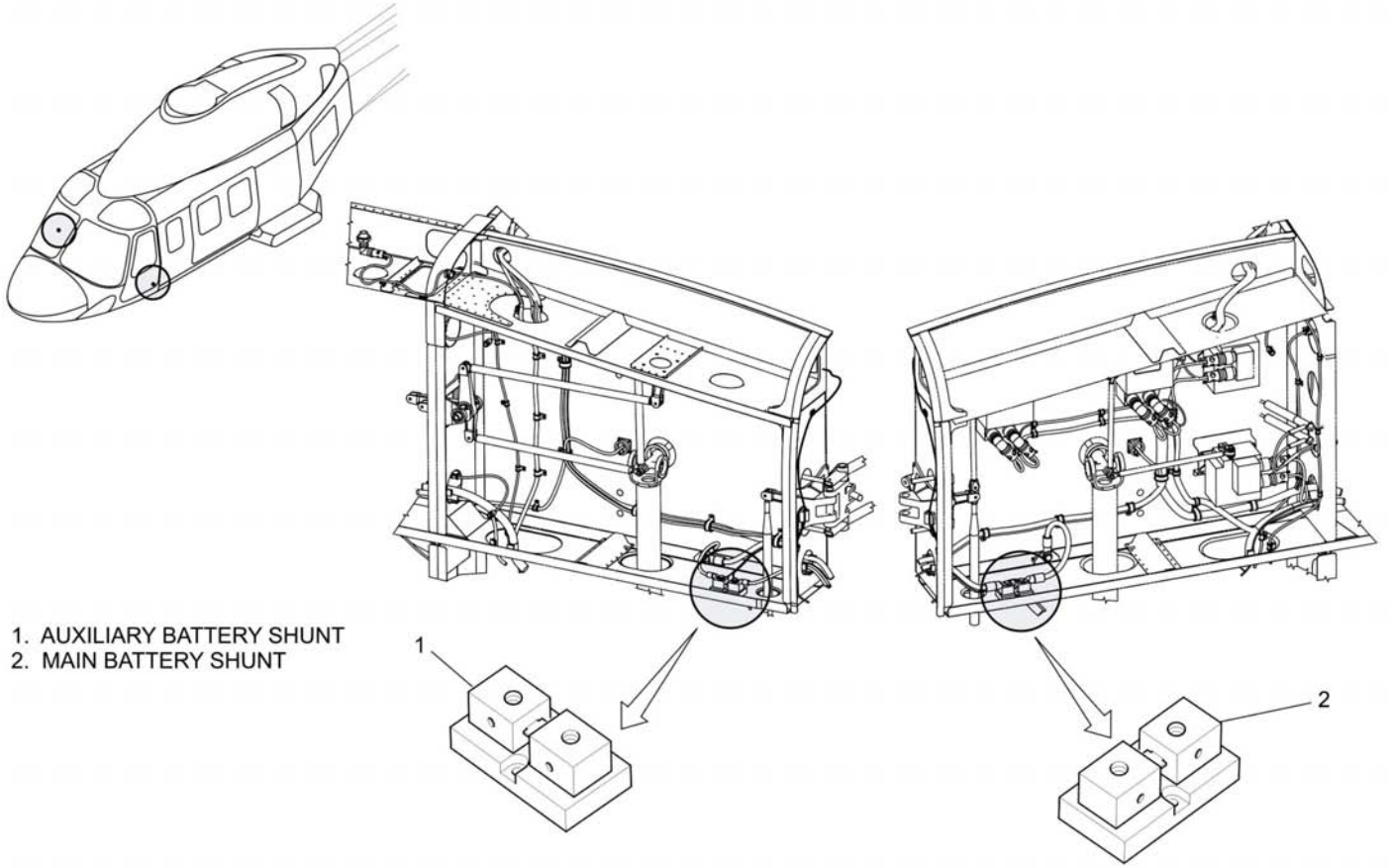
NUMBER 1 AND NUMBER 2 SHUNTS

MAIN BATTERY SHUNT

The main battery shunt (R1) is installed under cockpit floor of the helicopter on the side of the main battery DC grounding point.
It is a resistor type shunt with a 200A rating. It is connected in series between the negative terminal of the battery and the grounding point on the airframe.

AUXILIARY BATTERY SHUNT

The auxiliary battery shunt (R4) is installed under cockpit floor of the helicopter on the side of the auxiliary battery grounding point.
It is a resistor type shunt with a 200 A rating. It is connected in series between the negative terminal of the auxiliary battery and the grounding point on the airframe.

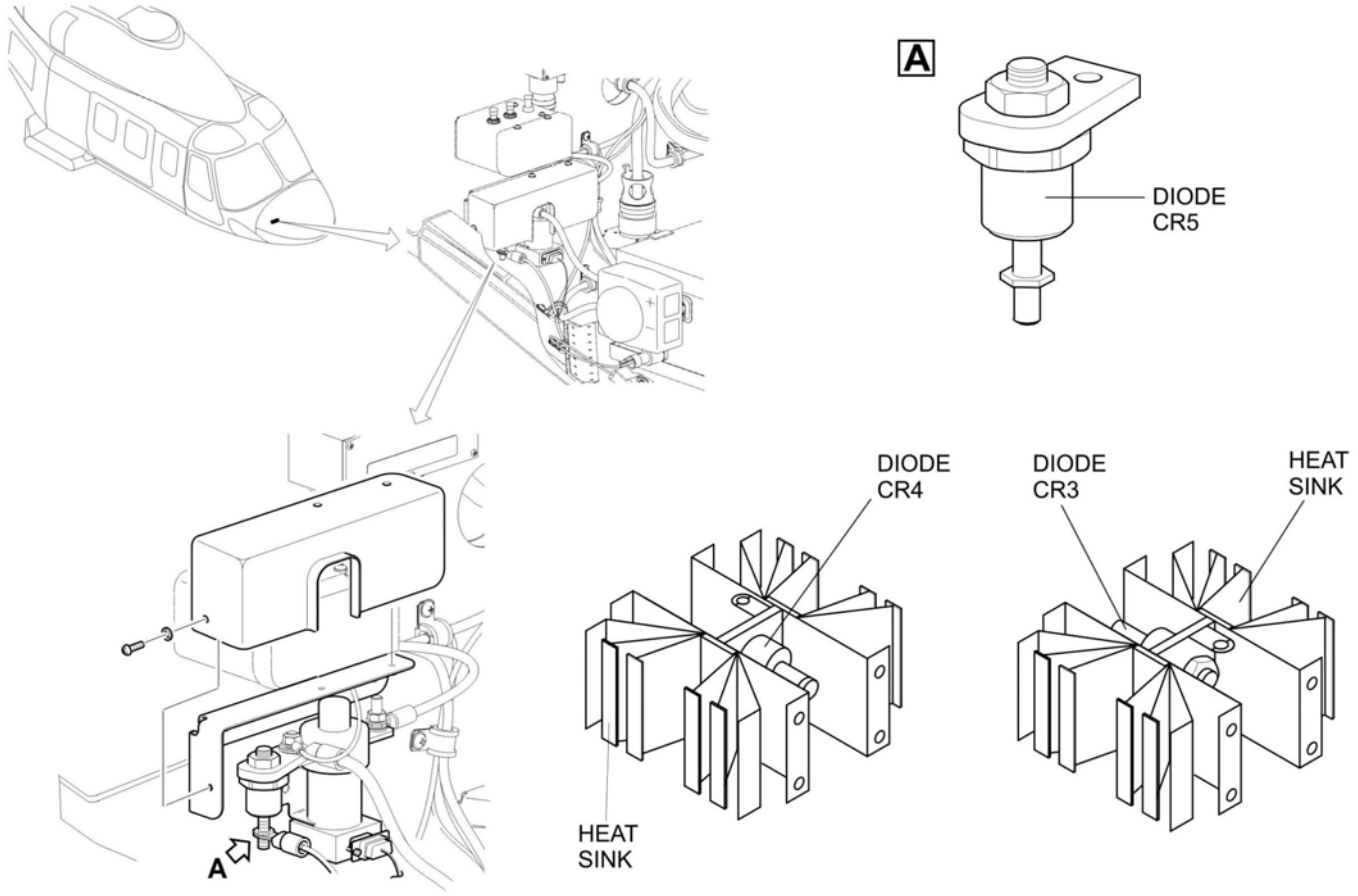


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MAIN BATTERY AND AUXILIARY BATTERY SHUNTS

DIODE

The diode is a two-terminal device; the function of a diode is to allow an electric current to flow in one direction and to block it in the opposite direction. Thus, the diode can be thought of as an electronic version of a check valve.
The diode CR3 is a part of the diode module (A77) and is attached with screws to its heat sink, connects the main battery (BT1) to the Number 1 essential bus.
The diode CR4 is a part of the diode module (A76) and it is attached with screws to its heat sink, connects the auxiliary battery to the Number 2 essential bus.
The heat sink (part of the diode module) removes the heat from the diode when the diode operates, and thus prevents over-temperature conditions.
The diode modules and the heat sinks are installed on the roof of the cabin, between the two power distribution panels (PDPs).
The diode CR5 or CR114 (long nose) are installed in the nose compartment, near the master battery relay (K2) after the AUX Battery, connects the auxiliary battery to the Number 2 main bus through the primary contact of the relay K4.
The diode has an inverse polarity, thus the diode lets the auxiliary battery be connected to the main bus only during the time the battery is charged.



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DIODE

DC GENERATION CONTROLS

The DC generation system controls are close to the circuit breaker panel located on the overhead console.
The overhead electrical switch control panel provides the following functions:

1 GEN 1(2) switch:

- OFF: ..... it disables or shuts down the generator N.1(2) and resets latched functions in the relative GCU.
- ON: ..... it enables the generator N.1(2) GCU, when at operating speed, power generation is activated.

2 BUS TIE:

- ON: ..... it always ties the MAIN BUS N.1 and N.2 in case of generator malfunction, except when an overcurrent malfunction exists. In this case the tie is inhibited and the main buses are normally isolated.
- AUTO: ..... the automatic activation of the bus tie is enabled during external power and single generator operation, except when an overcurrent malfunction exists.
- RESET: ..... it resets the logic involved in the cross-tie of the main buses and isolation of the batteries from the main buses.

NOTE B

If the batteries are connected, the bus activation is enabled with the BUS TIE switch set to “ON”.

NOTE C

If the external power is connected, the bus tie activation is normally enabled whichever position the BUS TIE switch is set.

3 BATTERY MASTER switch:

- OFF/ON: ..... allows the Essential Bus Bar operational condition through the main batteries.

4 MAIN BATTERY switch:

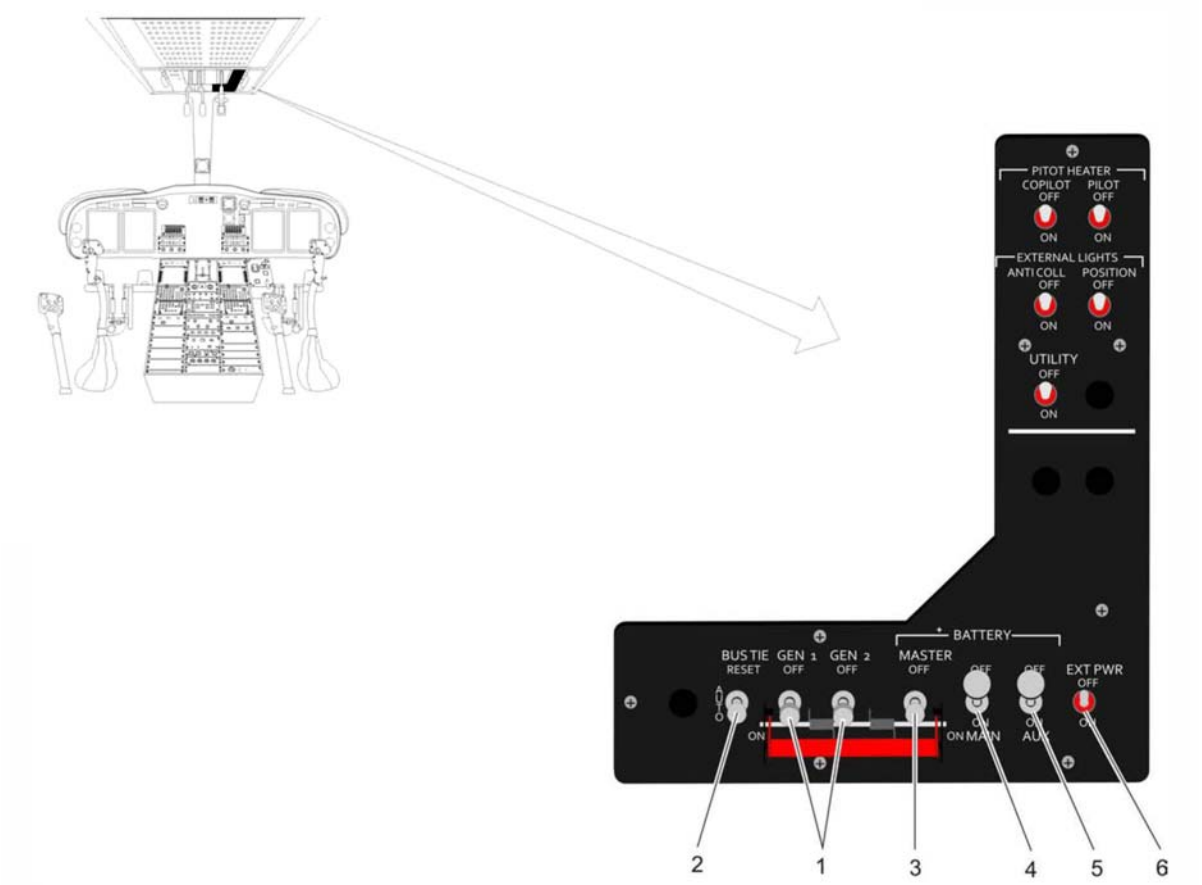
- OFF/ON: ..... allows the MAIN BUS 1 operational condition through the main battery.

5 AUX BATTERY switch:

- OFF/ON: ..... allows the AUX BATTERY recharging from the generator.

6 EXT PWR switch:

- OFF/ON: ..... allows the external power connection to the aircraft bus bars.



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DC GENERATOR CONTROL

DC GENERATION INDICATING

The Multifunction Displays (MFD's) provide the:

- DC electrical operative parameter.
- DC electrical synoptic page.
- Caption messages in the CAS (Crew Alerting System) window.

DC ELECTRICAL OPERATIVE PARAMETERS

Voltmeters

The volt meters are the digital readouts of the voltage on MAIN BUS 1, MAIN BUS 2, ESS BUS 1 and ESS BUS 2.
When voltage is below 22.0 V the values are displayed in amber reverse video; if it occurs during flight following a dual generator failure it means that the battery is about to discharge completely and a total black-out of electrical power may be expected soon.
When voltage is below 12.0 V values are set to 0.0.

NOTE D

ESS BUS 1 and 2 voltage is normally less than MAIN BUS 1 and 2 voltage by about 0.8 VDC because of the drop of voltage across the diodes that protect the ESS Busses.

Battery Ampere Meters

Green band of the analogue vertical scale of the MAIN and AUX battery ampere meters represents a battery charge condition and is associated to positive digital readout values in Ampere.

Amber band represents a battery discharge condition and is associated to amber reverse video negative digital readout values in Ampere.

Generator Load Meters

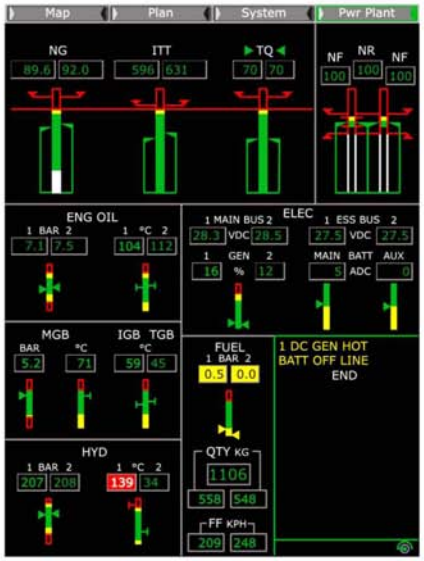
The generator load is displayed in % of maximum continuous generator output power (100% ≡ 300 A).
Overload in the amber cautionary range is only admitted for short time periods since a generator over temperature may be expected.
Overload in the red warning range must be avoided.

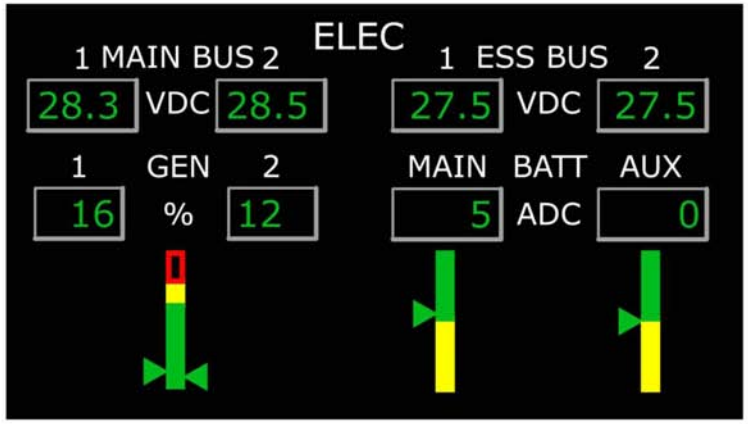
NOTE E

When starting the second engine on batteries (Assisted Start) the generator that is already on-line is allowed to operate in the red range overload condition.

NOTE F

In case of the invalid parameters digital readouts are replaced by yellow dashed and pointer are not displayed.





Maximum load for each generator vs. Pressure altitude

ft Hp	below 15000	16000	17000	18000	19000	20000
GEN LOAD %	100	87	74	60	47	33

NOTE: This placard is located on the instrument panel

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DC GENERATOR INDICATING

REF	FUNCTION	RANGE	DIGITAL READOUT COLOR MARKINGS	ANALOG SCALE COLOR MARK-INGS	NOTE
1	MAIN BUS 1 VOLTMETER digital readout	10,0 to 40,0 VDC	Green ≥ 22 V Amber < 22 V		
2	MAIN BUS 2 VOLTMETER digital readout	10,0 to 40,0 VDC	Green ≥ 22 V Amber < 22 V		
3	ESS BUS 1 VOLTMETER digital readout	10,0 to 40,0 VDC	Green ≥ 22 V Amber < 22 V		
4	ESS BUS 2 VOLTMETER digital readout	10,0 to 40,0 VDC	Green ≥ 22 V Amber < 22 V		
5	AUX BATTERY LOADMETER digital readout	-200 to 300 ADC	Green when 1 to 200 White at 0 Amber when -200 to -1		
6	MAIN BATTERY LOADMETER digital readout	-999 to 300 ADC	Green when 1 to 200 White at 0 Amber when -200 to -1		
7	AUX BATTERY LOADMETER digital readout	-200 to 200 ADC		Green band 0 to +200 Amber band -200 to -1	+ battery charge - battery discharge
8	MAIN BATTERY LOADMETER analog scale	-200 to 200 ADC		Green band 0 to +200 Amber band -200 to -1	+ battery charge - battery discharge
9	DC GENERATOR LOADMETER analog scale	0 to 200%		Red band 151 to 200% Red line 151% Amber band 101 to 150% Green band 0 to 100%	

REF	FUNCTION	RANGE	DIGITAL READOUT COLOR MARKINGS	ANALOG SCALE COLOR MARKINGS	NOTE
10	DC GENERATOR 2 LOADMETER digital readout	0 to 200%	Red when 151 to 200% Amber when 101 to 150% Green when 0 to 100%		
11	DC GENERATOR 1 LOADMETER digital readout	0 to 200%	Red when 151 to 200% Amber when 101 to 150% Green when 0 to 100%		

DC ELECTRICAL SYNOPTIC PAGE

On each Multi-Function Display (MFD) a synoptic page representing a diagram with status of the DC power system operation, it can be displayed via the “System/DC Electrical” menu.
The DC Electrical synoptic page shows the status of:

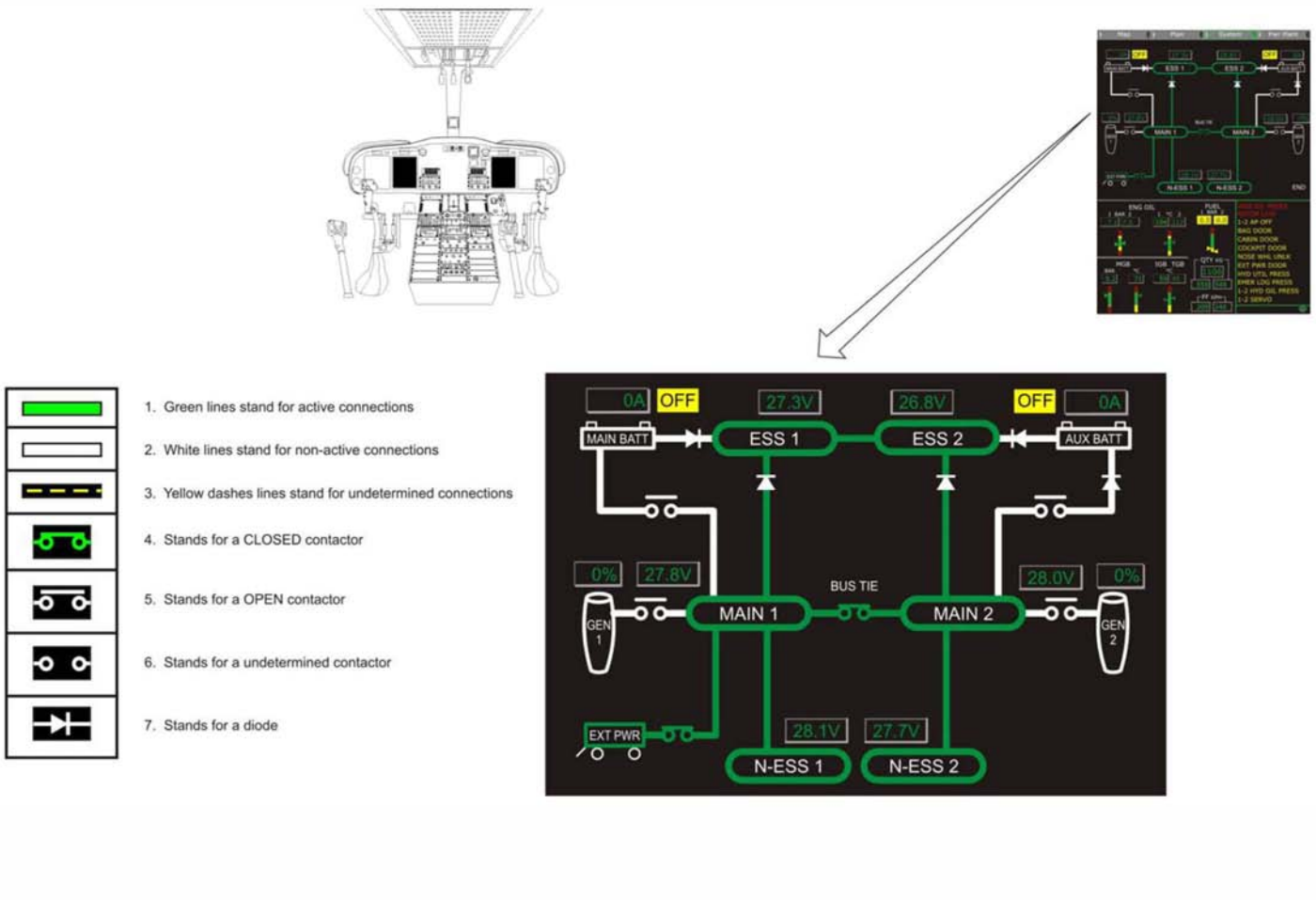
- MAIN Battery and AUX Battery connection status, current flow (amperemeter).
- GEN 1 and GEN 2 status, connection to MAIN buses and electrical load (loadmeter in % of max continuous power).
- ESSENTIAL, MAIN and NON-ESSENTIAL buses status and voltage (voltmeters).
- BUS TIE contactor status
- EXTERNAL POWER status

DESCRIPTION OF SYMBOLS

1 Green lines stand for active connections.
2 White lines stand for non-active connections.
3 Yellow dashes lines stand for undetermined connections.
4 Stand for a CLOSED contactor.
5 Stand for an OPEN contactor.
6 Stand for an undetermined contactor.
7 Stand for a diode.

NOTE G

Battery amperemeter show yellow reverse-video negative values when battery is discharging.
Voltmeter show yellow reverse-video values when voltage is less than 22 VDC.



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DC POWER SYNOPTIC PAGE-EXTERNAL POWER ON

CAPTIONS

The captions messages are provided in the CAS (Crew Alerting System) window in the MFD's are as follow:

- WARNING
- CAUTION
- ADVISORY

WARNING MESSAGES

The warning messages provide in the CAS message windows are as follow:

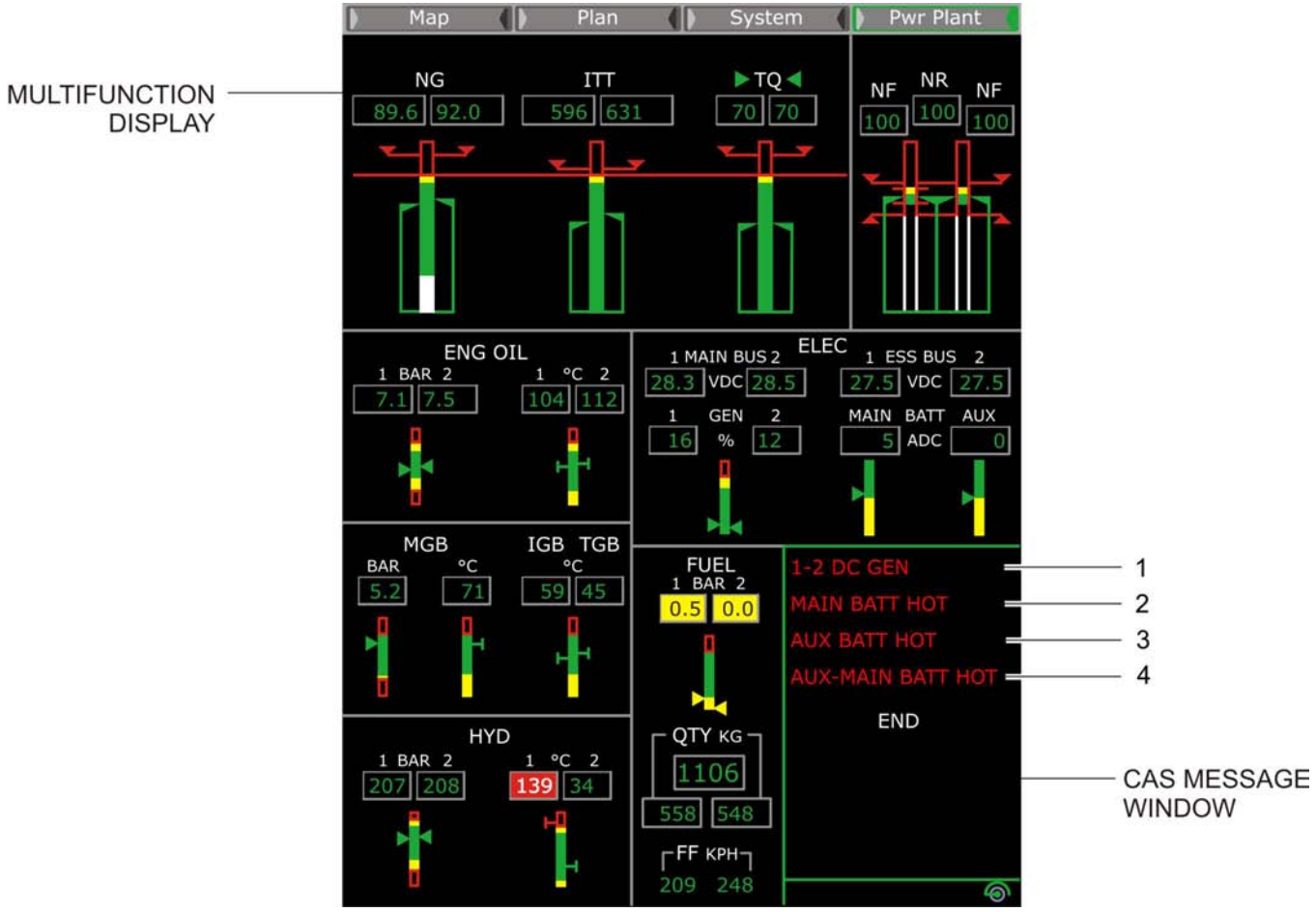
1	1-2 DC GEN:	when DC GEN N.1 off-line or shut down by the GCUs due to malfunction.
2	MAIN BATT HOT:	when main battery temperature exceeds 65°C.
3	AUX BATT HOT:	when auxiliary battery temperature exceeds 65°C.
4	AUX – MAIN BATT HOT: …..	when auxiliary and main batteries temperature exceed 65°C.

NOTE H

When one of the above warning is detected, besides the MWL activation, the AWG shall provide the following:

- No tone + WARNING – WARNING (aural message) once only.

This message has priority number 5.



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CAPTIONS – WARNING MESSAGES

WARNING MESSAGES			
CAS CAPTION	**FAILURE DESCRIPTION**	**PROCEDURE NAME**	**AW139-RFM-4D**
1-2 DC GEN	Failure of both generators in flight • Voice warning: WARNING – WARNING repeated once	DOUBLE GENERATOR FAILURE EXTENDED FLIGHT ENDURANCE AFTER DOUBLE DC GENERATOR FAILURE	Section 3 EMERGENCY AND MALFUNCTION PROCEDURES ELECTRICAL SYSTEM
MAIN BATT HOT	Main battery overheating • Voice warning: WARNING – WARNING repeated once	MAIN AND AUXILIARY BATTERY HOT	
AUX BATT HOT	Auxiliary battery overheating • Voice warning: WARNING – WARNING repeated once		
AUX MAIN BATT HOT	Main and Auxiliary battery overheating • Voice warning: WARNING – WARNING repeated once		

CAUTION MESSAGES

The caution messages provide in the CAS message window are as follow:

1 1(2) DC GEN: ..... when DC GEN N.1(2) off-line or shut down by the relative GCU due to malfunction.

2 1(2) DC GEN HOT: ..... when DC GEN N.1(2) temperature exceeds 166°C ± 13°C (330°F ± 8°F).

3 BUS TIE OPEN ..... when BUS TIE line is open in presence of one generator failure and with external power on.

4 DC BUS FAIL: ..... when a bus fault has been detected on MAIN buses due to an overcurrent.

5 MAIN BATT OFF: ..... when the main battery off-line or battery is not being charged.

6 AUX BATT OFF: ..... when the auxiliary battery off-line or battery is not being charged.

7 BATT OFF LINE: ..... when one or both batteries are disconnected from the Essential Bus Bar.

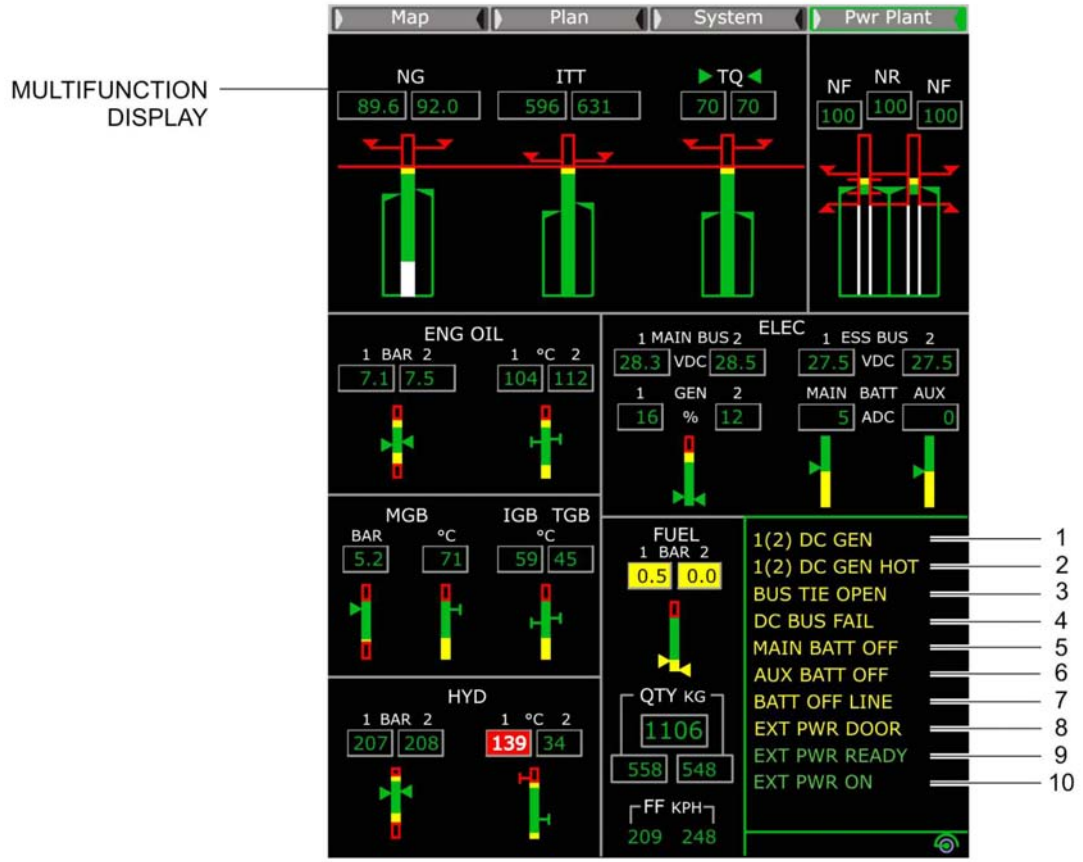
8 EXT PWR DOOR: ..... when the external power door is open.

ADVISORY MESSAGES

The advisory messages provide in the CAS message window are as follow:

9 EXT PWR READY: ..... when the external power is available and voltage within the range for application to aircraft bus bars.

10 EXT PWR ON: ..... when the external power is connected to aircraft bus bars.



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CAPTIONS – CAUTION AND ADVISORY MESSAGES

CAUTION MESSAGES			
CAS CAPTION	**FAILURE DESCRIPTION**	**PROCEDURE NAME**	**AW139-RFM-4D**
1(2) DC GEN	Associated generator failure	SINGLE GENERATOR FAILURE	Section 3 EMERGENCY AND MALFUNCTION PROCEDURES ELECTRICAL SYSTEM
1(2) DC GEN HOT	Associated generator overheat	DC GENERATOR OVERHEAT	
MAIN BATT OFF	Failure of MAIN battery to MAIN BUS 1	MAIN BATTERY OFF	
AUX BATT OFF	Failure of AUX battery to MAIN BUS 2	AUXILIARY BATTERY OFF	
BUS TIE OPEN	BUS TIE open with one or both generators off line	BUS TIE OPEN	
BATT OFF LINE	Failure of MAIN and/or AUX battery connection to ESS BUS	LOSS OF MAIN AND/OR AUXILIARY BATTERY SUPPLY	
DC BUS FAIL	DC MAIN BUS 1 and/or 2 fault detected (overcurrent detected by either GCU)	DC MAIN BUS FAILURE	

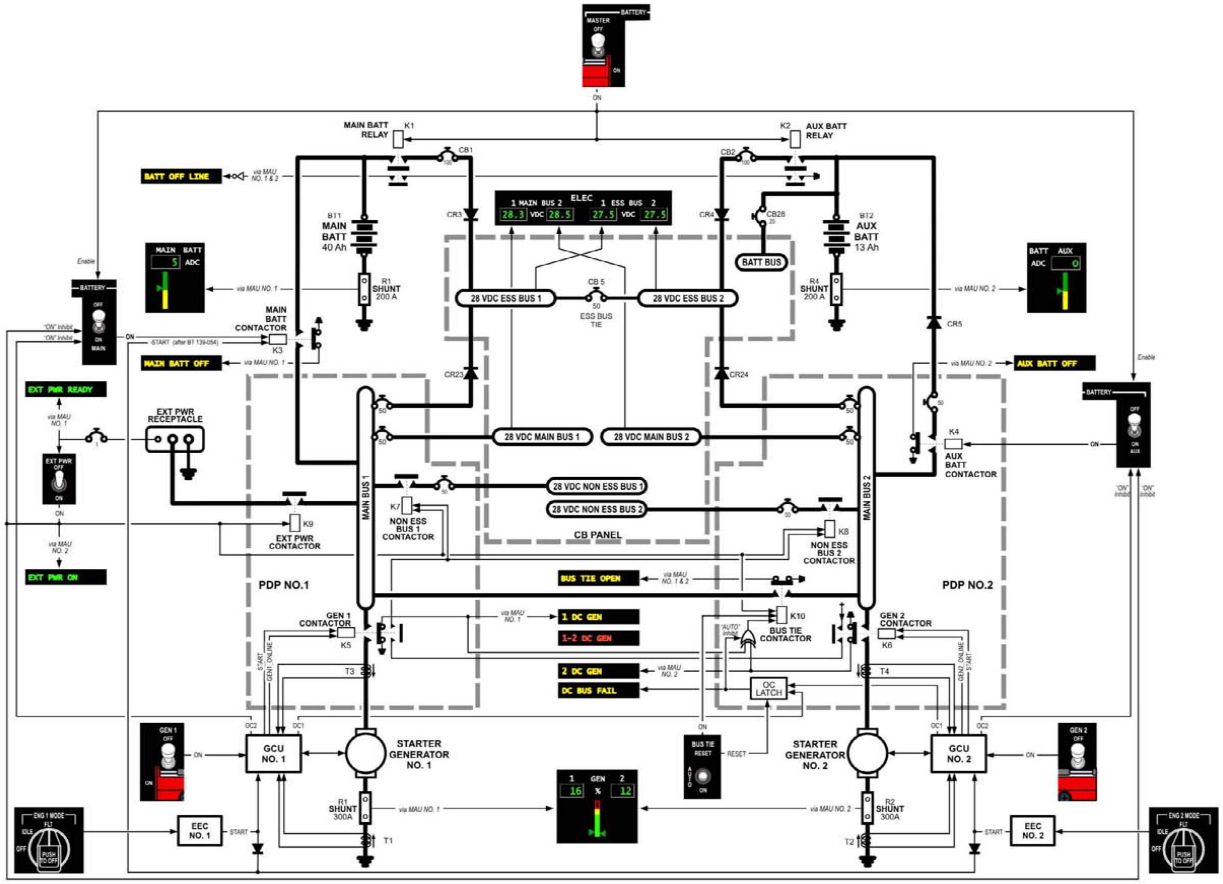
CAUTION MESSAGES			
CAS CAPTION	**FAILURE DESCRIPTION**	**PROCEDURE NAME**	**AW139-RFM-4D**
EXT PWR DOOR	External power socket door not closed	EXTERNAL POWER SOCKET DOOR OPEN	Section 3 EMERGENCY AND MAL-FUNCTION PROCEDURES MISCELLANEOUS

ADVISORY MESSAGES			
CAS CAPTION	**FAILURE DESCRIPTION**	**PROCEDURE NAME**	**AW139-RFM-4D**
EXT PWR READY	External power voltage available at receptacle but external power not connected to MAIN BUS		Section 2 NORMAL PROCEDURES ADVISORY CAPTION DEFINITIONS
EXT PWR ON	External power connected to MAIN BUS		

ELECTRICAL POWER – SIMPLIFIED SCHEMATIC DIAGRAM

A simplified schematic diagram is used to describe the operation of the electrical power system in the following normal and emergency situations:

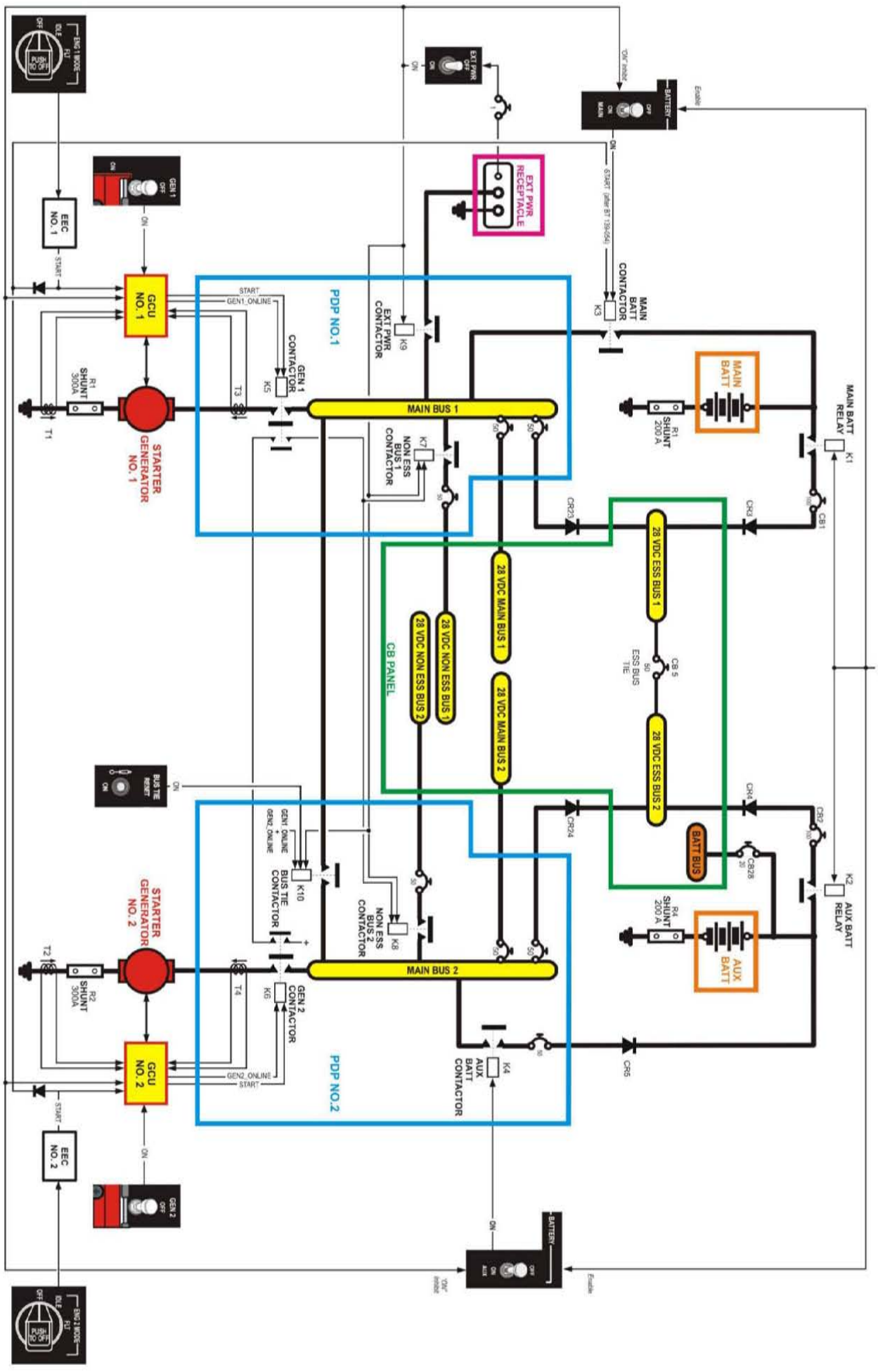
- Normal Procedures
  - Engine starting on batteries (BATTERY Starting)
  - Engine starting on external power (EXT PWR Starting)
  - In-flight normal operation
- Emergency Procedures
  - Single Generator Failure
  - Single DC Bus Failure
  - Dual Generator Failure



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ELECTRICAL POWER-SIMPLIFIED SCHEMATIC DIAGRAM

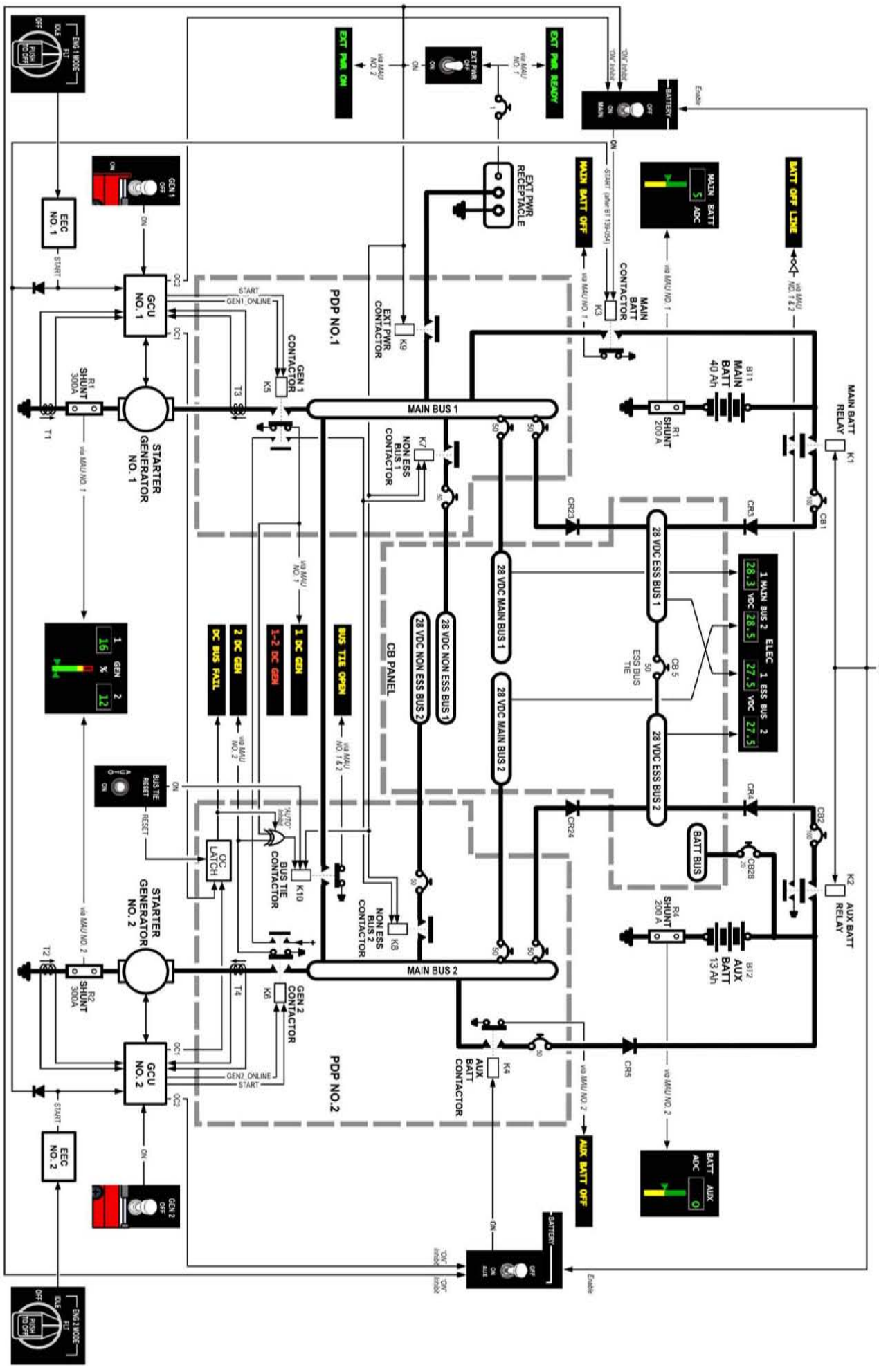
(Printed A3 format at the end of data module)



DC ELECTRICAL POWER SYSTEM-BLOCK DIAGRAM

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ELECTRICAL POWER-SIMPLIFIED SCHEMATIC DIAGRAM

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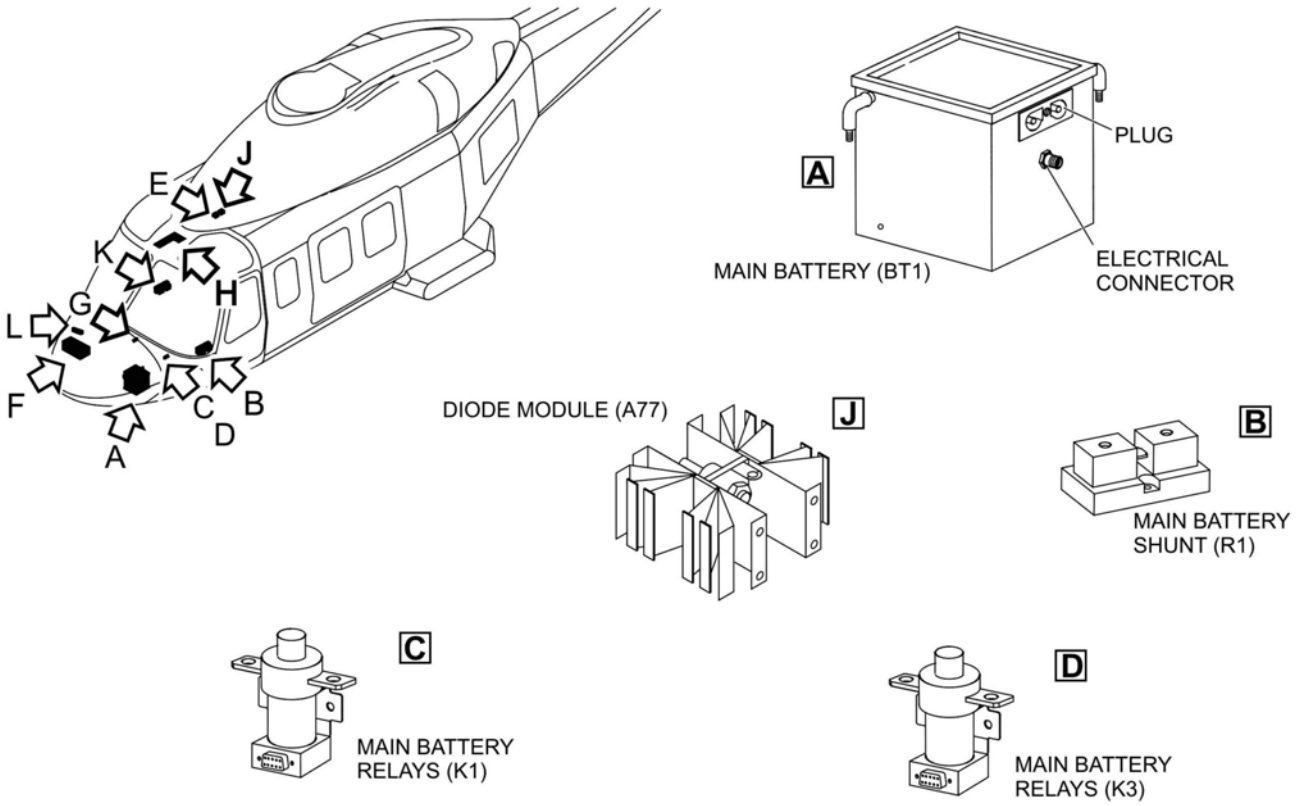
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SECTION 30 - DC MAIN GENERATION

BATTERY POWER GENERATION SYSTEM DESCRIPTION

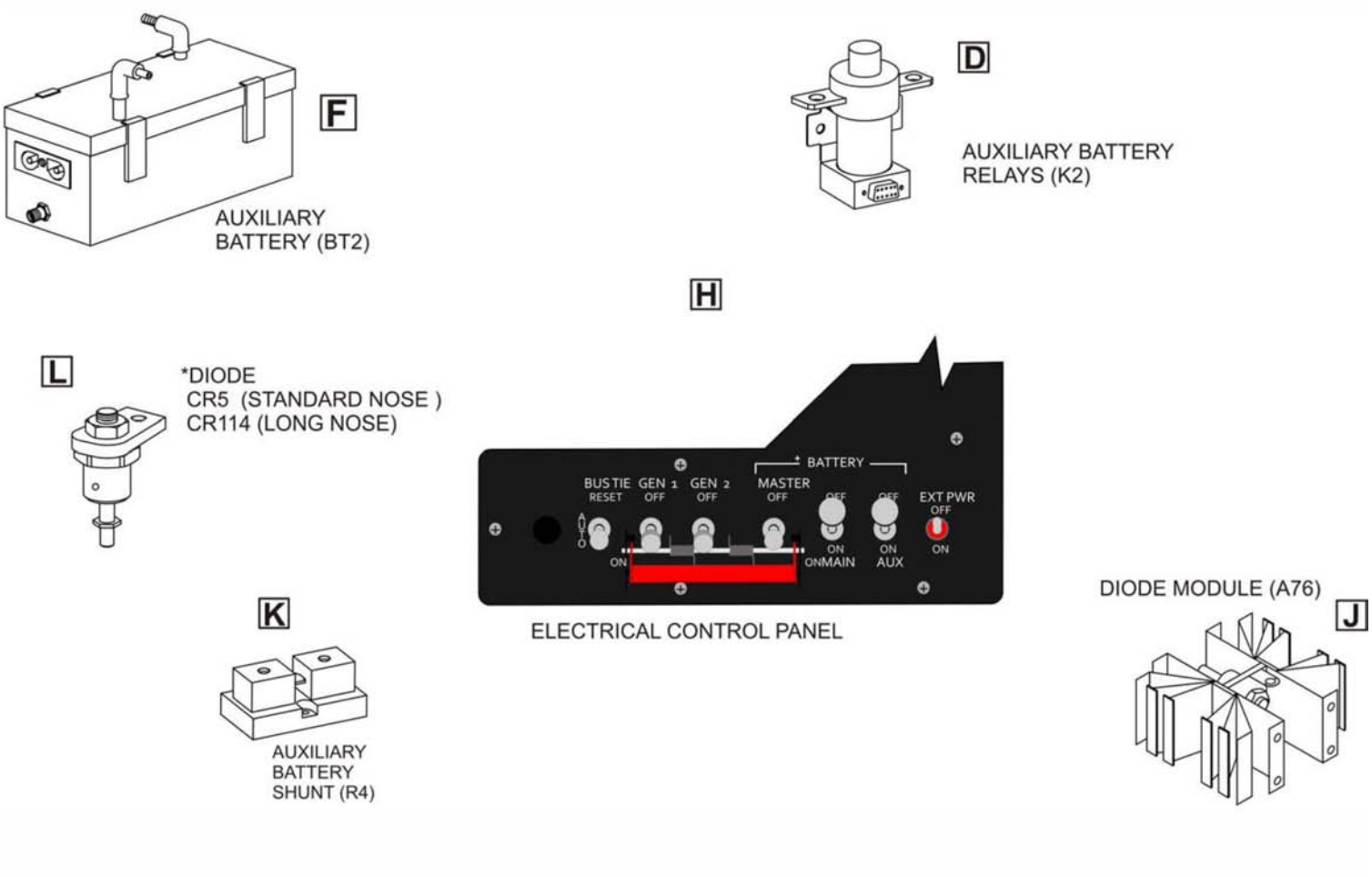
The battery power generation system includes these primary components:

- The main battery (BT1)
- The main battery shunt (R1)
- The main battery relays (K1) and (K3)
- The Diode (CR3)
- The master battery control-switch (S6)
- The main battery control-switch (S10)
- The auxiliary battery (BT2)
- The auxiliary battery shunt (R4)
- The master battery relay (K2) and auxiliary battery relay (K4)
- The auxiliary battery control-switch (S9)
- The Diodes (CR2) and (CR4).



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BATTERY POWER GENERATOR SYSTEM-PRIMARY COMPONENTS (SHEET 1 OF 2)



ICN-39-A-243000-G-A0126-02915-A-01-1

BATTERY POWER GENERATOR SYSTEM-PRIMARY COMPONENTS (SHEET 2 OF 2)

MAIN BATTERY

The main battery (BT1) is installed on the left side in the nose compartment.
It is an assembly made of many cells connected in series. The primary component of each cell is a nickel-cadmium plate put in an electrolyte (potassium hydroxide).
Solid copper bars connect the cells. Bolts attach the bars to the positive and negative terminals of each cell.
All the cells are contained in a steel case with a cover that you can remove. The case cover has tabs that are used to attach the battery to the helicopter structure.
The battery case has openings to which there are connected the tubes that release the battery gases overboard.
A plug installed on the case is the primary electrical connection of the battery.
The battery also includes an internal temperature switch that monitors the temperature in the battery. The switch is of the two-metal type; a special connector connects it to the case.
The main battery has a nominal power output of 24 V (40 Ah).

MAIN BATTERY SHUNT

The main battery shunt (R1) is installed under the cockpit floor of the helicopter on the side of the main battery DC grounding point.
It is a resistor type shunt with a 200 A rating. It is connected in series between the negative terminal of the battery and the grounding point on the airframe.

MAIN BATTERY RELAYS

The main battery relays (K1) and (K3) are installed in the nose compartment, aft of the main battery.
The rating of these relays is 300 A.

DIODE CR3

The diode CR3 is a part of the diode module (A77) and is attached with screws to its heat sink. The diode and the heat sink are installed on the roof of the cabin, between the two power distribution panel.

MASTER BATTERY CONTROL SWITCH

The master battery control-switch (S6) (BATTERY MASTER) is installed on the overhead console in the cockpit.
It is a two position switch with the two selection that follow:

- OFF (up)
- ON (down)

MAIN BATTERY CONTROL SWITCH

The main battery switch (S10) (BATTERY MAIN) is installed on the overhead console in the cockpit.
It is a two position switch with the two selection that follow:

- OFF (up)
- ON (down)

AUXILIARY BATTERY

The auxiliary battery (BT2) is installed on the right side of the nose compartment.
It is an assembly made of many cells connected in series.
The auxiliary battery construction is the same as that of the main battery. But the auxiliary battery is smaller than the main battery.
A sensor that senses the internal temperature is installed in the auxiliary battery.

The auxiliary battery has a nominal power output of 24 V (13 Ah).

AUXILIARY BATTERY SHUNT

The auxiliary battery shunt (R4) is installed under the cockpit floor of the helicopter on the side of the auxiliary battery grounding point.
It is a resistor type shunt with a 200 A rating. It is connected in series between the negative terminal of the auxiliary battery and the grounding point on the airframe.

MASTER BATTERY RELAY AND AUXILIARY BATTERY RELAY

The master battery relays (K2) is installed in the nose compartment, aft of the main battery.
The auxiliary battery relay (K4) is installed in the Number 2 power-distribution panel.
The rating of these relays is 300 A.

AUXILIARY BATTERY CONTROL SWITCH

The auxiliary battery control-switch (S9) (BATTERY AUX) is installed on the overhead console in the cockpit.
It is a two position switch with the two selection that follow:

- OFF (up)
- ON (down)

DIODES CR4 AND CR5

The diode CR4 is a part of the diode module (A76) and it is attached with screws to its heat sink.
The diode and the heat sink are installed on the roof of the cabin, between the two power distribution panels.
The diode CR5 is installed in the nose compartment, near the master battery relay (K2).

OPERATION

MAIN BATTERY

When the master battery control-switch (S6) is set to ON, the main battery (BT1) supplies power to the Number 1 and Number 2 essential busses.
If the generator power is fully lost, the battery can supply power to the essential busses.
If the battery internal temperature increases above a dangerous value, a temperature switch supplies a signal to cause the MAIN BATT HOT warning to come on in the MFDs.

MAIN BATTERY SHUNT

The main battery shunt (R1) senses the current that flows through it.
The shunt transmits an electrical signal to the MAUs. The signal is used to show the current data on the MFDs.

MAIN BATTERY RELAYS

The main battery relays (K1) and (K3) connect the output voltage of the main battery to the Number 1 essential bus (ESS BUS1) and the Number 1 main bus (MAIN BUS1).

DIODE CR3

The diode CR3 (part of the diode module A77) connects the main battery (BT1) to the Number 1 essential bus.

The heat sink (part of the diode module) removes the heat from the diode when the diode operates, and thus prevents over-temperature conditions.

MASTER BATTERY CONTROL SWITCH

When the master battery control-switch (S6) (BATTERY MASTER) is set to OFF, the main battery is disconnected from the two essential busses.
When set to ON, it connects the main battery to the Number 1 and Number 2 essential busses (ESS BUS1 and ESS BUS2).

MAIN BATTERY CONTROL SWITCH

When the main battery control-switch (S10) (MAIN BATT) is set to OFF, the main battery is disconnected from the Number 1 main bus.
When set to ON, it connects the main battery to the Number 1 main bus (MAIN BUS1).

AUXILIARY BATTERY

The auxiliary battery (BT2) supplies power directly to the BATTERY BUS. When the BATTERY MASTER switch is set to ON, it supplies power to the essential buses. It is recharged through the Number 2 main bus (MAIN BUS2) when the BATTERY MASTER and AUX BATT switches are set to ON and a generator is on line.
If the battery internal temperature increases above a dangerous value, a temperature switch supplies a signal to cause the AUX BATT HOT warning to come on in the MFDs.

AUXILIARY BATTERY SHUNT

The auxiliary battery shunt (R4) senses the current that flows through it.

The shunt transmits an electrical signal to the MAUs. The signal is used to show the current data on the MFDs.

MASTER BATTERY RELAY AND AUXILIARY BATTERY RELAY

The master battery relay K2 connects the auxiliary battery to the Number 2 essential bus, thus it can supply the essential buses with power.
The auxiliary battery relay K4 connects the auxiliary battery to the Number 2 main bus, only for charging.

AUXILIARY BATTERY CONTROL SWITCH

With the auxiliary battery control-switch (S9) (BATTERY AUX) set to OFF, the auxiliary battery is disconnected from the Number 2 main bus.
When the switch is set to ON, it connects the auxiliary battery to the Number 2 main bus (MAIN BUS2).

DIODES CR4 AND CR5

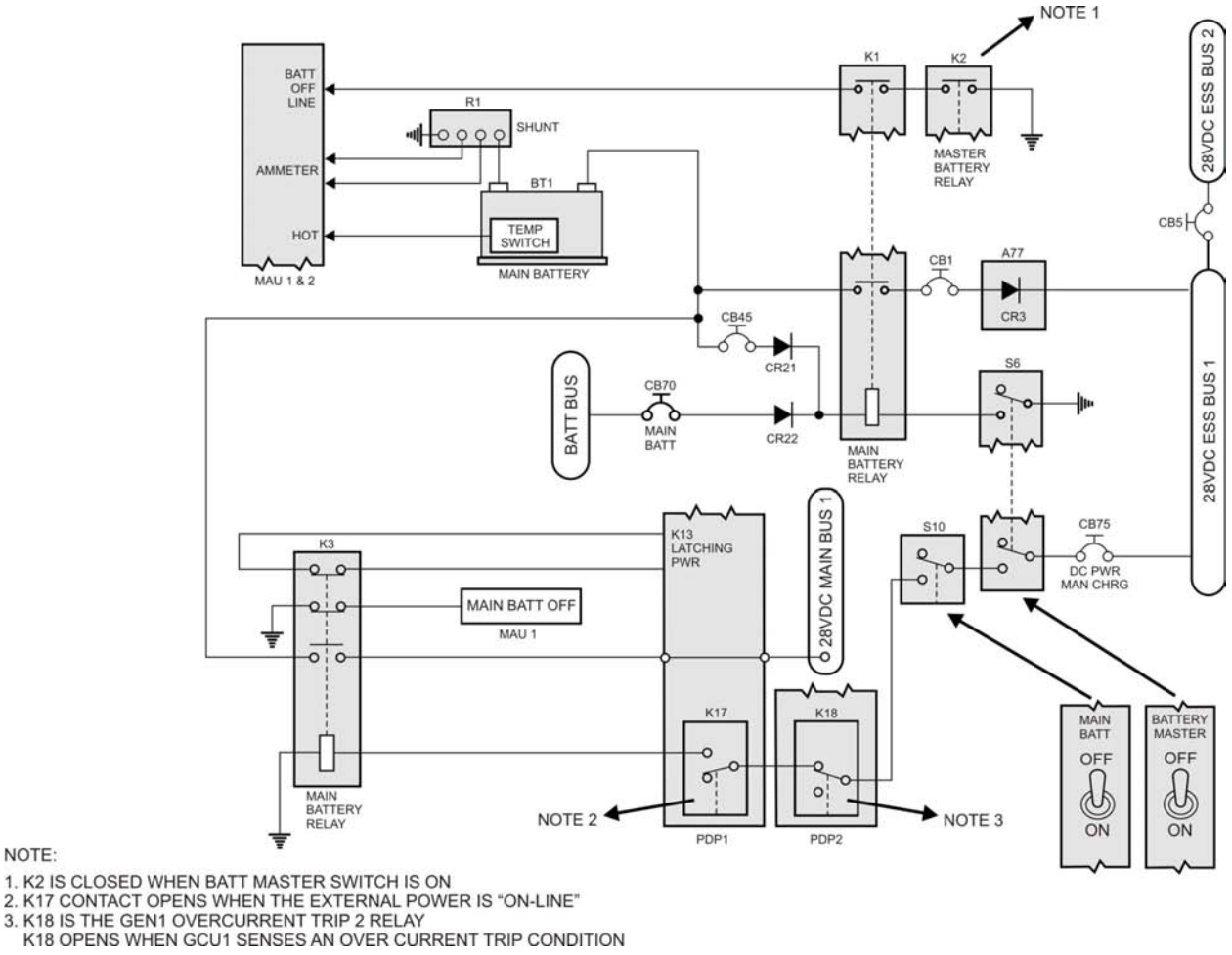
The diode CR4 (part of the diode module A76) connects the MAIN BUS2 to the Number 2 essential bus.
The heat sink (part of the diode module) removes the heat from the diode when the diode operates, and thus prevents over-temperature conditions.
The diode CR5 connects the auxiliary battery to the Number 2 main bus through the primary contact of the relay K4.
The diode has an inverse polarity. Thus the diode lets the auxiliary battery be connected to the main bus only during the time the battery is charged.

SYSTEM OPERATION

MAIN BATTERY POWER GENERATION AND DISTRIBUTION

The main battery power generation and distribution system does two primary functions:

- Main battery connection to the Number 1 essential bus
- Main battery connection to the Number 1 main bus.



NOTE:

1. K2 IS CLOSED WHEN BATT MASTER SWITCH IS ON
2. K17 CONTACT OPENS WHEN THE EXTERNAL POWER IS "ON-LINE"
3. K18 IS THE GEN1 OVERCURRENT TRIP 2 RELAY
   K18 OPENS WHEN GCU1 SENSES AN OVER CURRENT TRIP CONDITION

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MAIN BATTERY POWER GENERATION SYSTEM SCHEMATIC

MAIN BATTERY CONNECTION TO THE NUMBER 1 ESSENTIAL BUS

When the plug is connected to the main battery connector, the positive pin of the battery energizes the relay K1.
The coil of relay K1 is connected to the positive terminal of the battery through:

- The circuit breaker CB45
- The diode CR21.

When the BATTERY MASTER switch is at ON, a ground signal is applied to the negative pin of the coil of relay K1. This ground signal is necessary to energize the relay.
The main contactor closes, and the battery becomes connected to the Number 1 essential bus through:

- The circuit breaker CB1
- The diode CR3.

The battery thus supplies all the DC essential loads of the helicopter.
The shunt R1 senses the current of the battery. The voltage from the shunt is sent to the two MAUs. It is used to cause the ammeter indications to come in view on the MFDs.
A temperature switch installed in the battery senses a possible over-temperature condition of the battery. The signal from the switch is sent to the MAUs. It is used to cause the MAIN BATT HOT cautions to come on in the MFDs.
With the BATTERY MASTER switch set to ON, a ground signal is also applied to the negative pin of the relay K2 that closes its contacts. In this condition, if the auxiliary battery is connected to the circuit, the BATT OFF LINE caution goes off on the MFD. This shows that the two batteries are connected to the essential busses.

MAIN BATTERY CONNECTION TO NUMBER 1 MAIN BUS

The main battery can be connected to the Number 1 main bus only when:

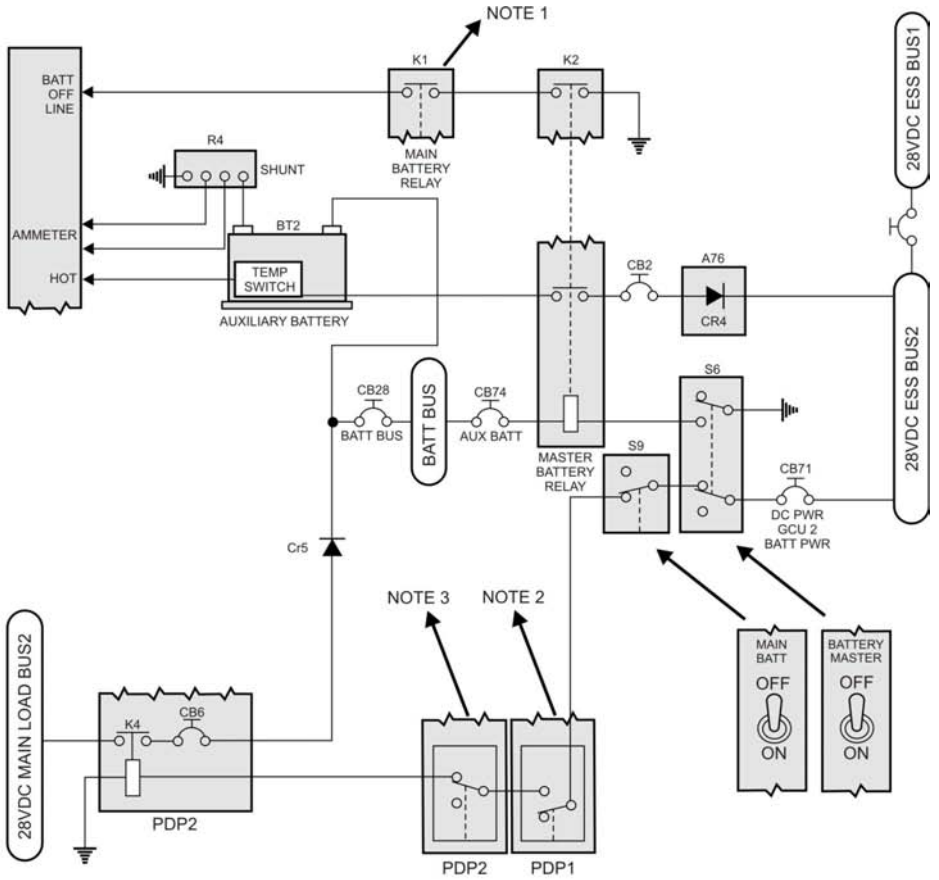
- The BATTERY MASTER switch is set to ON and the coil of the relay K3 is energized
- The MAIN BATT switch is set to ON
- The GCU1 does not give an indication of generator 1 over-current condition (K18 closed)
- The external power source does not supply electrical power to the helicopter (K17 closed).

Thus the positive pin of the main battery is connected to the Number 1 main bus through the Number 1 power-distribution panel (PDP1).

AUXILIARY BATTERY POWER GENERATION AND DISTRIBUTION

The power generation and distribution system of the auxiliary battery does three primary functions:

- The connection of the auxiliary battery to the BATT BUS
- The connection of the auxiliary battery to the Number 2 essential bus
- The connection of the auxiliary battery to the Number 2 main bus.



NOTE:
1. K1 IS CLOSED WHEN BATT MASTER SWITCH IS ON
2. K17 CONTACT OPENS WHEN THE EXTERNAL POWER IS "ON-LINE"
3. K20 IS THE GEN2 OVERCURRENT TRIP 2 RELAY
   K20 OPENS WHEN GCU2 SENSES AN OVER CURRENT TRIP CONDITION

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AUXILIARY POWER GENERATOR SYSTEM-PRIMARY COMPONENT (SHEET 2 OF 2)

AUXILIARY BATTERY CONNECTION TO THE BATT BUS

The auxiliary battery supplies electrical power to the BATT BUS when the auxiliary battery connector is connected. Power supply occurs through the circuit breaker BATT BUS.

AUXILIARY BATTERY CONNECTION TO THE NUMBER 2 ESSENTIAL BUS

The BATT BUS is energized when:

- The plug is connected to the auxiliary battery connector
- The main contactor of relay K2 is connected to the positive pin of the battery.

The coil of relay K2 is also energized by the BATT bus through the AUX BATT circuit breaker.
The negative pin of the coil of relay K2 is grounded when the BATTERY MASTER switch is at ON. Thus the relay becomes energized. The main contactor closes, and the battery is connected to the Number 2 essential bus. This connection occurs through the circuit breaker CB1 and the diode CR4.
Thus the auxiliary battery supplies all the DC essential loads of the helicopter.
The shunt R4 senses the current of the battery. The voltage from the shunt is sent to the two MAUs. It is used to cause the ammeter indications to come in view on the MFDs.
A temperature switch installed in the battery senses the possible over-temperature conditions in the battery. The signal from the switch is sent to the MAUs. It is used to cause the AUX BATT HOT caution to come on the MFDs.
If the main battery switch is set to ON, the K1 relay closes because the contacts of K1 and K2 are closed. In these conditions, the BATT-OFF LINE caution goes off on the MFDs. This shows that the two batteries are connected to the essential busses.

AUXILIARY BATTERY CONNECTION TO THE NUMBER 2 MAIN BUS

The auxiliary battery can be connected to the Number 2 main bus only when the BATTERY MASTER switch is set to ON.
The coil of relay K4 is energized when:

- The BATTERY MASTER switch is to ON
- The AUX BATT switch is to ON
- The GCU2 does not give an indication of generator 2 over-current condition (K20 closed)
- The external power source does not supply electrical power to the helicopter (K17 closed).

Thus the positive pin of the auxiliary battery is connected to the Number 2 main bus through:

- The diode CR5
- The circuit breaker CB6.

In this condition, the power generation system of the helicopter charges the auxiliary battery through Number 2 main bus.
This prevents the supply of power from the auxiliary battery to the Number 2 main bus when the battery is connected through the relay K4.

DC MAIN GENERATION SYSTEM DESCRIPTION

The DC main generation system includes these primary components:

- The Number 1 starter generator (G1)
- The Number 1 starter-generator mount-flange
- The Number 1 starter-generator air-inlet duct
- The Number 1 starter-generator air-outlet duct
- The Number 1 generator-control unit (A13)

- The Number 1 shunt (R3)
- The Number 2 starter generator (G2)
- The Number 2 starter-generator mount-flange
- The Number 2 starter-generator air-inlet duct
- The Number 2 starter-generator air-outlet duct
- The Number 2 generator-control unit (A14)
- The Number 2 shunt (R2)
- The Number 3 and Number 4 DC current-transformers (T1 and T2)
- The Number 1 and Number 2 DC current-transformers (T3 and T4)
- The GEN 1 and GEN 2 switches (S66 and S58).

NUMBER 1 STARTER GENERATOR

The Number 1 starter generator (G1) is installed on the accessory gearbox of the Number 1 engine. It is attached to a starter-generator mount-flange through a V-band clamp. Thus you can remove and install the starter generator quickly.
The starter generator has a drive shaft that engages the accessory gearbox of the engine. This is the mechanical coupling between the starter generator and the engine.
The drive shaft has a shear neck. This shear neck is the protection of the drive train of the accessory gearbox. It breaks when the applied torque becomes too much.
The primary components of the 300 A DC starter generator number 1 are the stator and the rotor. They are contained in a cylindrical housing.
The housing is made from an iron alloy and is a part of the magnetic circuit of the starter assembly.
The rotor assembly includes a winding the interface of which is the brush/ switching assembly that connects it to the housing.
The generator is of the 4-pole type. It has a four-brush block-assembly that applies the input and output current to the correct rotor winding.
Ball bearings installed in the housing hold the rotor at each end.
The bearings are of the pre-lubricated type and no maintenance is necessary between overhauls.
Air keeps the temperature of the starter generator at the correct value.
A fan installed at the starter generator end opposite the input spline gives the flow of air necessary to keep the assembly cool. The air flows axially on the switching assembly and through the rotor and starter windings. It is then released overboard through the Number 1 starter-generator air-outlet duct.
The same air also decreases the temperature of the bearings that hold the rotor.

NUMBER 1 STARTER GENERATOR MOUNT FLANGE

The Number 1 starter-generator mount-flange is installed on the drive pad of the Number 1 engine. It is attached to four studs of the engine drive pad with four washers and four nuts.
The mount flange also has the opening that lets the cooling air go into the airframe duct to be then released overboard through the Number 1 air-outlet duct.
A bonding wire is attached on the top of the mount flange.

NUMBER 1 STARTER GENERATOR AIR INLET DUCT

The Number 1 starter-generator air-inlet duct is installed on the inlet port of the starter generator. A V-band attaches the air inlet duct to the inlet port.

NUMBER 1 STARTER GENERATOR AIR OUTLET DUCT

The Number 1 starter-generator air-outlet duct is installed on the outlet port of the mount flange. Five bolts and five washers attach the air outlet duct to the mount flange.

NUMBER 1 GENERATOR CONTROL UNIT

The Number 1 Generator Control Unit (GCU1) (A13) is installed on the top left side of the avionic bay at STA X6849. Four screws and four washers attach the GCU1 to the avionic bay structure.

NUMBER 1 SHUNT

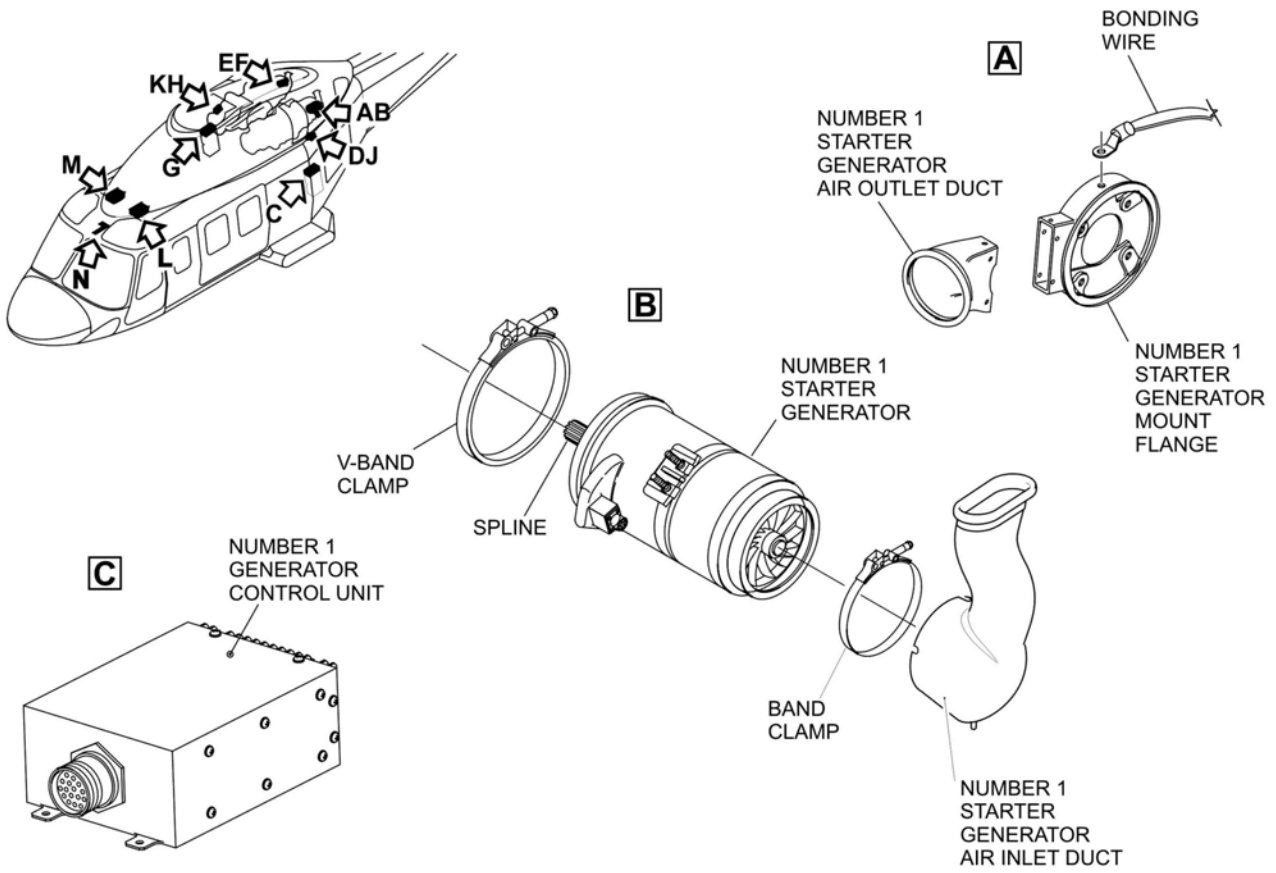
The Number 1 shunt (R3) is installed on the top left side of the aft fuselage at STA X7382. Two screws attach it to a support made of insulating material. Two nuts attach the terminals of the power supply cables to the shunt.

NUMBER 2 STARTER GENERATOR

The Number 2 starter generator (G2) is installed on the accessory gearbox of the Number 1 engine. It is attached to a starter-generator mount-flange through a V-band clamp. Thus you can remove and install the starter generator quickly.
The starter generator has a drive shaft that engages the accessory gearbox of the engine. This is the mechanical coupling between the starter generator and the engine.
The drive shaft has a shear neck. This shear neck is the protection of the drive train of the accessory gearbox. It breaks when the applied torque becomes too much.
The primary components of the 300 A DC starter generator Number 1 are the stator and the rotor. They are contained in a cylindrical housing.
The housing is made from an iron alloy and is a part of the magnetic circuit of the starter assembly.
The rotor assembly includes a winding the interface of which is the brush/switching assembly that connects it to the housing.
The generator is of the 4-pole type. It has a four-brush block-assembly that applies the input and output current to the correct rotor winding.
Ball bearings installed in the housing hold the rotor at each end.
The bearings are of the pre-lubricated type and no maintenance is necessary between overhauls.
Air keeps the temperature of the starter generator at the correct value.
A fan installed at the starter generator end opposite the input spline gives the flow of air necessary to keep the assembly cool. The air flows axially on the switching assembly and through the rotor and starter windings. It is then released overboard through the Number 1 starter-generator air-outlet duct.
The same air also decreases the temperature of the bearings that hold the rotor.

NUMBER 2 STARTER GENERATOR MOUNT FLANGE

The Number 2 starter-generator mount-flange is installed on the drive pad of the Number 2 engine. It is attached to four studs of the engine drive pad with four washers and four nuts.
The mount flange also has the opening that lets the cooling air go into the airframe duct to be then released overboard through the Number 2 air-outlet duct.
A bonding wire is attached on the top of the mount flange.



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DC MAIN GENERATION PRIMARY COMPONENTS (SHEET 1 OF 3)

NUMBER 2 STARTER GENERATOR AIR INLET DUCT

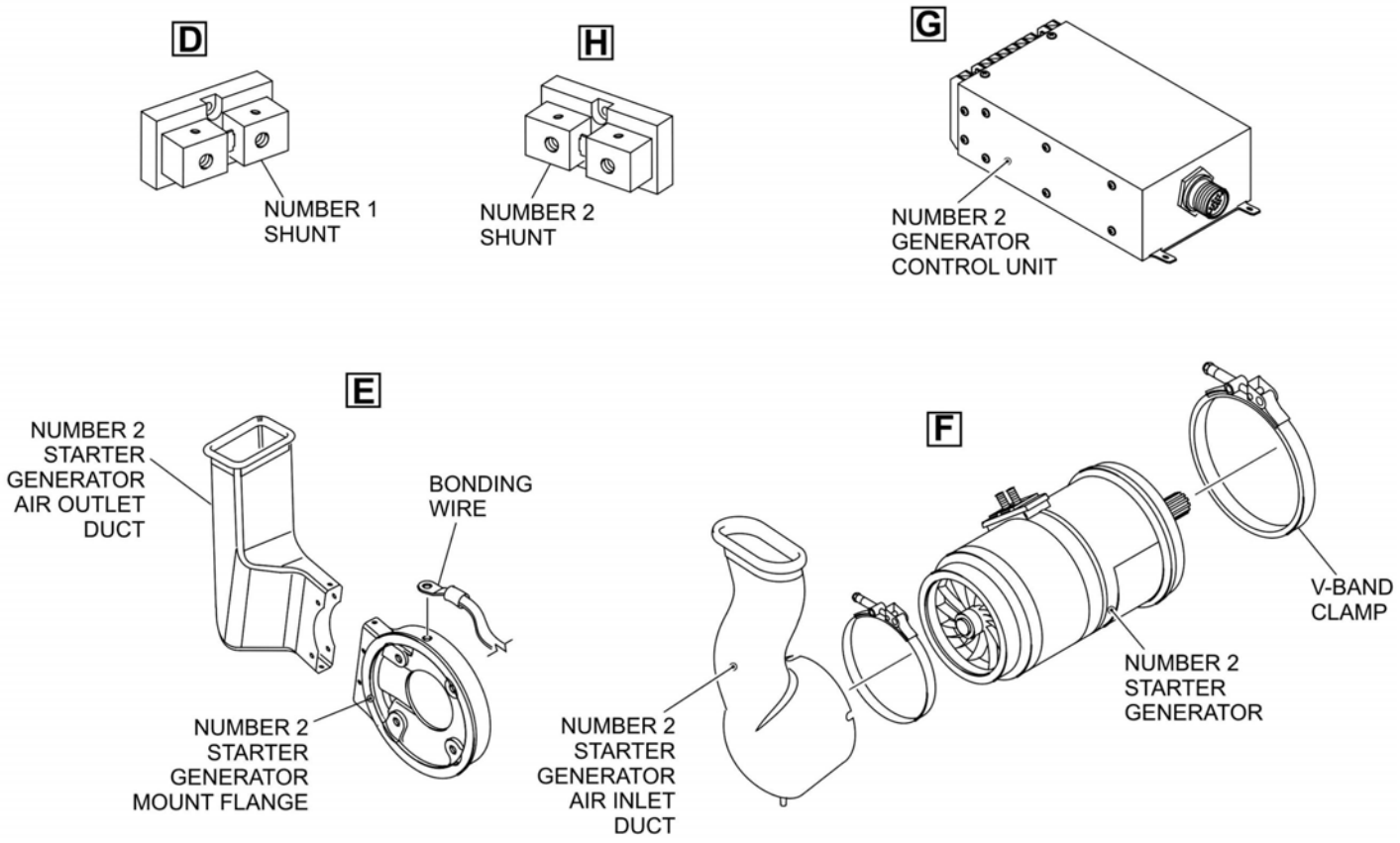
The Number 2 starter-generator air-inlet duct is installed on the inlet port of the starter generator. A V-band attaches the air inlet duct to the inlet port.

NUMBER 2 STARTER GENERATOR AIR OUTLET DUCT

The Number 2 starter-generator air-outlet duct is installed on the outlet port of the mount flange. Five bolts and five washers attach the air outlet duct to the mount flange.

NUMBER 2 GENERATOR CONTROL UNIT

The Number 2 Generator Control Unit (GCU2) (A14) is installed in the top right side of the avionic bay at STA X 7073. Four screws and four washers attach the GCU2 to a support of the avionic bay.



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DC MAIN GENERATION PRIMARY COMPONENTS (SHEET 2 OF 3)

NUMBER 2 SHUNT

The Number 2 shunt (R2) is installed on the top right side of the aft fuselage at STA X7382. Two screws attach it to a support made of insulating material. Two nuts attach the terminals of the power supply cables to the shunt.

NUMBER 3 AND NUMBER 4 DC CURRENT TRANSFORMERS

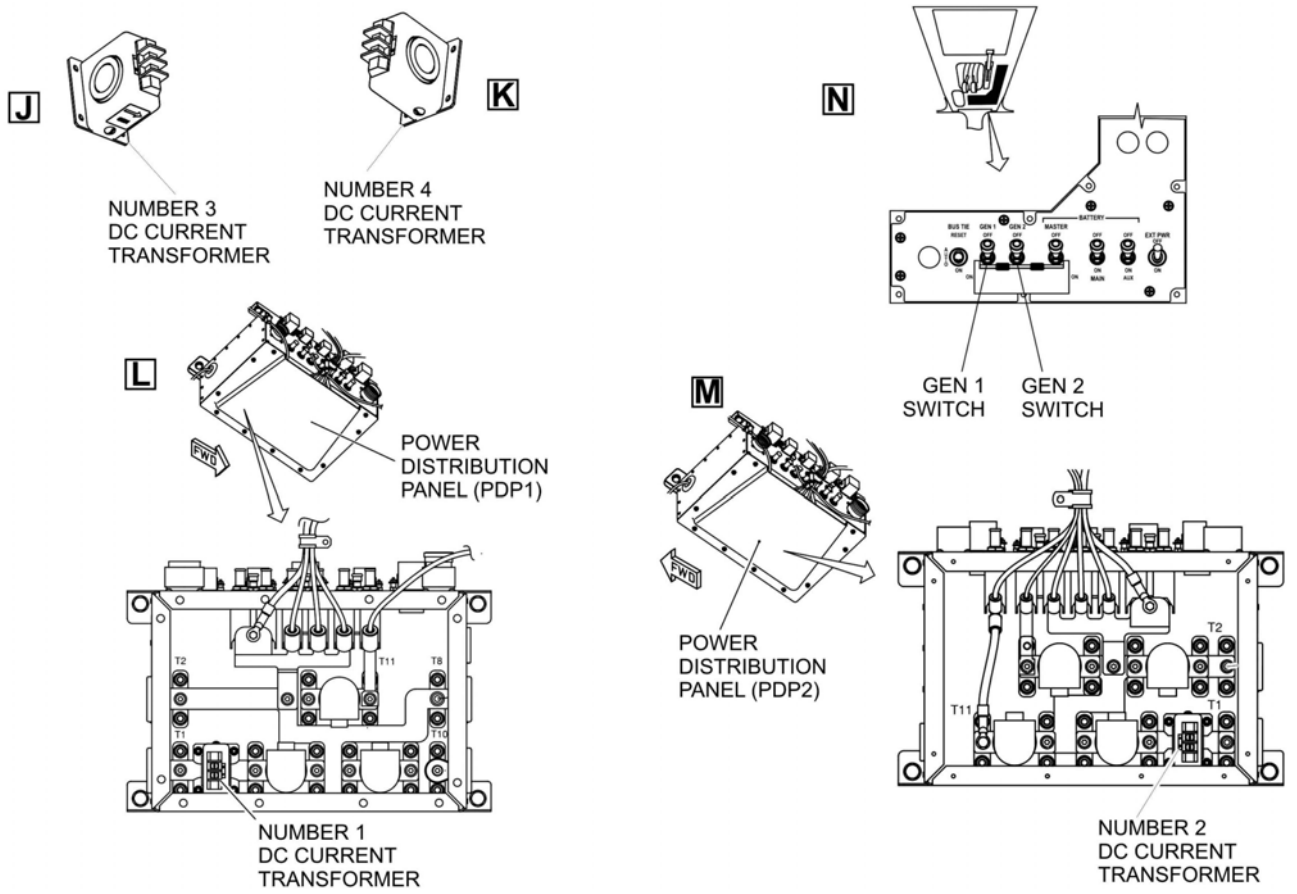
The Number 3 and Number 4 DC current-transformer (T1) and (T2) are installed near the related shunts.
The transformers are of the annular type. Their output is supplied to the power cable and gives a monitoring signal. The power cable goes through the current transformer. It is installed in the best position to let the system monitor the current.
The current transformers are installed on the support, and four screws attach each of them. Access to the transformers for maintenance and inspection is easy.
The T1 transformer is installed on the right side of the avionic equipment bay.
The T2 transformer is installed on the left side of the avionic equipment bay.

NUMBER 1 AND NUMBER 2 DC CURRENT TRANSFORMERS

The Number 1 DC current-transformer (T3) is installed in the Number 1 Power Distribution Panel (PDP1).
The Number 2 DC current-transformer (T4) is installed in the Number 2 Power Distribution Panel (PDP2).
The transformers are of the annular type. Their output is supplied to the power cable and gives a monitoring signal. The power cable goes through the current transformer. It is installed in the best position to let the system monitor the current.

GEN 1 AND GEN 2 SWITCHES

The GEN 1 and GEN 2 switches are installed on the right side at the bottom of the electrical control panel.
The electrical control panel is a part of the overhead panel in the cockpit.
The GEN 1 and GEN 2 switches are of the ON/OFF type.



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DC MAIN GENERATION PRIMARY COMPONENTS (SHEET 3 OF 3)

DC MAIN GENERATION SYSTEM – DESCRIPTION OF FUNCTION

NUMBER 1 STARTER GENERATOR

The Number 1 starter generator (G1) is of the 4-pole type, and its design speed is 12,000 rpm.
It operates as a starter or a generator.
The windings of the interpole are a part of the electrical circuit of the DC starter generator. They increase the flux path between the stator and the rotor to make the operation of the starter generator better.
The generator control unit controls the starter generator electrically through the supply of a control current to the shunt field winding.

NUMBER 1 STARTER GENERATOR MOUNT FLANGE

The Number 1 starter-generator mount-flange is used to connect the Number 1 starter generator to the engine. It also lets the cooling air to be released overboard through the Number 1 air-outlet duct.

NUMBER 1 STARTER GENERATOR AIR INLET DUCT

The Number 1 starter-generator air-inlet duct supplies ambient air to the inlet port of the starter generator. The air then flows axially on the switching assembly and through the rotor and starter windings. The same air also decreases the temperature of the bearings that hold the rotor.

NUMBER 1 STARTER GENERATOR AIR OUTLET DUCT

The Number 1 starter-generator air-outlet duct releases overboard the flow of air used to cool the starter generator assembly.

NUMBER 1 GENERATOR CONTROL UNIT

The Number 1 Generator Control Unit (GCU1) (A13) monitors and controls the starter generator during engine start and when it operates as a generator.
It is also the protection of the system from damage when a failure occurs.

GENERATOR VOLTAGE CONTROL

When: – The engine turns at the self-sustaining speed – The driven generator voltage is available – The field control relays open the internal-field build-up circuit.
Then, the regulating field output is connected to the circuit. The voltage regulator supplies it.
The voltage regulator is a control circuit of the closed-loop type. It adjusts the duty cycle of the exciter field. Thus the voltage at the Point Of Regulation (POR) is the same also when the loads change.
When: – The generator output is voltage is less than 0.5 V lower than the bus voltage – The GEN switch is at ON.
The GCU closes the related Generator Line Contactor (GLC), and the generator output becomes connected to the main bus.

Protection

The GCU is a protection for the system. When it senses an incorrect condition, it removes power from the starter generator, and disconnects it from the bus through the GLC.

This occurs if there is a condition of:

Overvoltage

The overvoltage protection operates when the generator voltage is higher than the limit.

Undervoltage

The undervoltage protection operates when the generator voltage is less than 5 V.

Feeder Fault

The feeder fault protection operates when the differential feeder current is higher than 125 A (nominal value).

Bus Fault

The Bus Fault signal is transmitted when the GCU senses an overcurrent condition in the generator. The overcurrent value must be higher than 550 A.

NOTE A

If the bus protection logic of the PDP Number 1 does not remove the failure in 6 seconds, the GCU disconnects the generator. The disconnection occurs 6 to 7 seconds from the start of the overcurrent condition.

Reverse Current

The reverse current protection operates when the GCU senses a reverse current higher than 10% of the rated load. The generator must be in the normal generation mode.

EXTERNAL POWER INHIBIT FUNCTION

When the external power is connected and supplied to the helicopter, the GCU receives a related signal this to prevent the supply of DC power from the external power source and the generator at the same time.
If the signal is received when the generator is not connected to the bus, the GCU opens the Generator Line Contactor (GLC). The GCL stays opened during the full period in which the external power is supplied.
If the signal is received when the generator is connected to the bus, the GCU opens the GLC.
When the GLC opens, the generator becomes disconnected from the bus.

NUMBER 1 SHUNT

The Number 1 shunt (R3) measures the DC current output of the Number 1 generator.
The Number 1 Modular Avionic Unit (MAU1) is the interface of the DC current signal. Thus the AMS can send the signal to the multifunctional displays to cause the indication of the DC ammeter to come in view.

NUMBER 3 DC CURRENT TRANSFORMER

The Number 3 DC current-transformer (T1) measures the changes in the DC current. The signal is sent to the Number 1 generator-control unit that uses this signal to control the generator and the generator line contactor.

NUMBER 2 STARTER GENERATOR

For the functional description of the Number 2 starter generator (G2) refer to the Number 1 starter generator (G1).

NUMBER 2 STARTER GENERATOR MOUNT FLANGE

The Number 2 starter-generator mount-flange is used to connect the Number 2 starter generator to the engine. It also lets the cooling air to be released overboard through the Number 2 air-outlet duct.

NUMBER 2 STARTER GENERATOR AIR INLET DUCT

The Number 2 starter-generator air-inlet duct supplies ambient air to the inlet port of the starter generator. The air then flows axially on the switching assembly and through the rotor and starter windings. The same air also decreases the temperature of the bearings that hold the rotor.

NUMBER 2 STARTER GENERATOR AIR OUTLET DUCT

The Number 2 starter-generator air-outlet duct releases overboard the flow of air used to cool the starter generator assembly.

NUMBER 2 GENERATOR CONTROL UNIT

For the functional description of the Number 2 Generator Control Unit (GCU2) (A14) refer to the Number 1 Generator Control Unit (GCU1).

NUMBER 2 SHUNT

The Number 2 shunt (R2) measures the DC current output of the Number 2 generator.
The Number 2 Modular Avionic Unit (MAU2) is the interface of the DC current signal. Thus the Aircraft Monitoring System (AMS) can send the signal to the multifunctional displays (MFDs) to cause the indication of the DC ammeter to come in view.

NUMBER 4 DC CURRENT TRANSFORMER

The Number 4 DC current-transformer (T2) measures the changes in the DC current. The signal is sent to the Number 2 generator-control unit that uses this signal to control the generator and the generator line contactor.

ELECTRICAL CONTROL PANEL

The electrical control panel of the DC main generation system (installed on the overhead console in the cockpit) has the GEN 1 (S66) and GEN 2 (S58) switches. The switches are of the two-position type ON and OFF.

ELECTRICAL POWER – SIMPLIFIED SCHEMATIC DIAGRAM

A simplified schematic diagram is used to describe the operation of the electrical power system in the following normal and emergency situations:

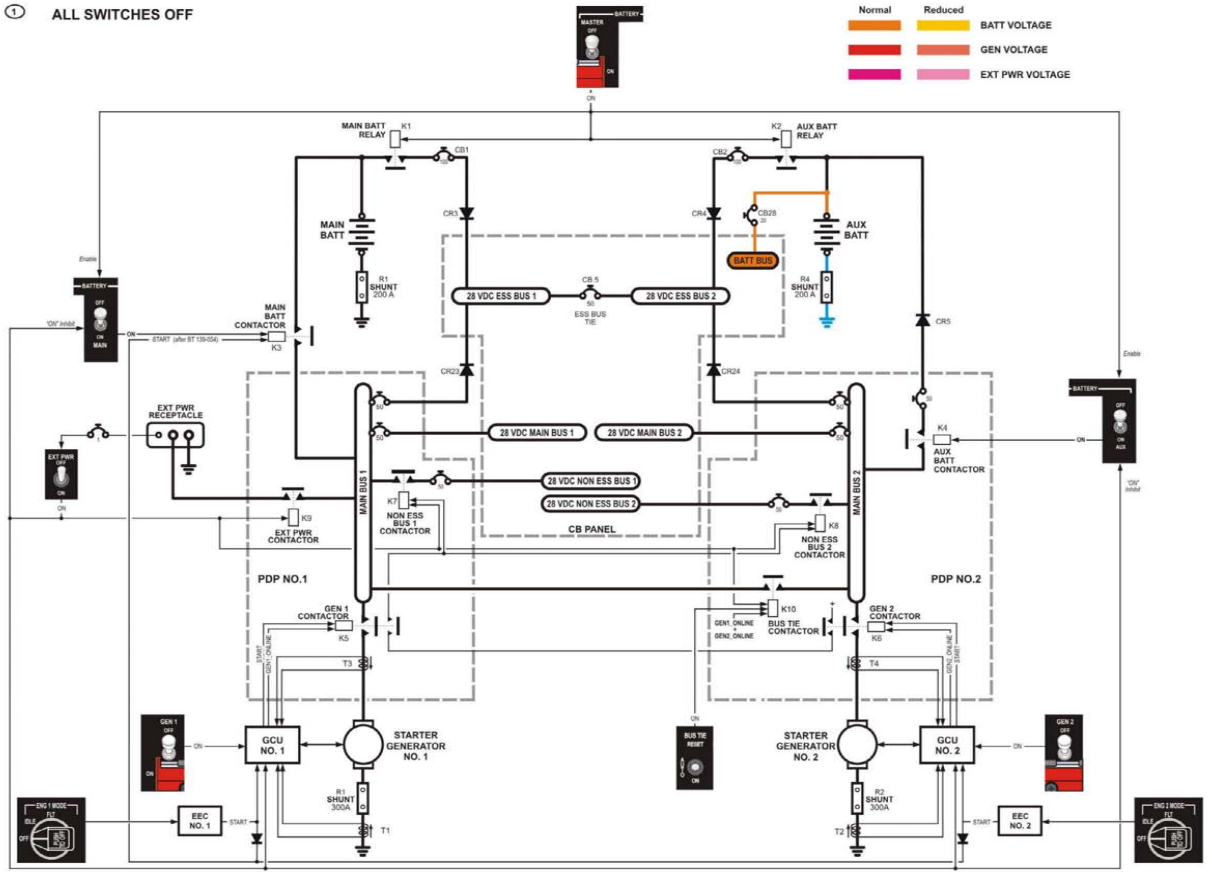
- Normal procedures:

- Engine starting on batteries (BATTERY Starting)
- Engine starting on external power (EXT PWR Starting)
- In-flight normal operation

- Emergency procedures:
  - Single Generator Failure
  - Single DC Bus Failure
  - Dual Generator Failure

BATTERY STARTING

The following diagrams represent the sequence of actions for a normal battery starting of the engines (steps 1 to 8 on the simplified schematic diagram).
In this example engine no. 2 is started first.
Refer to AW139-RFM-4D – Section 2 for Normal Procedures.



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STEP 1 – ALL SWITCHES OFF

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STEPS 2 TO 3

When BATTERY MASTER switch is set to ON, Main and Aux Batteries are connected to ESS BUS 1 and ESS BUS 2 thus providing power to the rotorcraft essential loads.
When BATTERY MAIN switch is set to ON, the Main Battery is connected to MAIN BUS 1 (Main Battery contactor K3 closes) which is then also powered.
When BATTERY AUX switch is set to ON, the Auxiliary Battery is connected to MAIN BUS 2 (Aux Battery contactor K4 closes), but MAIN BUS 2 is not powered because of the reverse biased diode (CR5) which only permits recharging of the Auxiliary battery from MAIN BUS 2.
When GEN 1 and GEN 2 switches are set to ON they give an input to the relevant GCU so that the GCU will put the relevant generator on-line as soon as conditions permit.

23 VOLT CHECK

Before attempting starting the engine on batteries, the pilot has to check that the involved MAIN BUS voltage is not less than 23 V.

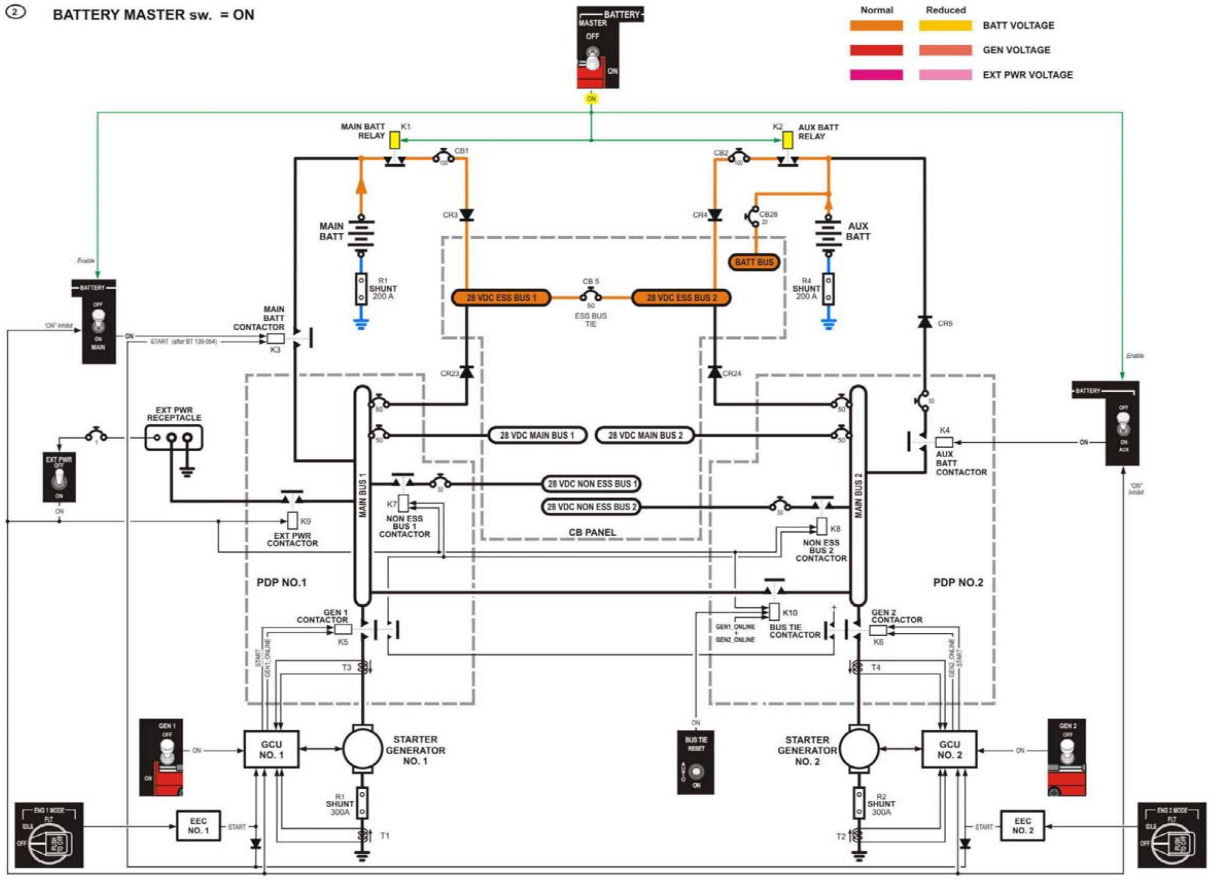
STEP 4

As in this example the engine no 2 is started first, the BUS TIE switch must be set to ON to power MAIN BUS 2.
BUS TIE Contactor closes thus connecting MAIN BUS 1 and MAIN BUS 2.

BUS TIE LOGIC

BUS TIE contactor closes if any of the following conditions is met:

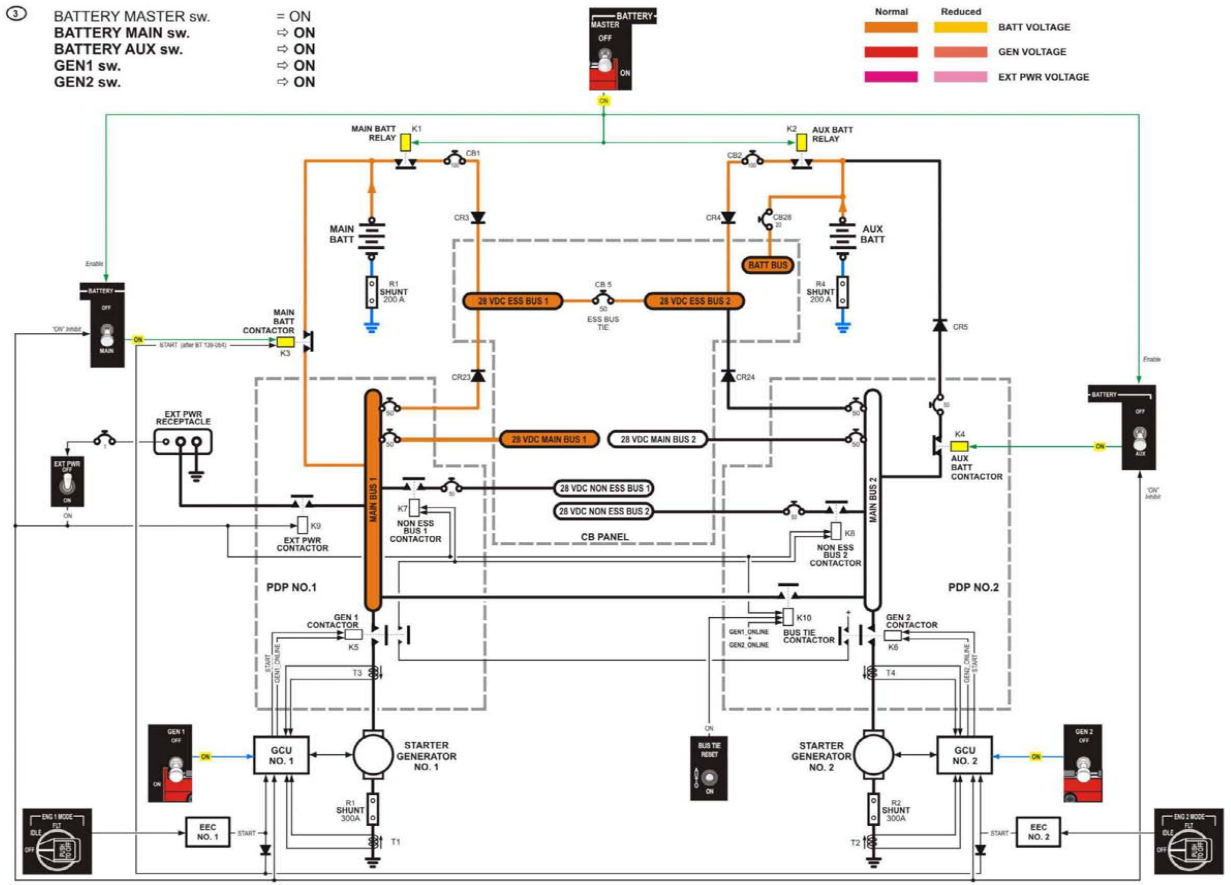
- BUS TIE switch = ON
- One GEN = ON-LINE and the other GEN = OFF-LINE
- EXT PWR is connected to helicopter bus bars



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STEP 2 – BATTERY MASTER SWITCH ON

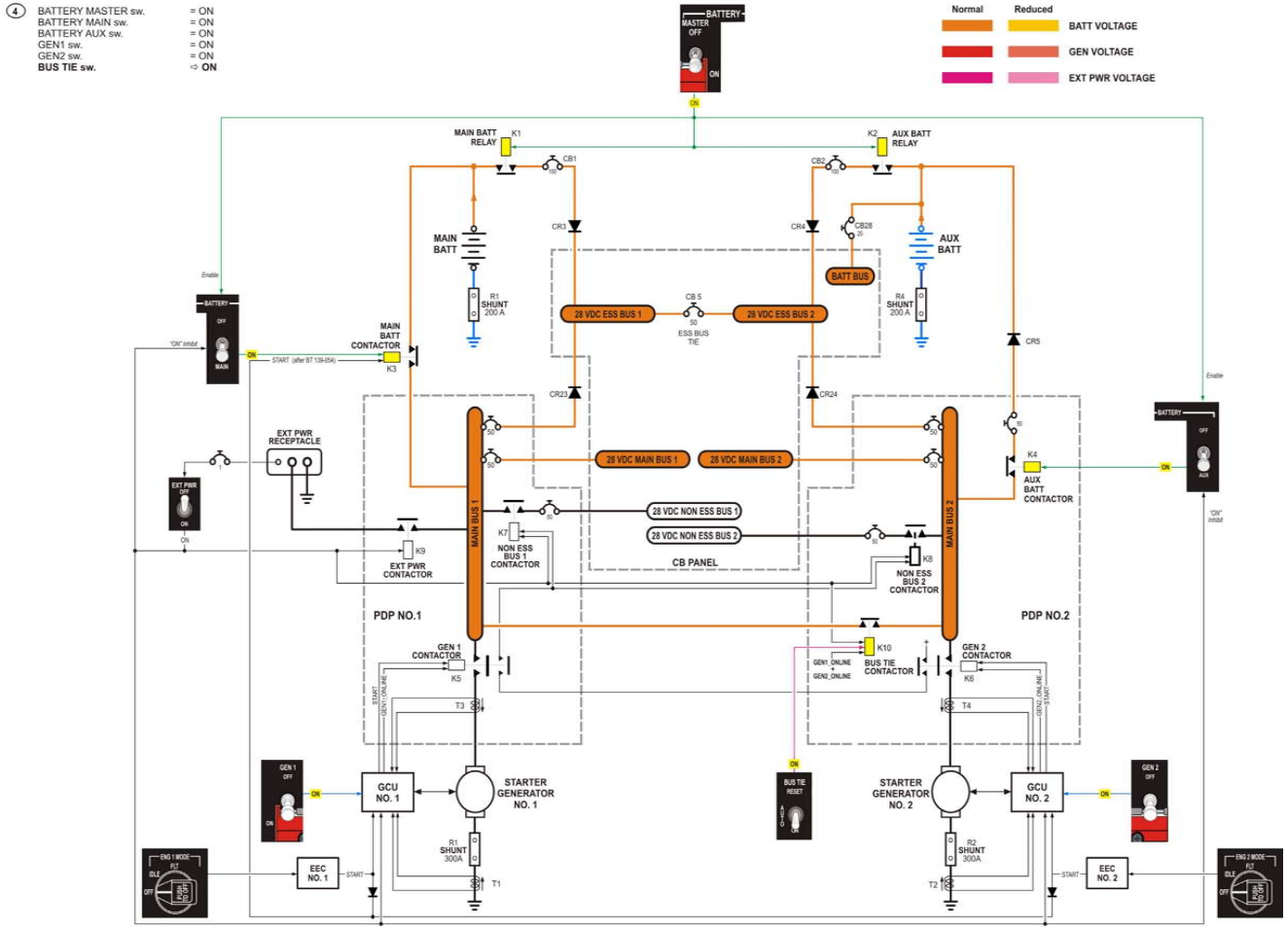
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STEP 3 – MAIN AND AUX BATTERY SWITCHES ON

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STEP 4 – BUS TIE SWITCH ON

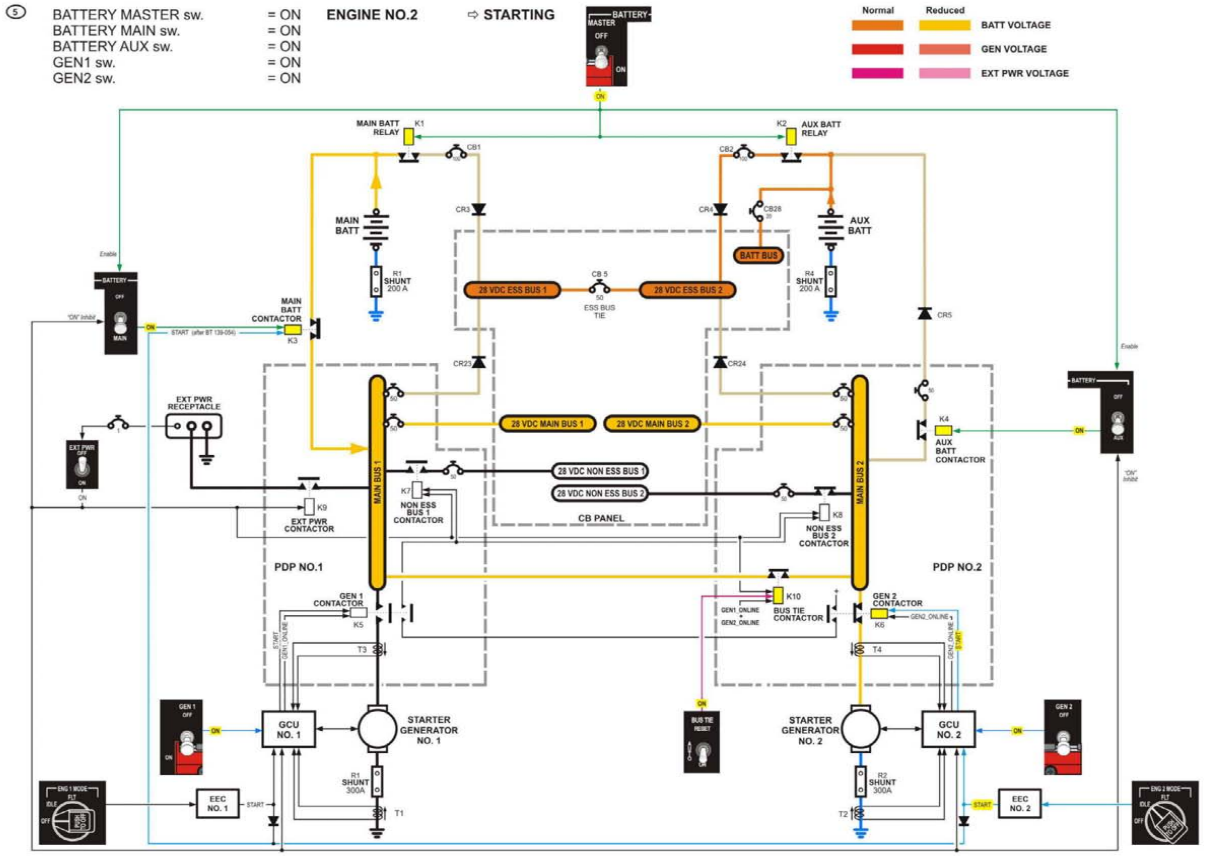
(Printed A3 format at the end of data module)

STEP 5 – ENGINE 2 STARTING ON BATTERIES

An Engine normal start (AUTO Mode) is initiated by the Engine Electronic Unit (EEC) when pilot moves the relevant ENG MODE switch on the ENG control panel to IDLE or FLIGHT.
This causes the GCU to close the relevant Generator contactor so the starter is fed by the MAIN Battery power via MAIN BUS 2 and cranks the engine.
Due to the great amount of power required to start the engine, MAIN battery voltage drops very much (represented with a paler shade of color).
The AUX battery only supplies the ESS BUS 1 and 2, in order to keep a sufficient high voltage for the pilot Display Units (PFD and MFD): this ensures pilots can monitor engine parameters during starting.
Engine start sequence is terminated by the EEC when it detects Ng = 49%: the START signal is removed and the GCU releases the Generator contactor.
The EEC then accelerates the engine to Nf = 65% (IDLE) or Nf = 100% (FLIGHT).

NOTE B

An Engine Manual Start (MANUAL Mode) is initiated by pressing the START push-button on the relevant Engine Control Lever (ECL) which directly sends a control signal to the relevant GCU.
In this case the EEC is not involved and the start sequence is terminated by the GCU thanks to the Generator speed sensing signal.



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STEP 5 – ENGINE NO.2 STARTING

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STEP 6 – GEN 2 ON-LINE

When the starting sequence is terminated, since the GEN 2 switch is at ON, the GCU closes the Generator 2 contactor when 50% Ng is exceeded: GEN 2 is on-line.
GEN 2 feeds all MAIN and ESS busses and recharges the MAIN and AUX batteries.
The BUS TIE contactor is closed because of the following two conditions:
BUS TIE switch = ON
One GEN = ON-LINE and the other GEN = OFF-LINE

STEP 7 – ENGINE 1 STARTING: ASSISTED START

When the second engine is started on batteries (in the example engine no. 1), the generator that is already on-line assists the MAIN battery in feeding the starter: this condition is called assisted start.

STEP 8 – BOTH GENERATORS ON-LINE

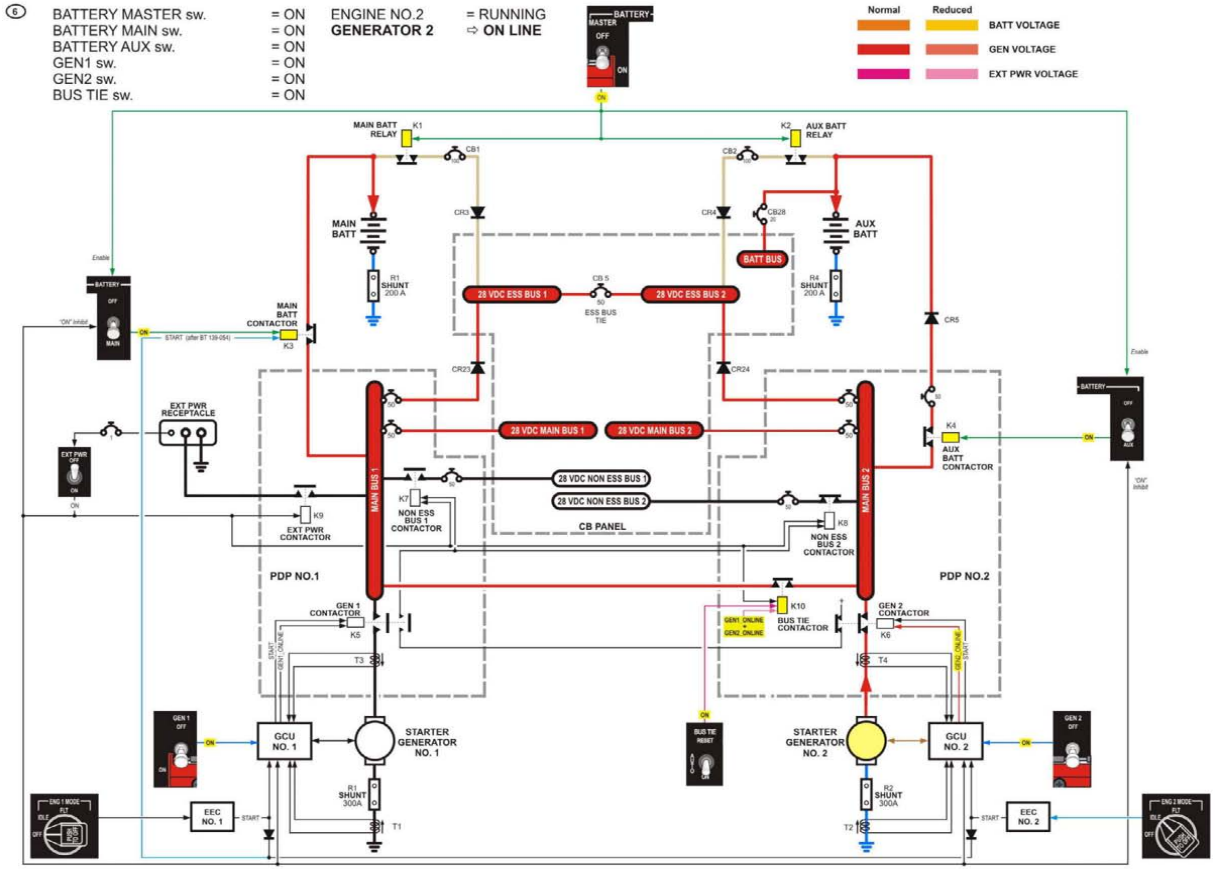
After the second engine is started, also the second generator is automatically connected on-line by the relevant GCU.
BUS TIE switch is then returned to AUTO for normal operation
With both generators on-line, either on the ground or during flight all DC busses are powered by the two generators separately:

- BUS TIE contactor is open
- NON ESS BUS 1 and NON ESS BUS 2 are energized
- The two batteries are charged by GEN 1 (MAIN battery) and GEN 2 (AUX battery)

NON ESS BUS LOGIC

NON ESS BUS 1 and NON ESS BUS 2 are energized if any of the following conditions is met:

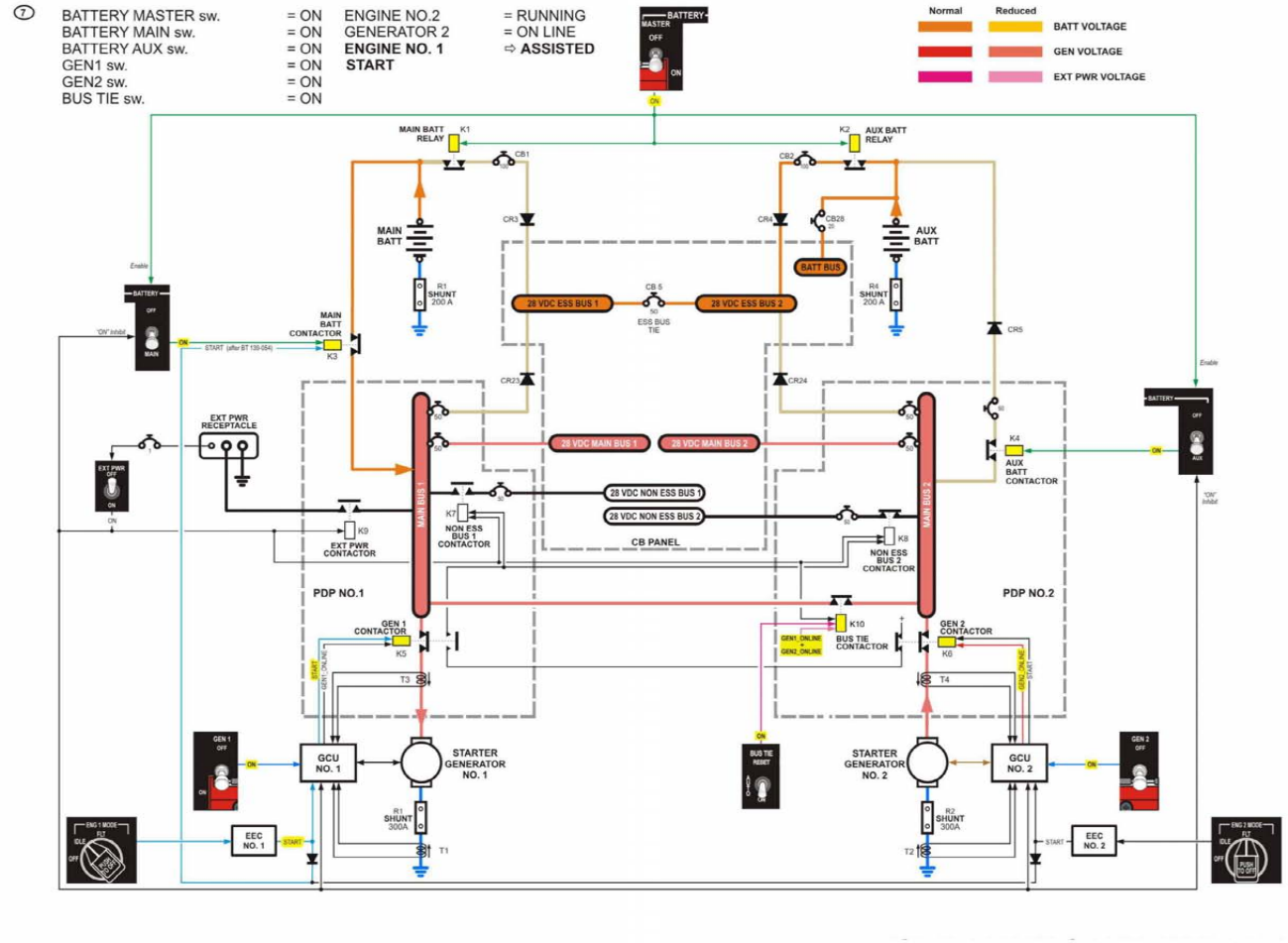
- Both GEN = ON-LINE
- EXT PWR is connected to helicopter bus bars



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STEP 6 – GEN 2 ON LINE

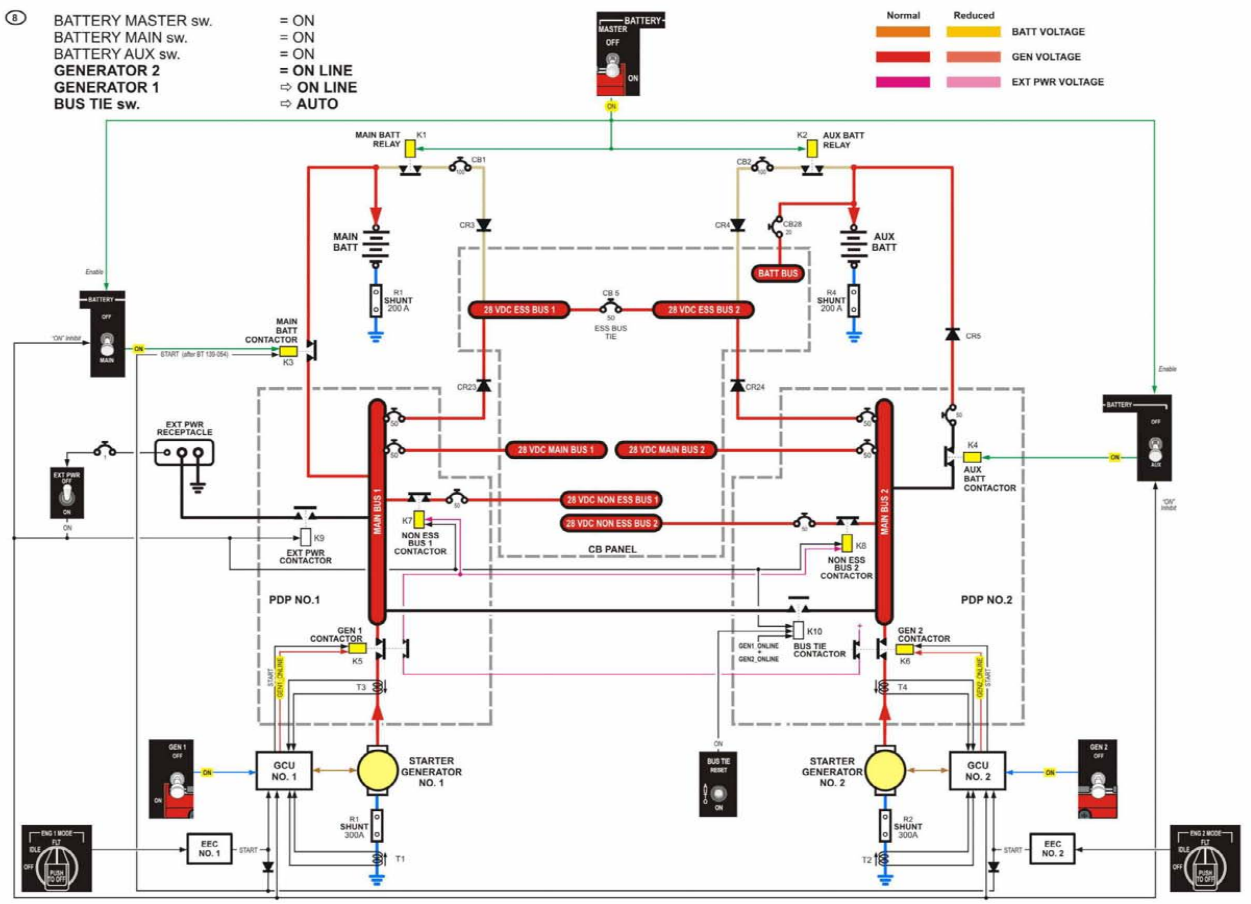
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STEP 7 – ENGINE NO.1 ASSISTED START

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STEP 8 – GEN 1 AND GEN 2 ON-LINE AND BUS TIE SWITCH AUTO

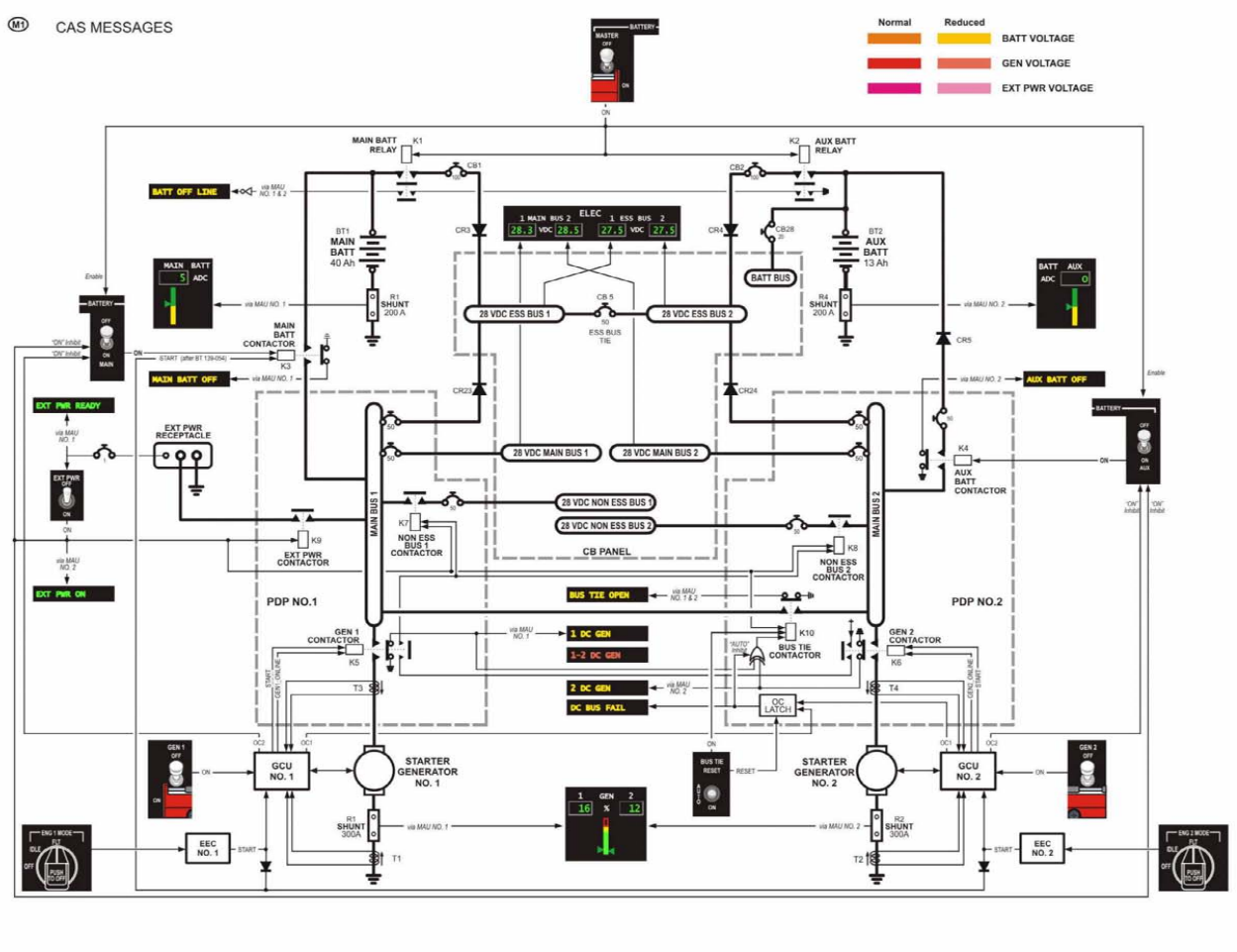
(Printed A3 format at the end of data module)

INDICATIONS AND IN – FLIGHT NORMAL OPERATION

The following simplified diagrams (M1, M2) integrate the previous diagram information with the interfacing of the Electrical power system indicators and the major CAS messages.
M1 is meant to just show the indications.
M2 shows the normal system operation and indications during flight.

NOTE C

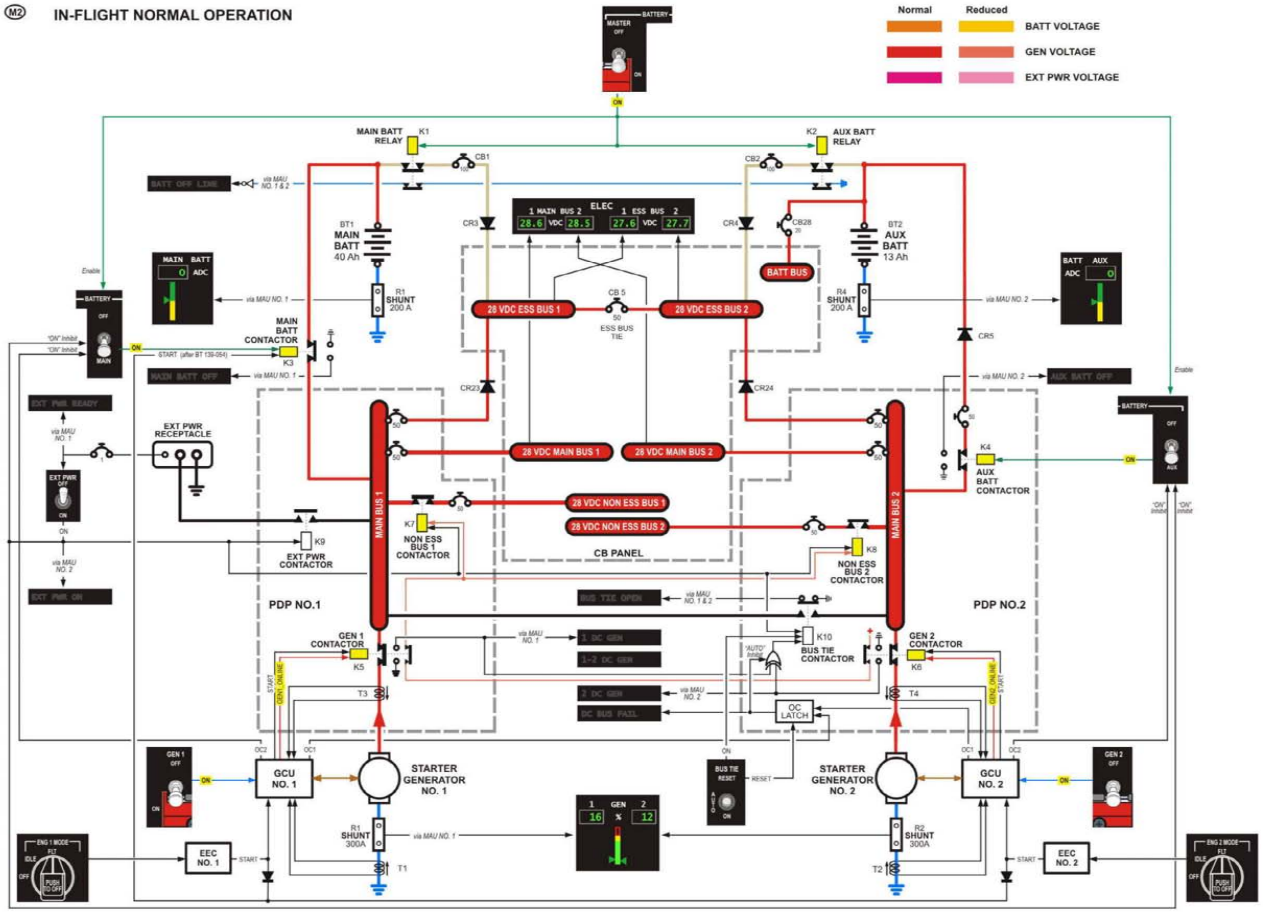
Values are only given as examples.



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M1 – CAS MESSAGES

(Printed A3 format at the end of data module)



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M2 – IN FLIGHT NORMAL OPERATION

(Printed A3 format at the end of data module)

IN FLIGHT EMERGENCIES AND MALFUNCTIONS

In the following pages the most significant emergencies and malfunctions are detailed using the simplified schematic diagram (F1 to F5). Refer to AW139-RFM-4D – Section 3 for Emergency and Malfunction procedures.

F1 – SINGLE GENERATOR FAILURE (1 DC GEN)

When a GCU detects that the relevant Generator has failed, it automatically open the Generator contactor.
The same CAS message is displayed when the GEN switch is moved to OFF while the relevant engine is running.

- BUS TIE contactor automatically closes
- NON ESS BUS 1 and NON ESS BUS 2 are disconnected

F2 – MOMENTARY OVERCURRENT CONDITION (DC BUS FAIL)

When a GCU detects an overcurrent exceeding a first threshold (OC1), it triggers a latching logic that:

- Drives the DC BUS FAIL message in the CAS
- Inhibits the BUS TIE contactor automatic operation (remains open)

Once triggered, the OC latching logic can only be reset by moving the BUS TIE switch to RESET.

F3 – SHORT CIRCUIT (DC BUS FAIL + 1 DC GEN)

If the overcurrent condition persists a second threshold is then exceeded (OC2): that is considered a short circuit.
In this case the relevant GCU also:

- disconnects the relevant Generator, causing the 1 DC GEN or the 2 DC GEN message to be displayed
- disconnects the relevant Battery to prevent discharging on the short circuit, causing the MAIN BATT OFF or the AUX BATT OFF message to be displayed

BUS TIE contactor remains open to prevent the failed bus from being connected to the only available generator.

NOTE D

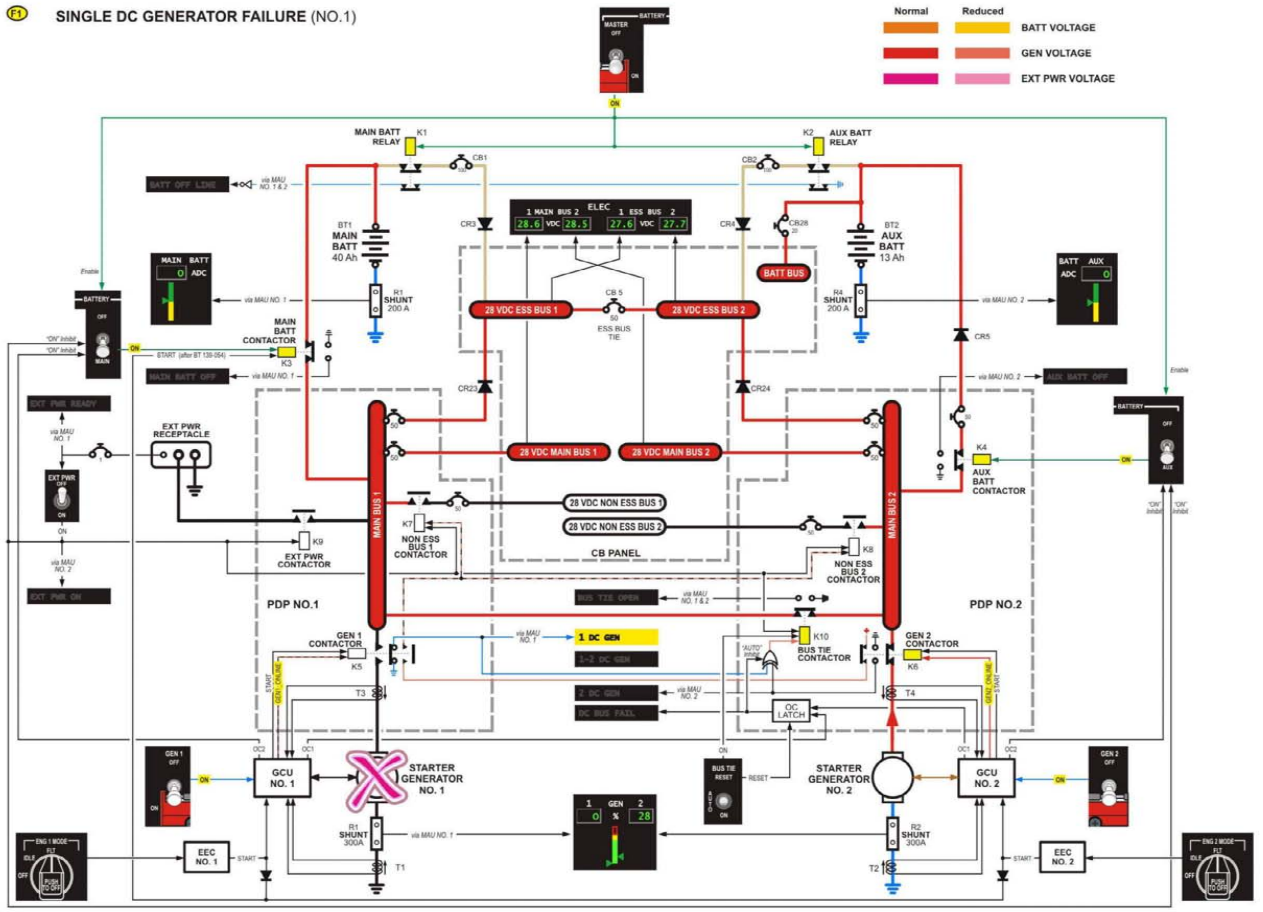
Pilot must do nothing.

F4, F5 – DUAL GENERATOR FAILURE (1 2 DC GEN)

When both Generators have failed only MAIN and AUX Batteries supply the aircraft loads. Manual shedding of the MAIN BUS 1 (BATTERY MAIN switch to OFF) is required in order to only supply the Essential (Emergency) loads.

F6 - ELECTRICAL BLACK OUT

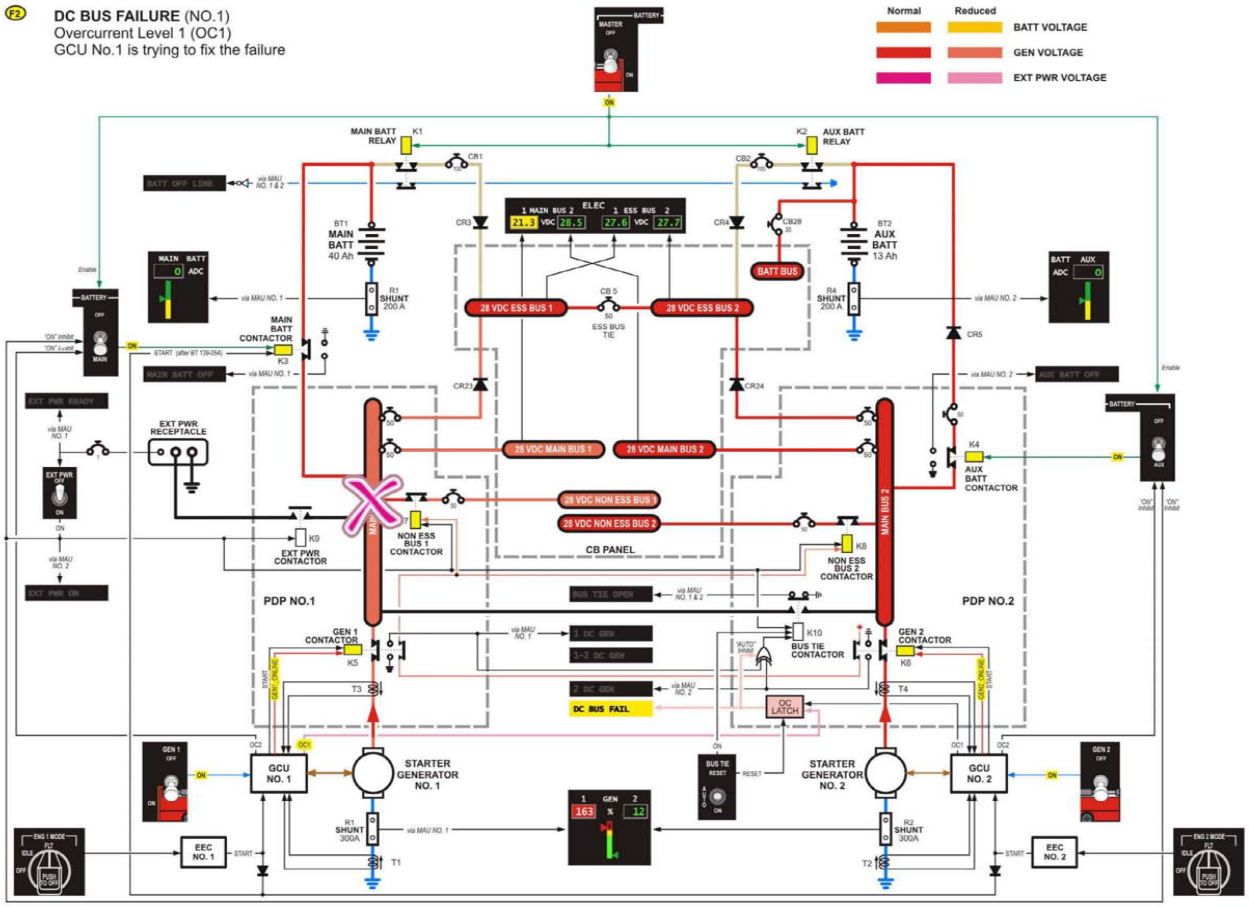
Following a dual generator failure, if batteries discharge completely (electrical black out) the engines continue to run in AUTO Mode: they are controlled by the EEC which is then electrically powered by the relevant Permanent Magnet Alternator (PMA). Since the landing gear is electrically actuated, it is necessary to extend it before losing electrical power completely; an increase in fuel consumption is then to be expected.



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F1 – SINGLE DC GENERATOR FAILURE (NO.1)

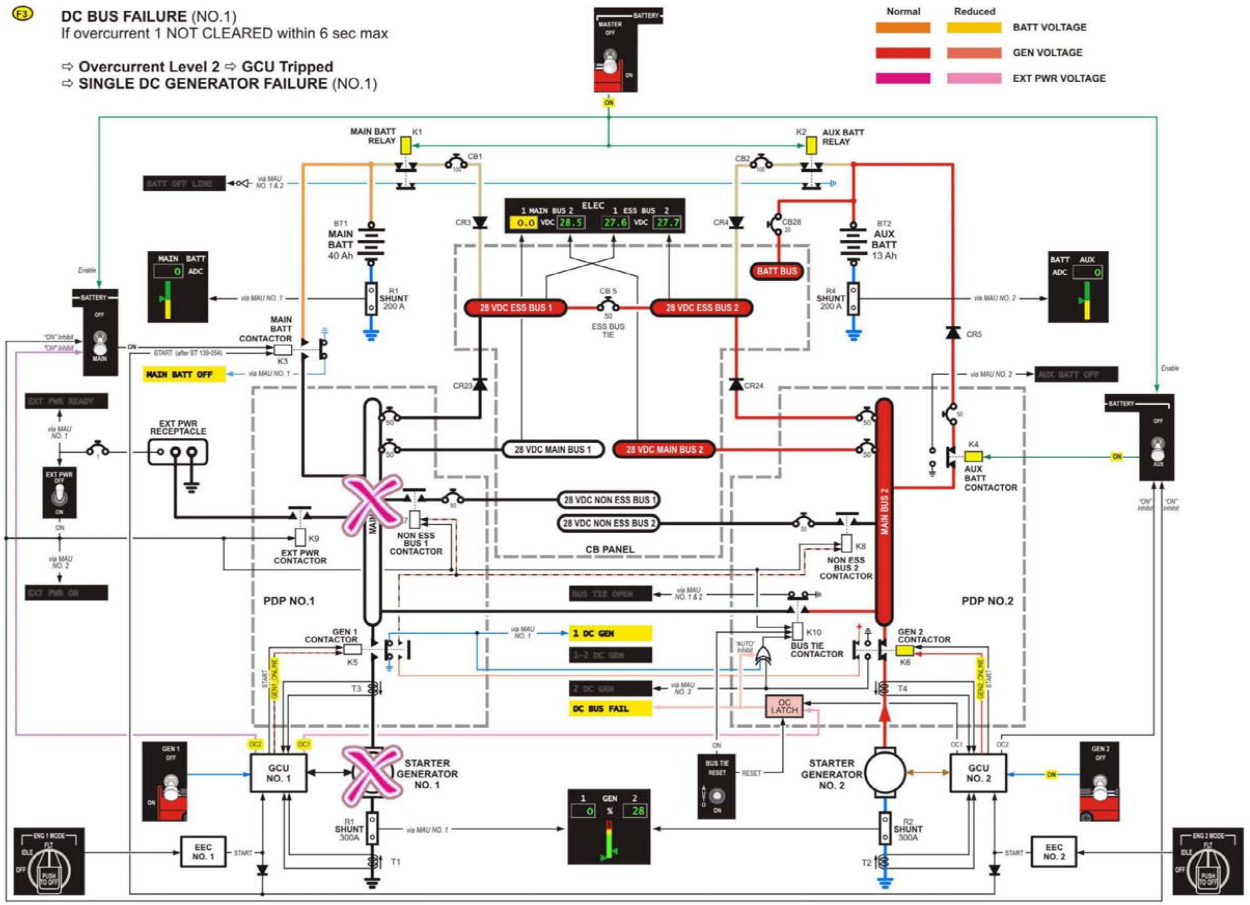
(Printed A3 format at the end of data module)



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F2 – DC BUS FAILURE (NO.1) – OVERCURRENT LEVEL 1

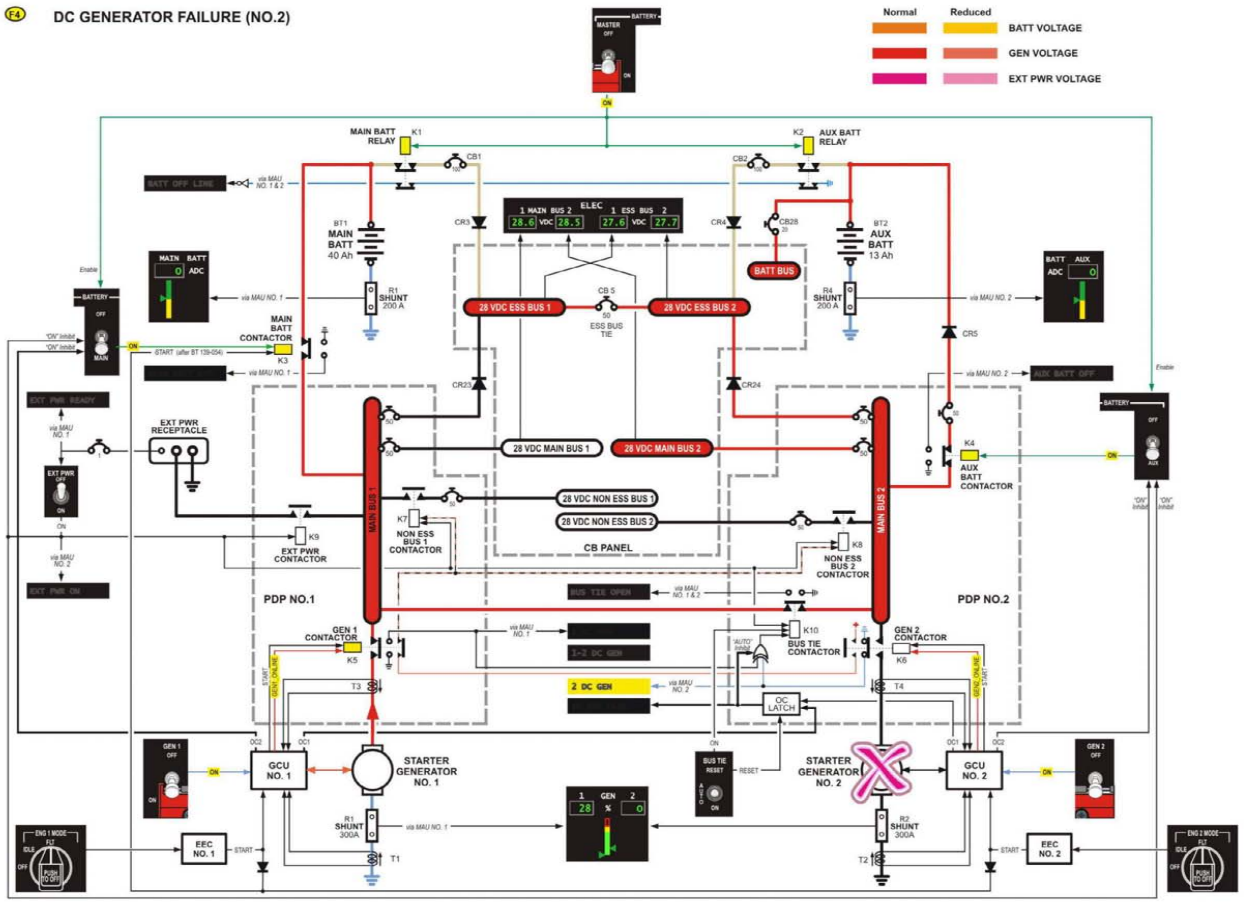
(Printed A3 format at the end of data module)



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F3 – DC BUS FAILURE (NO.1) – IF OVERCURRENT NOT CLEARED WITHIN 6 SEC MAX

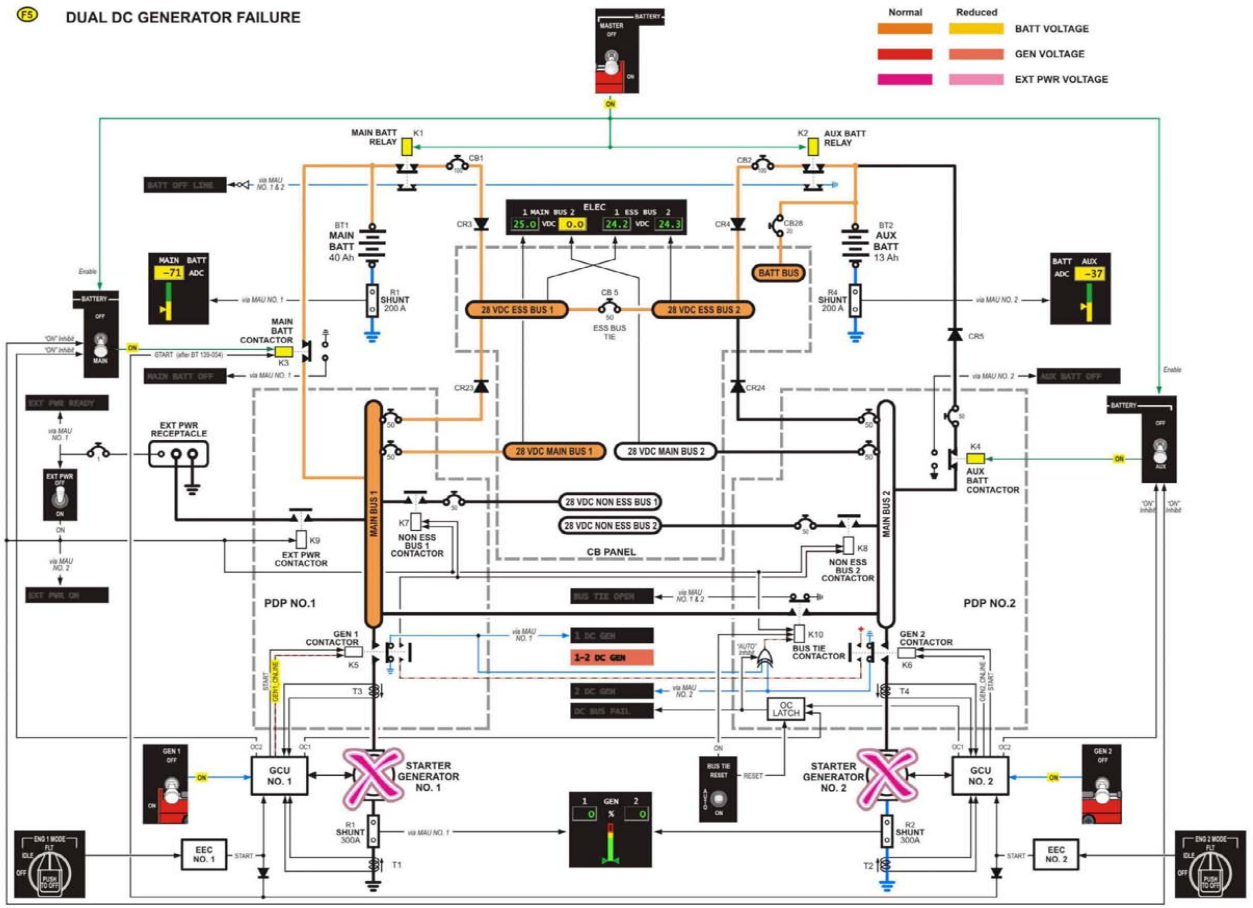
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F4 – DC GENERATOR FAILURE (NO.2)

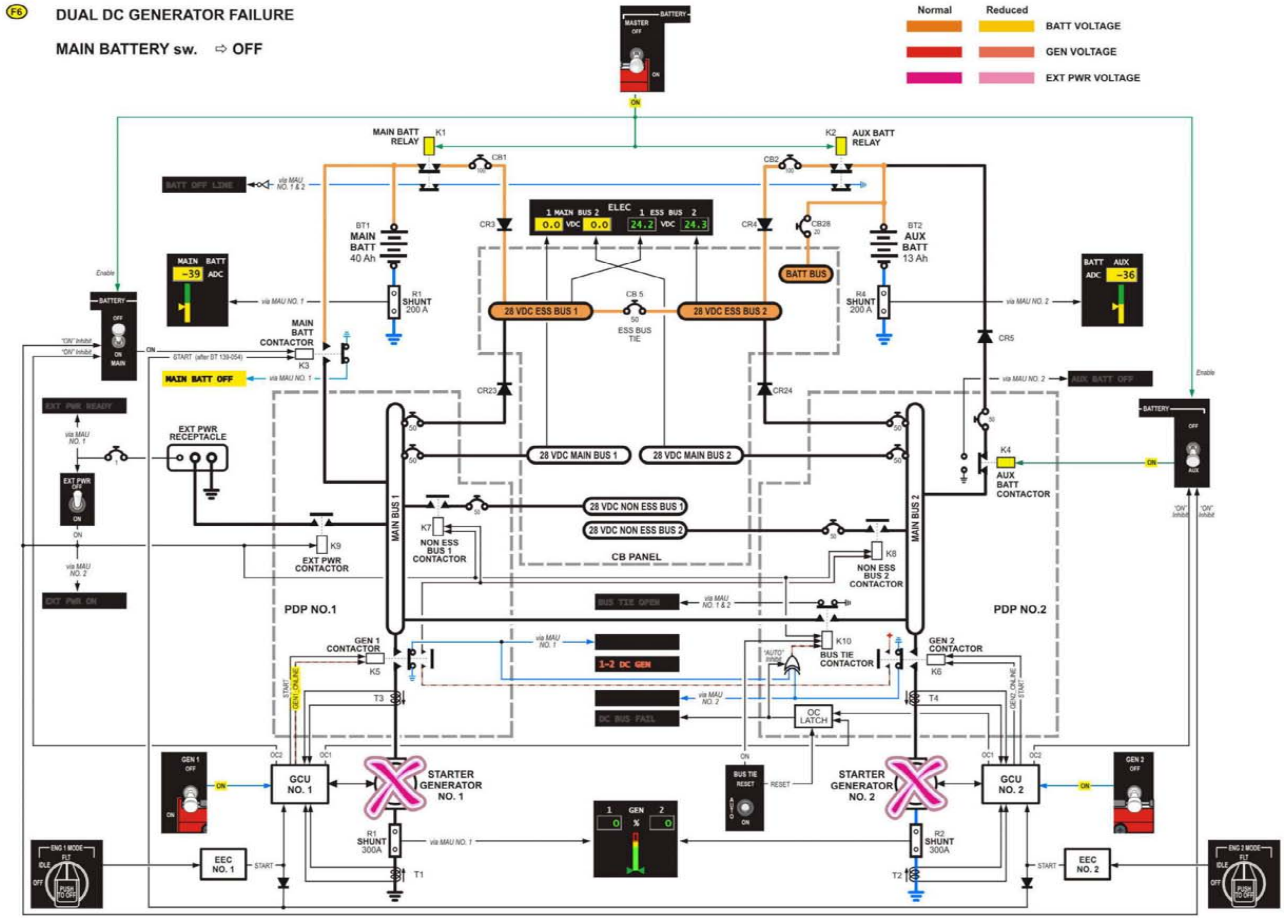
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F5 – DUAL DC GENERATOR FAILURE

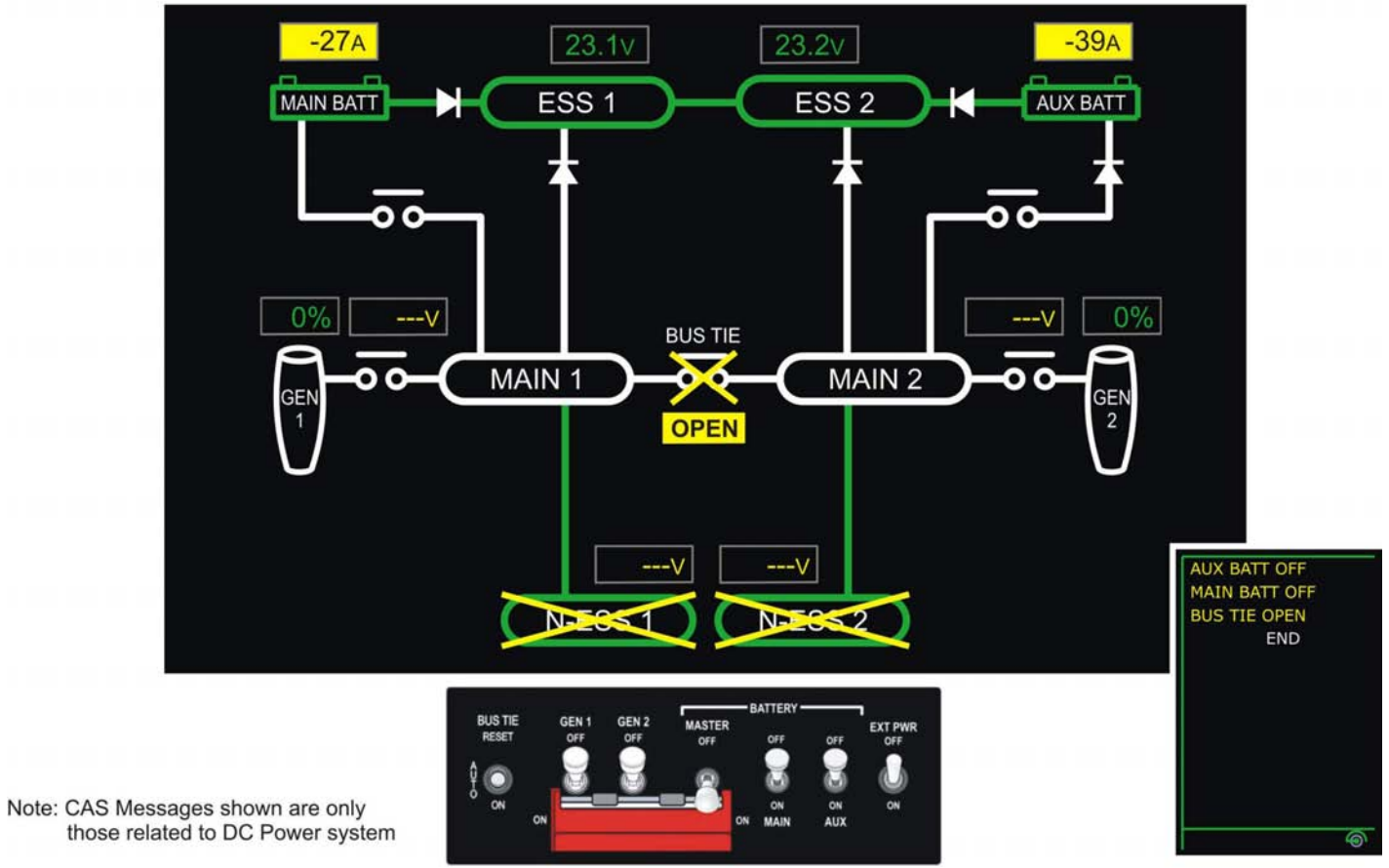
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F6 – DUAL DC GENERATOR FAILURE – MAIN BATTERY SWITCH OFF

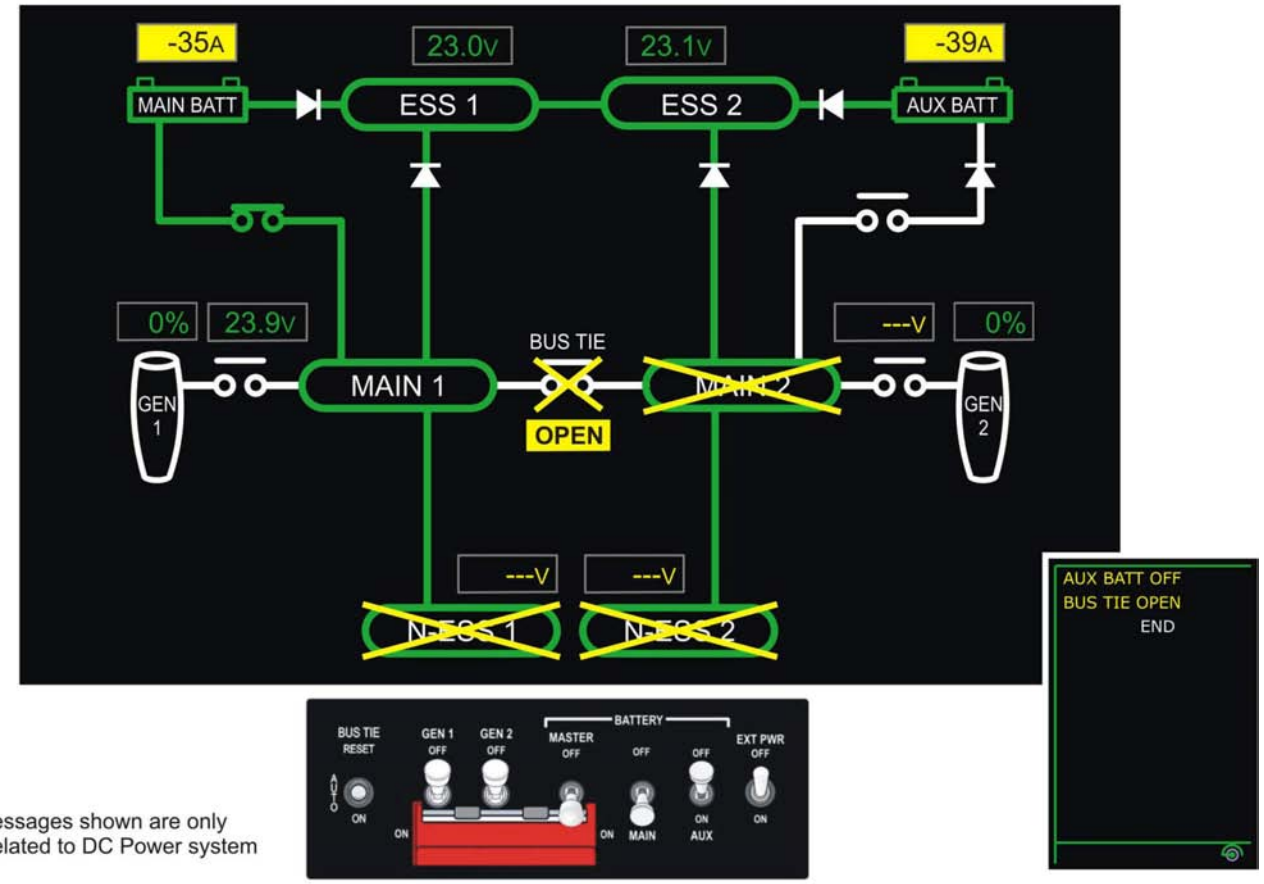
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Note: CAS Messages shown are only those related to DC Power system

ICN-39-A-243000-G-A0126-00357-A-01-1

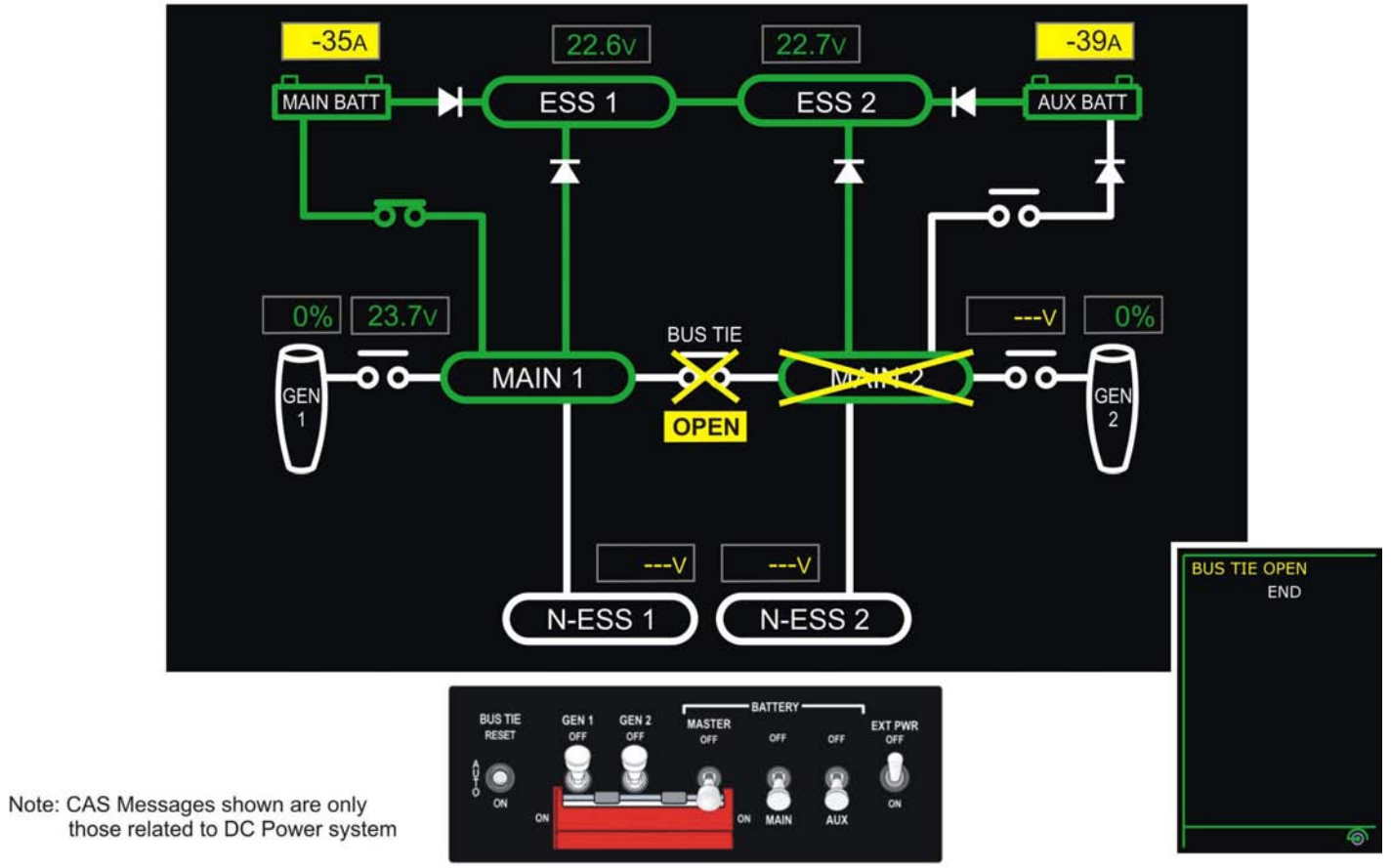
DC POWER SYNOPTIC – BATT MASTER = ON



Note: CAS Messages shown are only those related to DC Power system

ICN-39-A-243000-G-A0126-00358-A-01-1

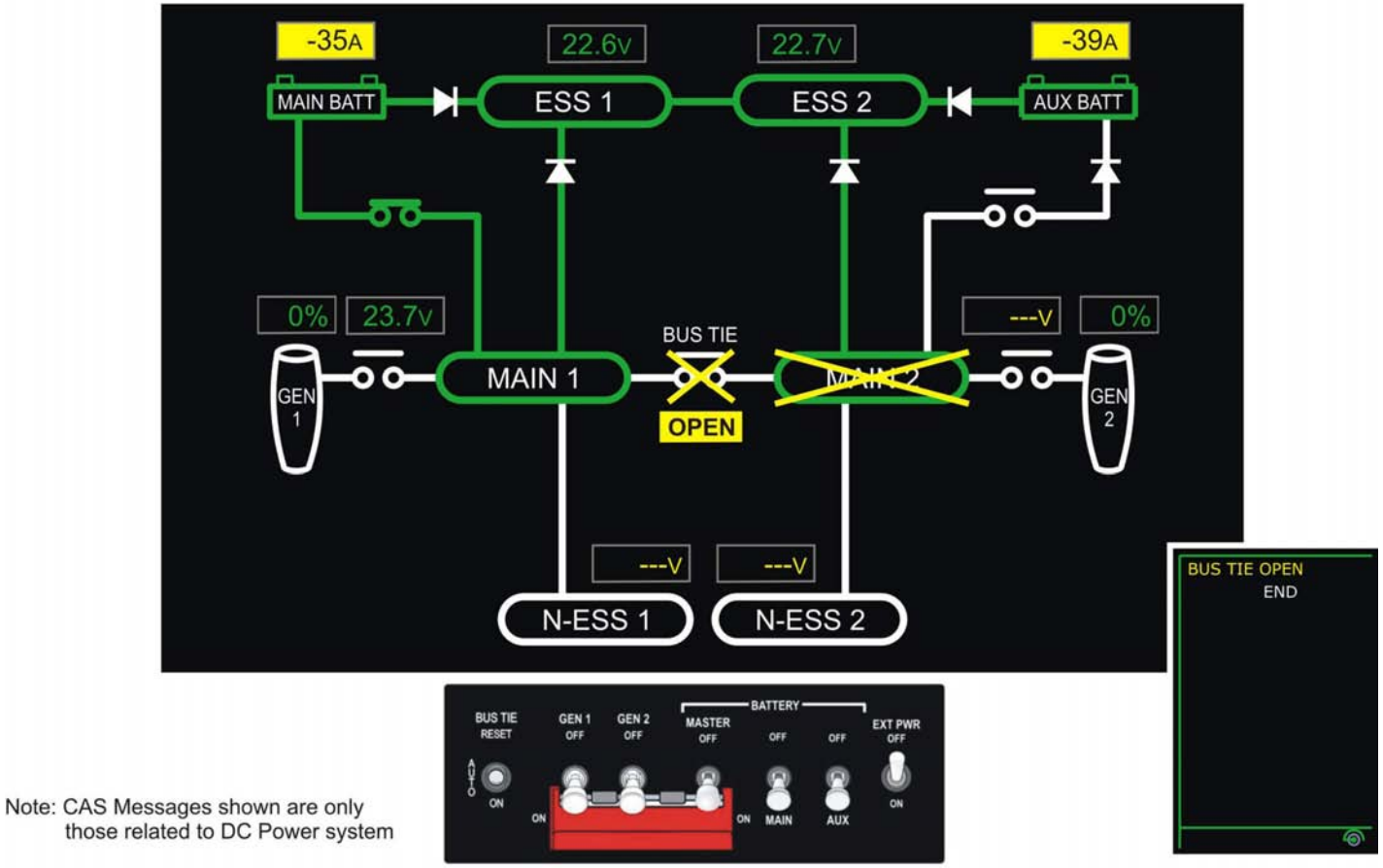
DC POWER SYNOPTIC – BATT MASTER + MAIN = ON



Note: CAS Messages shown are only those related to DC Power system

ICN-39-A-243000-G-A0126-00359-A-01-1

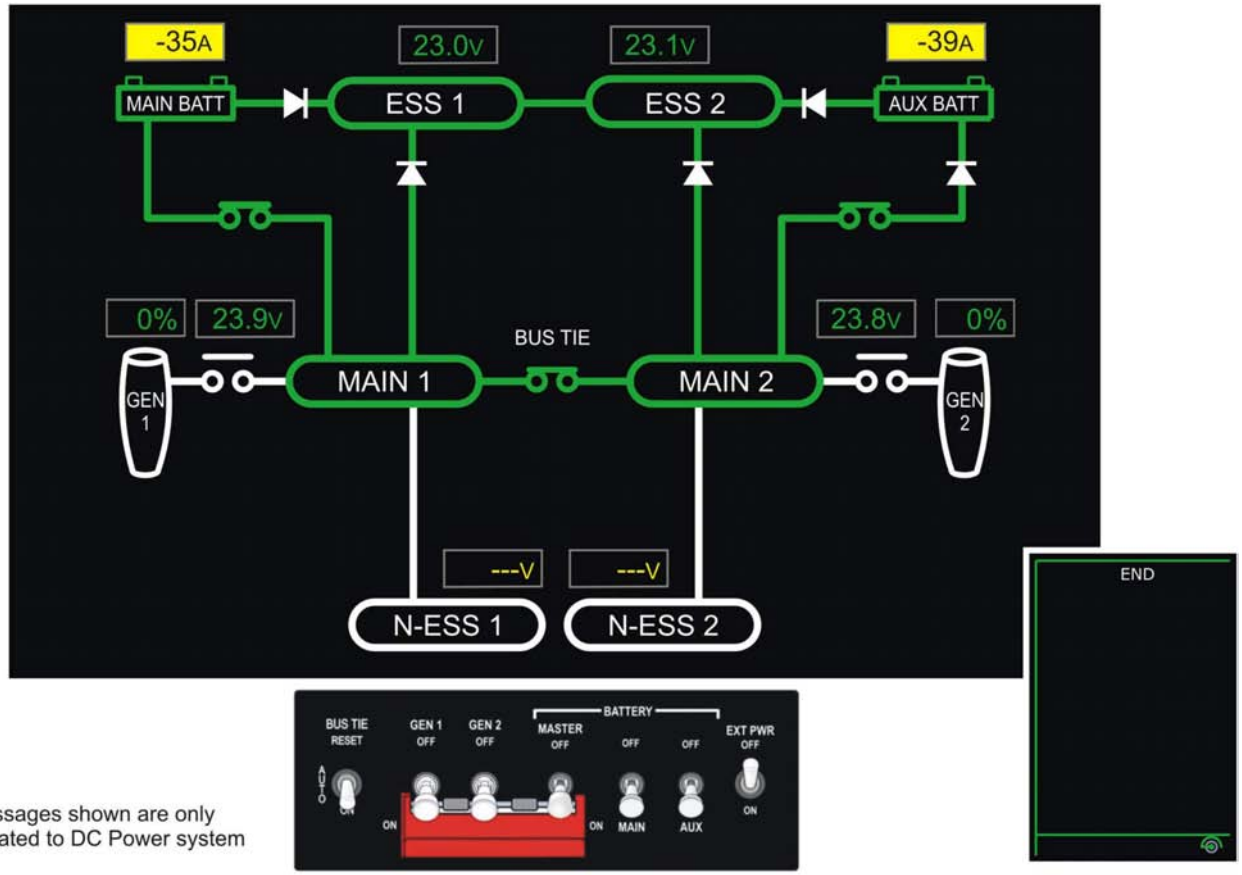
DC POWER SYNOPTIC – BATT MASTER + MAIN + AUX = ON



Note: CAS Messages shown are only those related to DC Power system

ICN-39-A-243000-G-A0126-00360-A-01-1

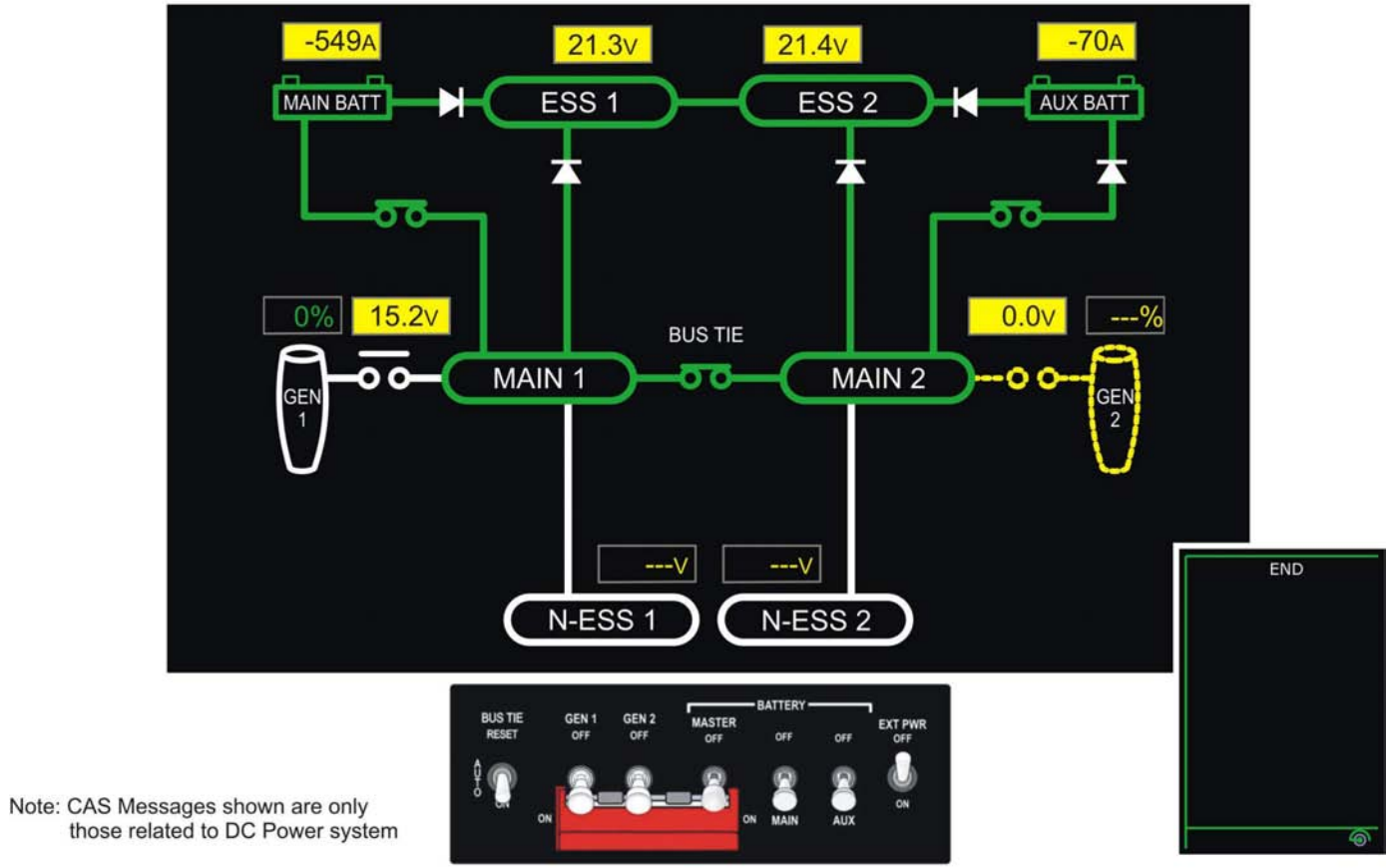
DC POWER SYNOPTIC – BATT + GEN = ON



Note: CAS Messages shown are only those related to DC Power system

ICN-39-A-243000-G-A0126-00361-A-01-1

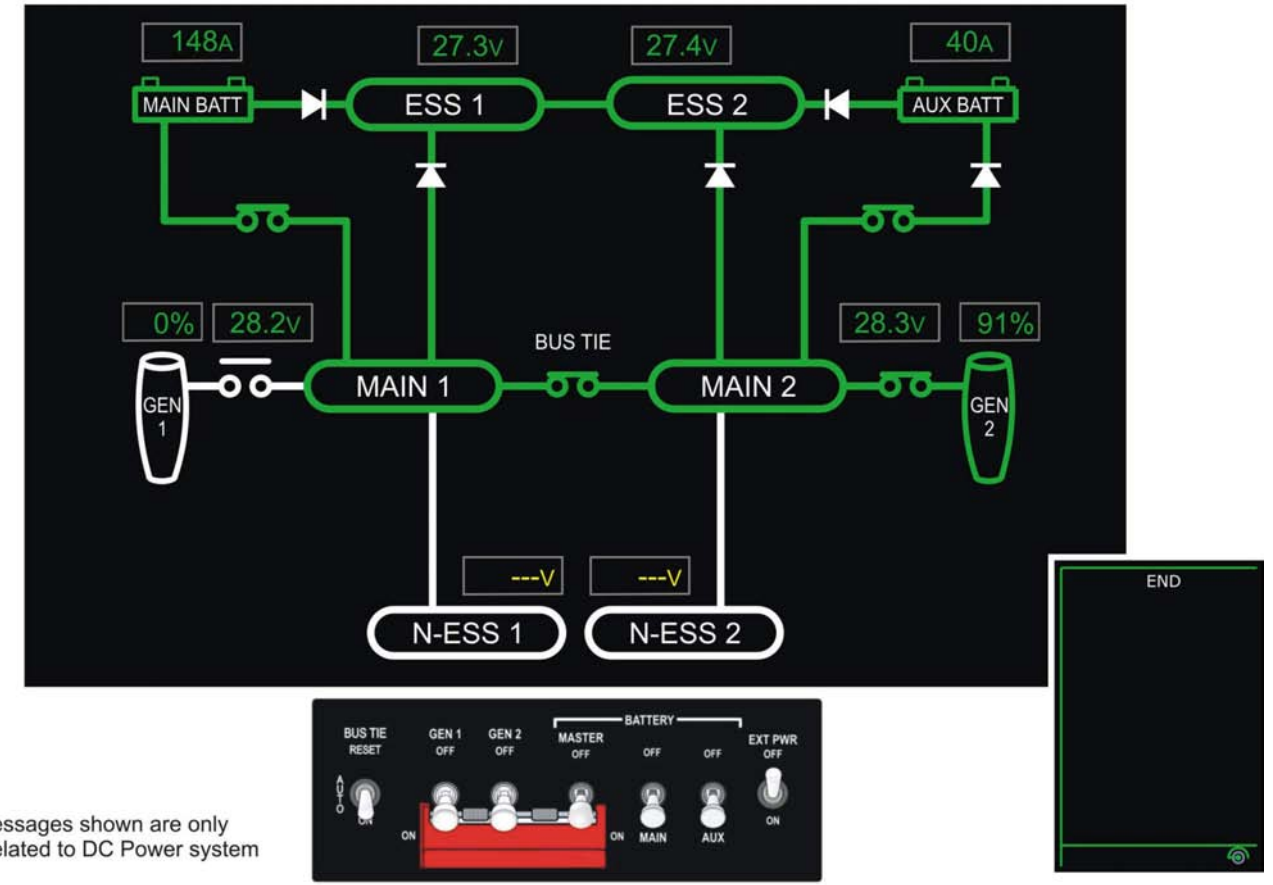
DC POWER SYNOPTIC – BATT + GEN + BUS TIE = ON



Note: CAS Messages shown are only those related to DC Power system

ICN-39-A-243000-G-A0126-00362-A-01-1

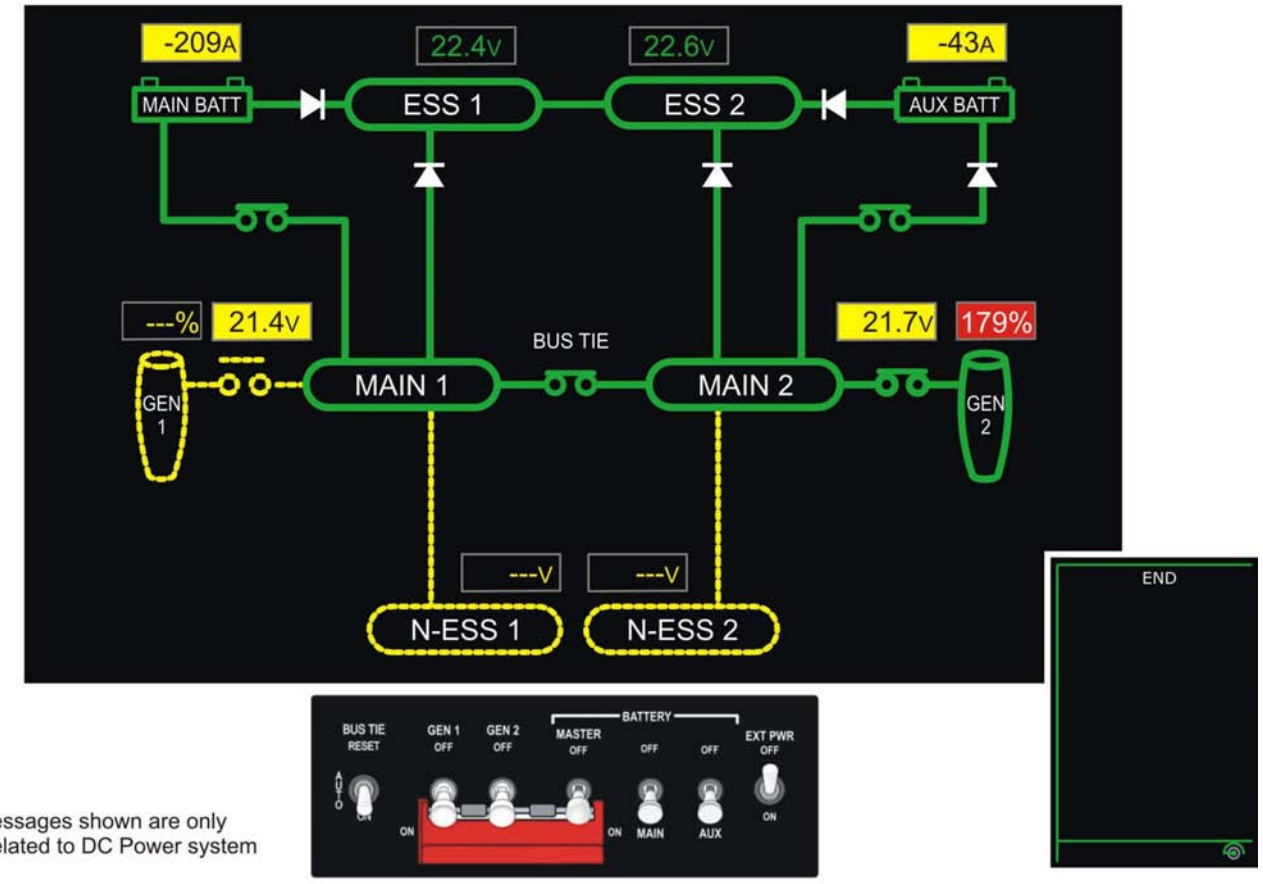
DC POWER SYNOPTIC – BATT STARTING (ENGINE NO.2)



Note: CAS Messages shown are only those related to DC Power system

ICN-39-A-243000-G-A0126-00363-A-01-1

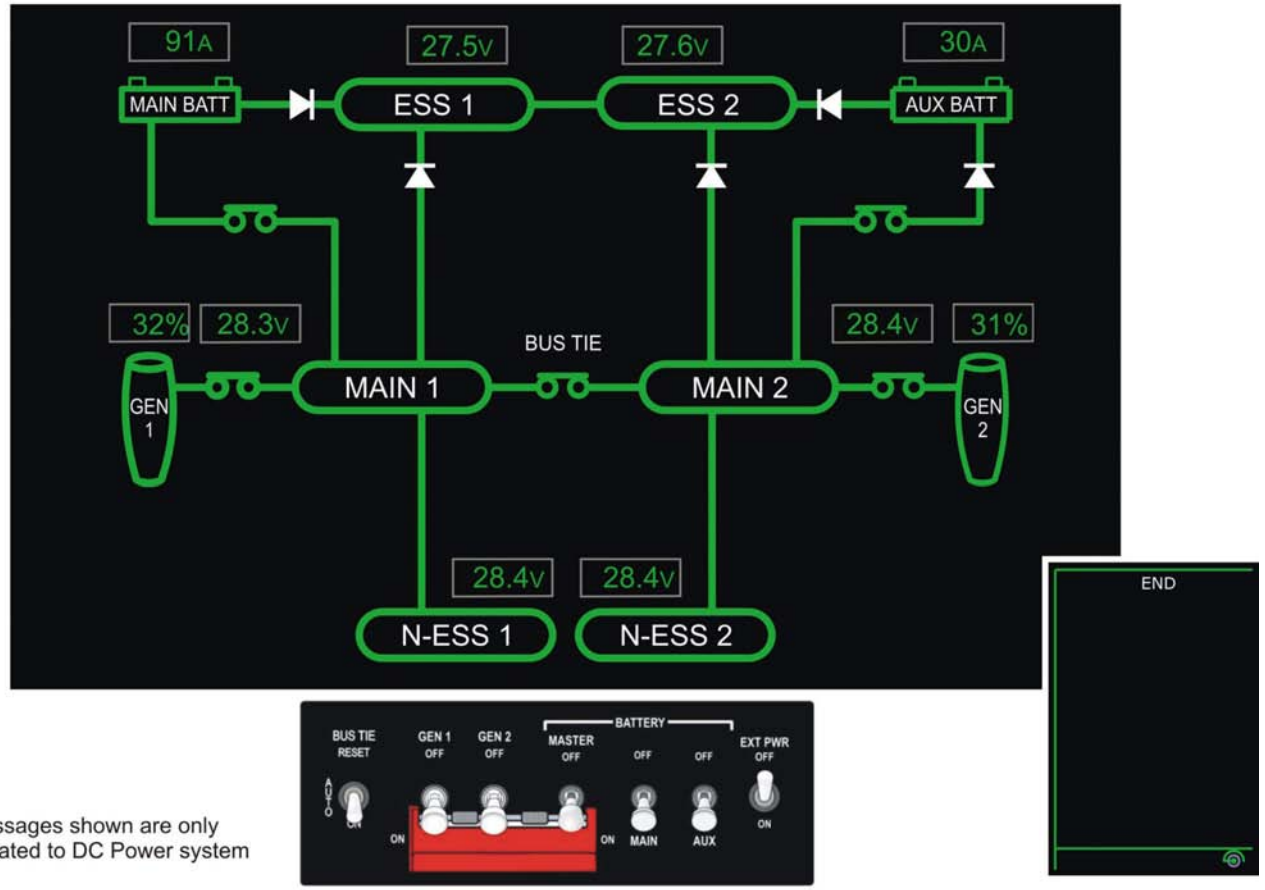
DC POWER SYNOPTIC – ENGINE NO.2 RUNNING



Note: CAS Messages shown are only those related to DC Power system

ICN-39-A-243000-G-A0126-00364-A-01-1

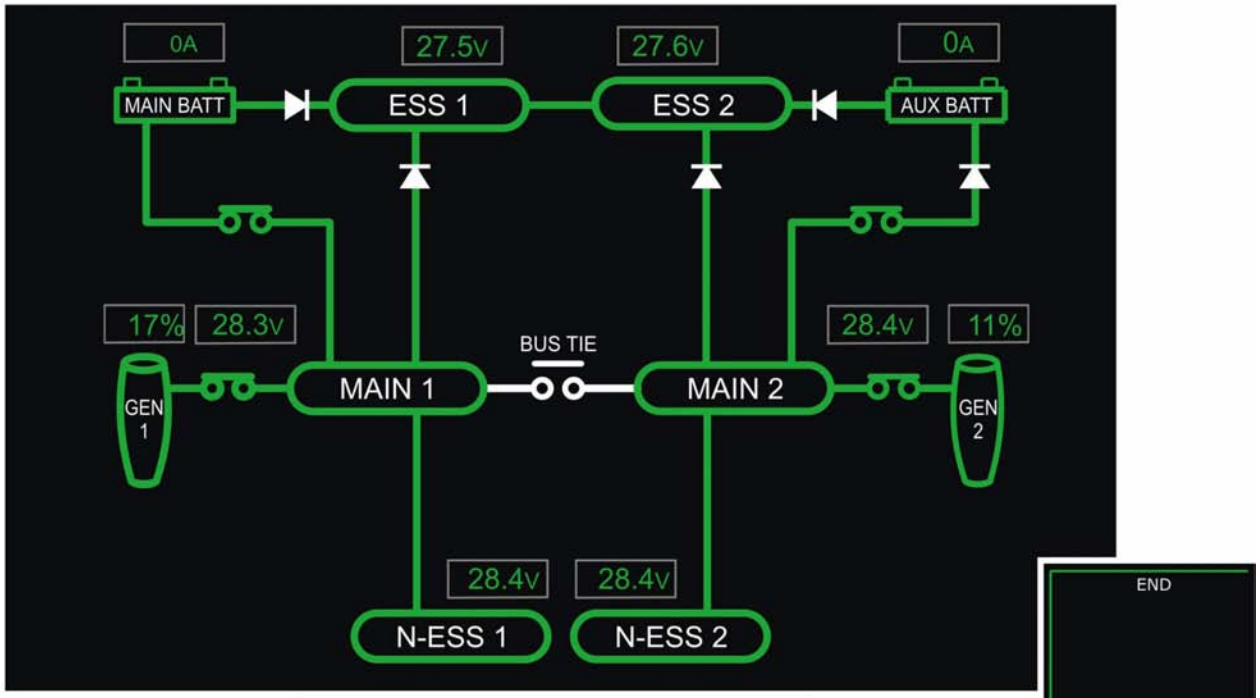
DC POWER SYNOPTIC – ENGINE NO.2 RUNNING



Note: CAS Messages shown are only those related to DC Power system

ICN-39-A-243000-G-A0126-00365-A-01-1

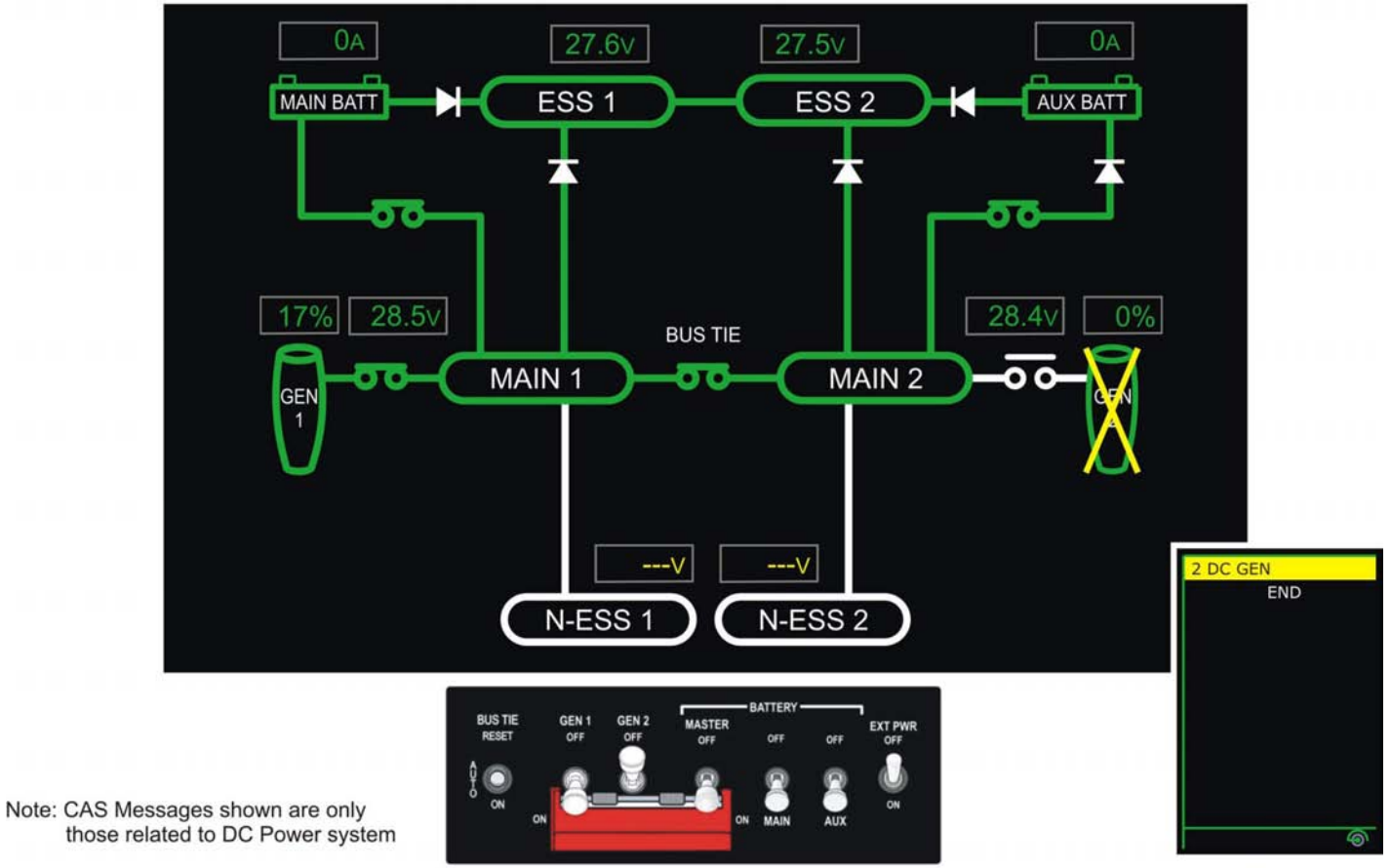
DC POWER SYNOPTIC PAGE – BOTH ENGINES RUNNING (BUS TIE = ON)



Note: CAS Messages shown are only those related to DC Power system

ICN-39-A-243000-G-A0126-00366-A-01-1

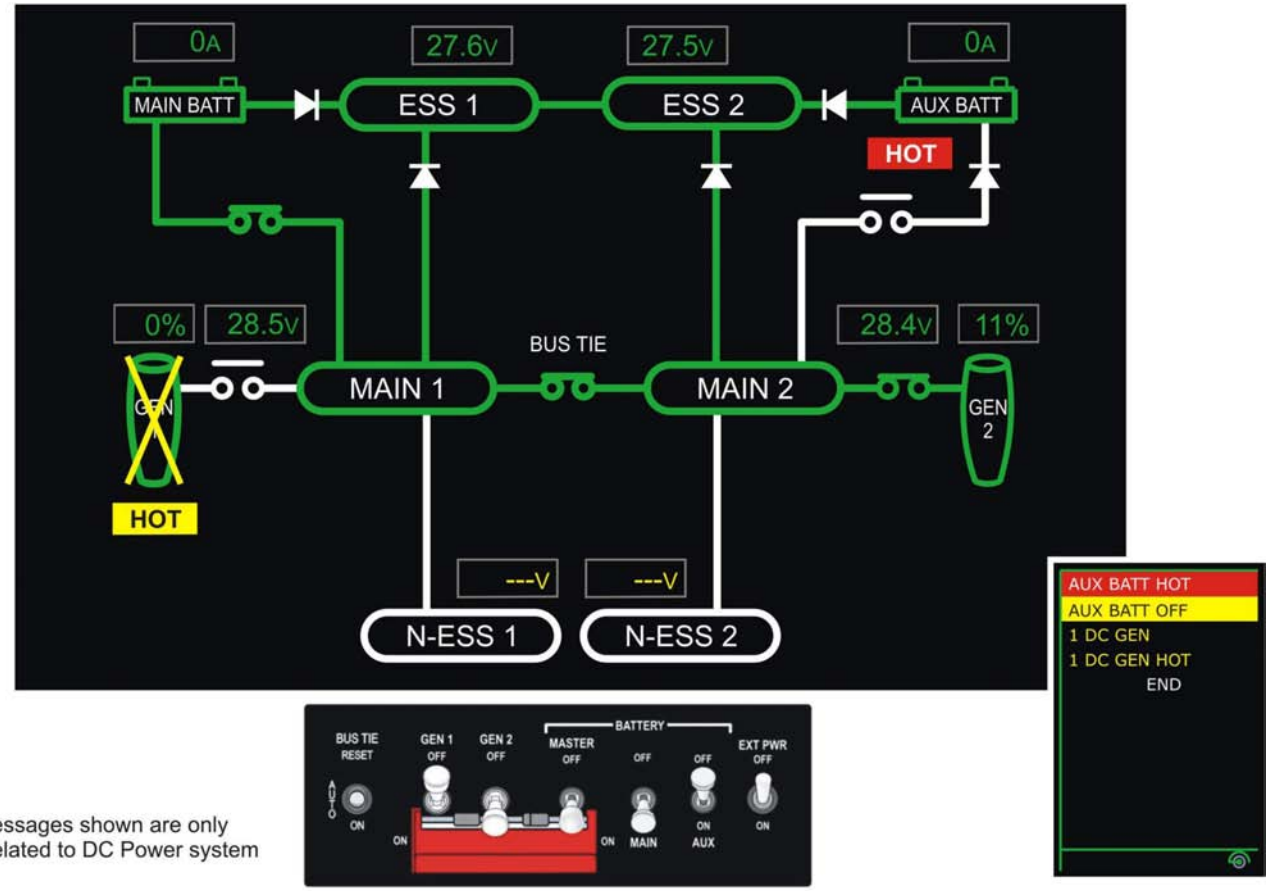
DC POWER SYNOPTIC – NORMAL IN-FLIGHT OPERATION



Note: CAS Messages shown are only those related to DC Power system

ICN-39-A-243000-G-A0126-00367-A-01-1

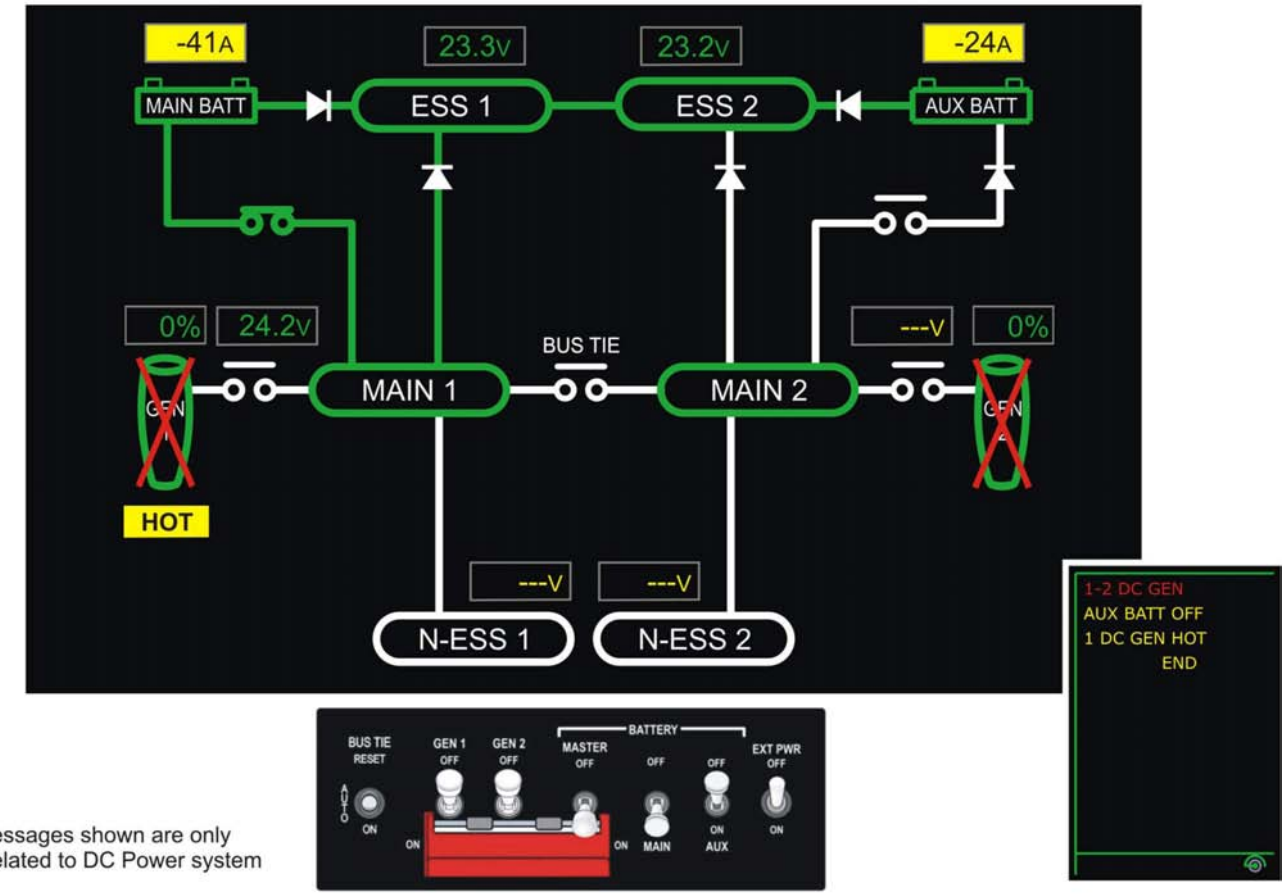
DC POWER SYNOPTIC – GEN 2 = FAIL



Note: CAS Messages shown are only those related to DC Power system

ICN-39-A-243000-G-A0126-00368-A-01-1

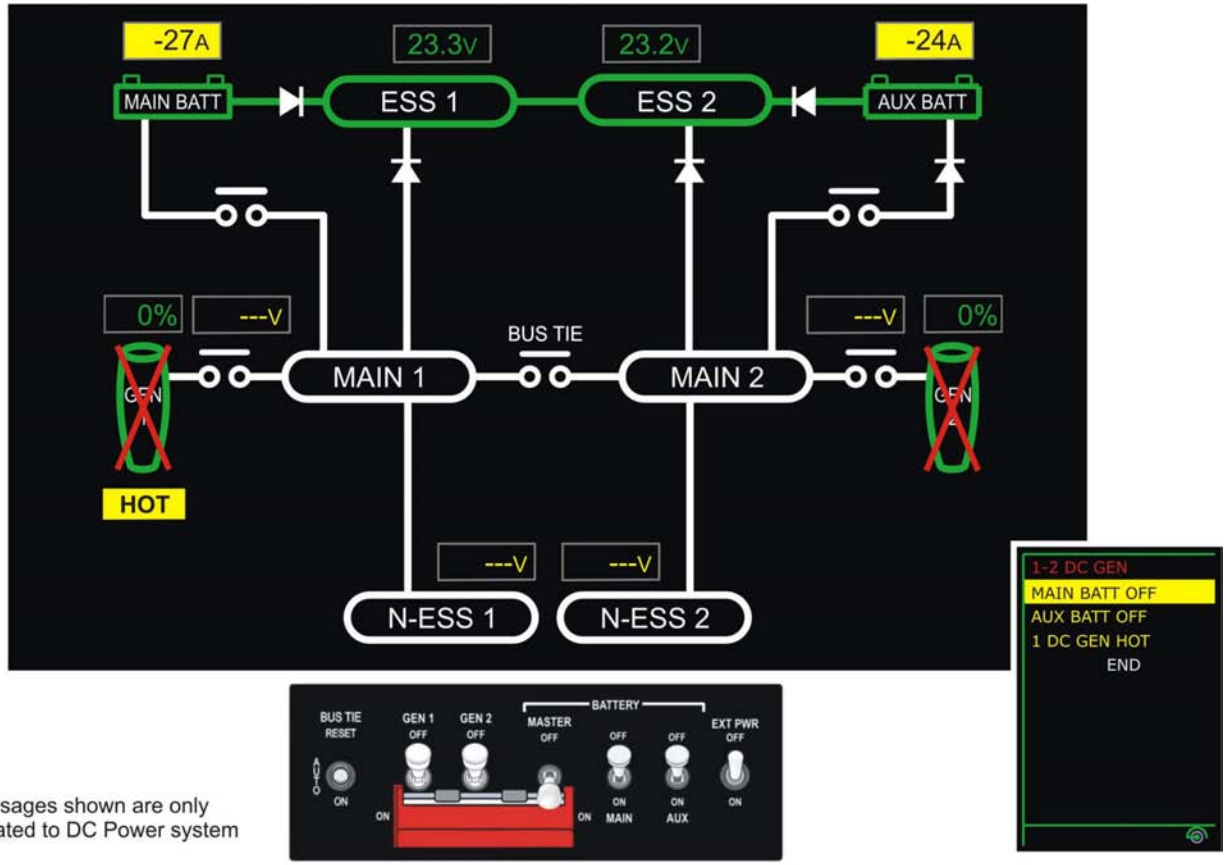
DC POWER SYNOPTIC – GEN 1 =HOT & OFF; AUX BATT = HOT & OFF



Note: CAS Messages shown are only those related to DC Power system

ICN-39-A-243000-G-A0126-00369-A-01-1

DC POWER SYNOPTIC – DUAL GEN FAIL

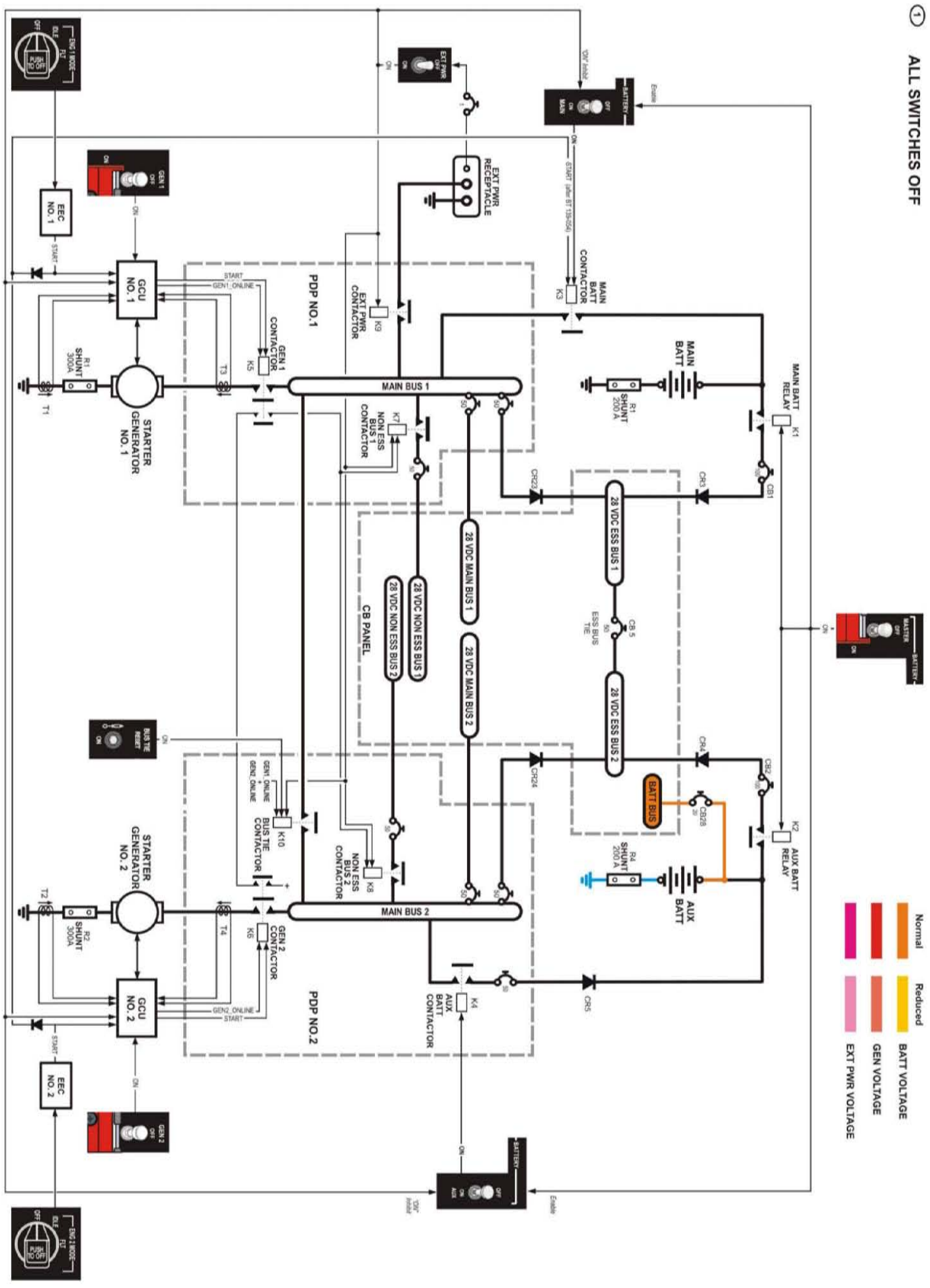


Note: CAS Messages shown are only those related to DC Power system

ICN-39-A-243000-G-A0126-00370-A-01-1

DC POWER SYNOPTIC – DUAL GEN FAIL, MAIN BATT = OFF

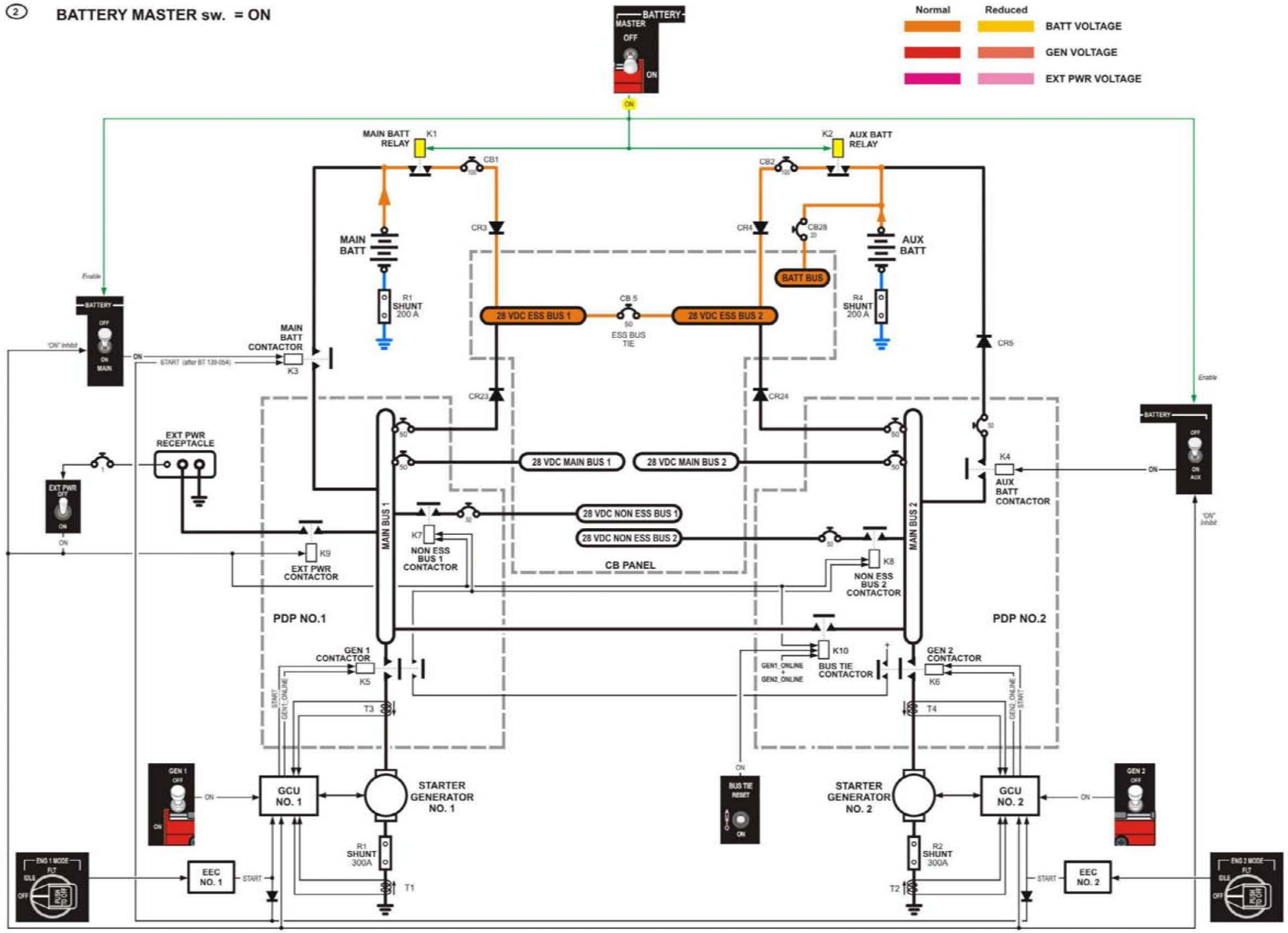
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STEP 1 – ALL SWITCHES OFF

ICN-39-A-243000-G-A0126-00341-A-01-1

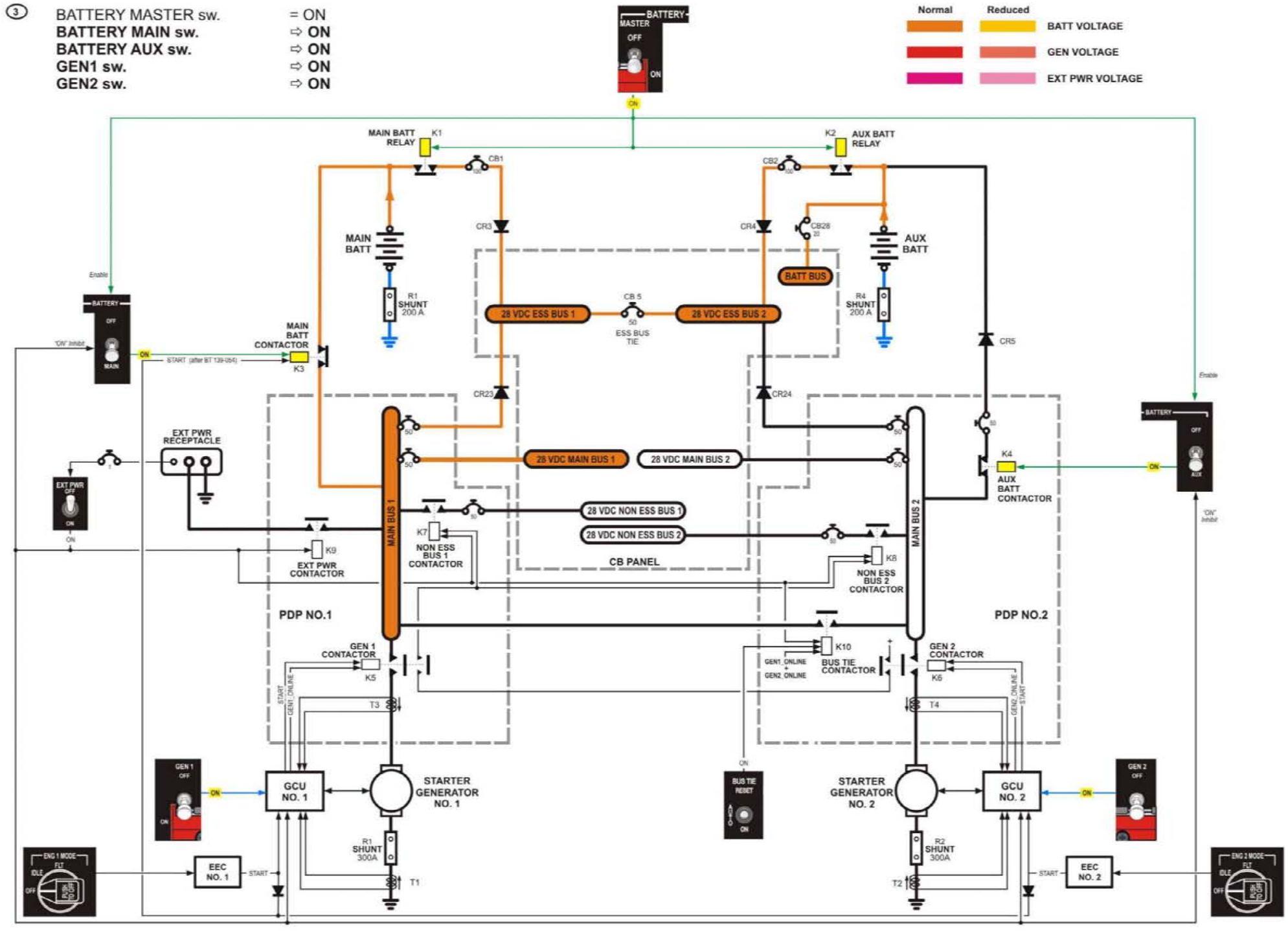
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ICN-39-A-243000-G-A0126-00342-A-01-1

STEP 2 – BATTERY MASTER SWITCH ON

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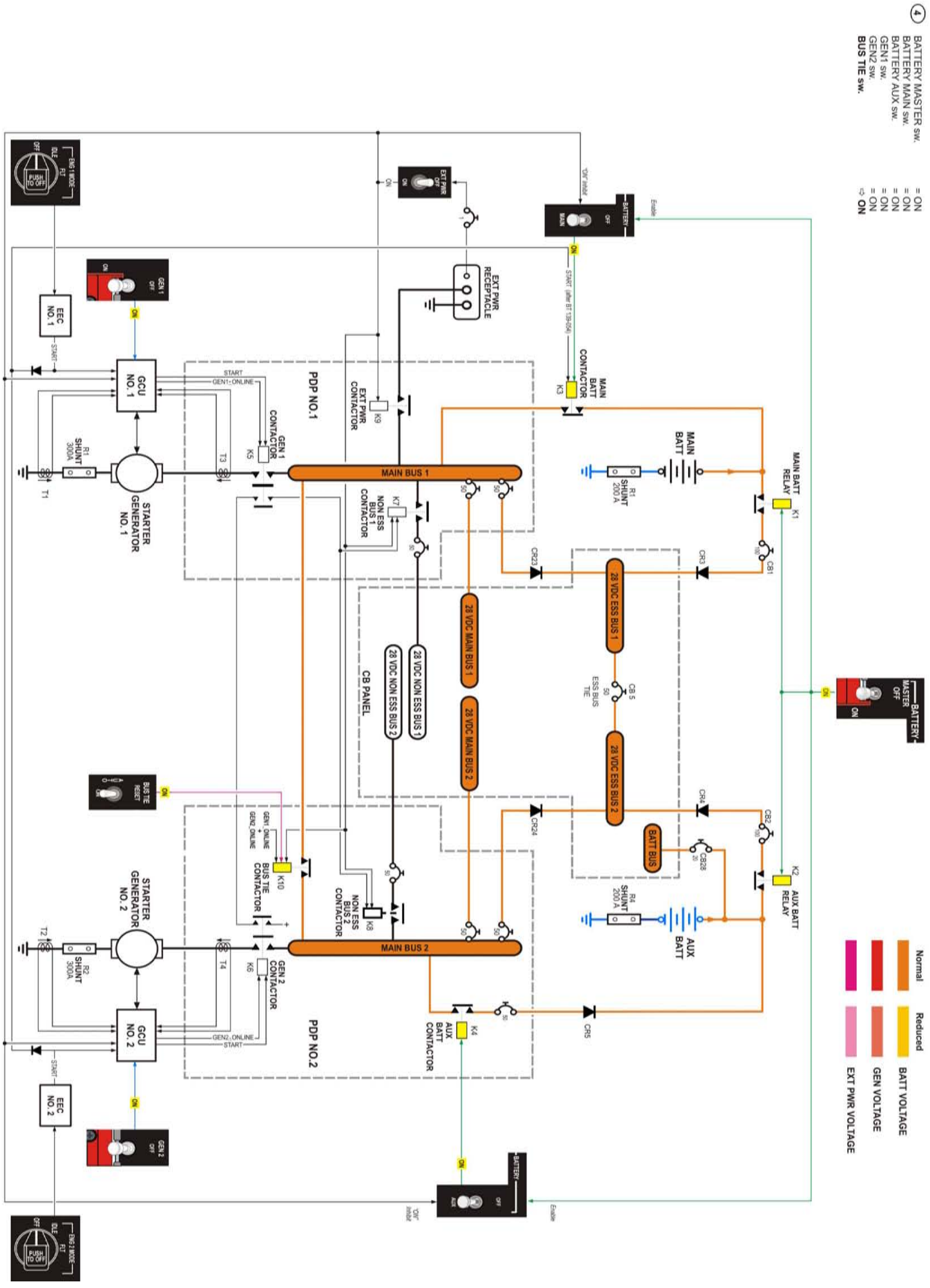


ICN-39-A-243000-G-A0126-00343-A-01-1

STEP 3 – MAIN AND AUX BATTERY SWITCHES ON

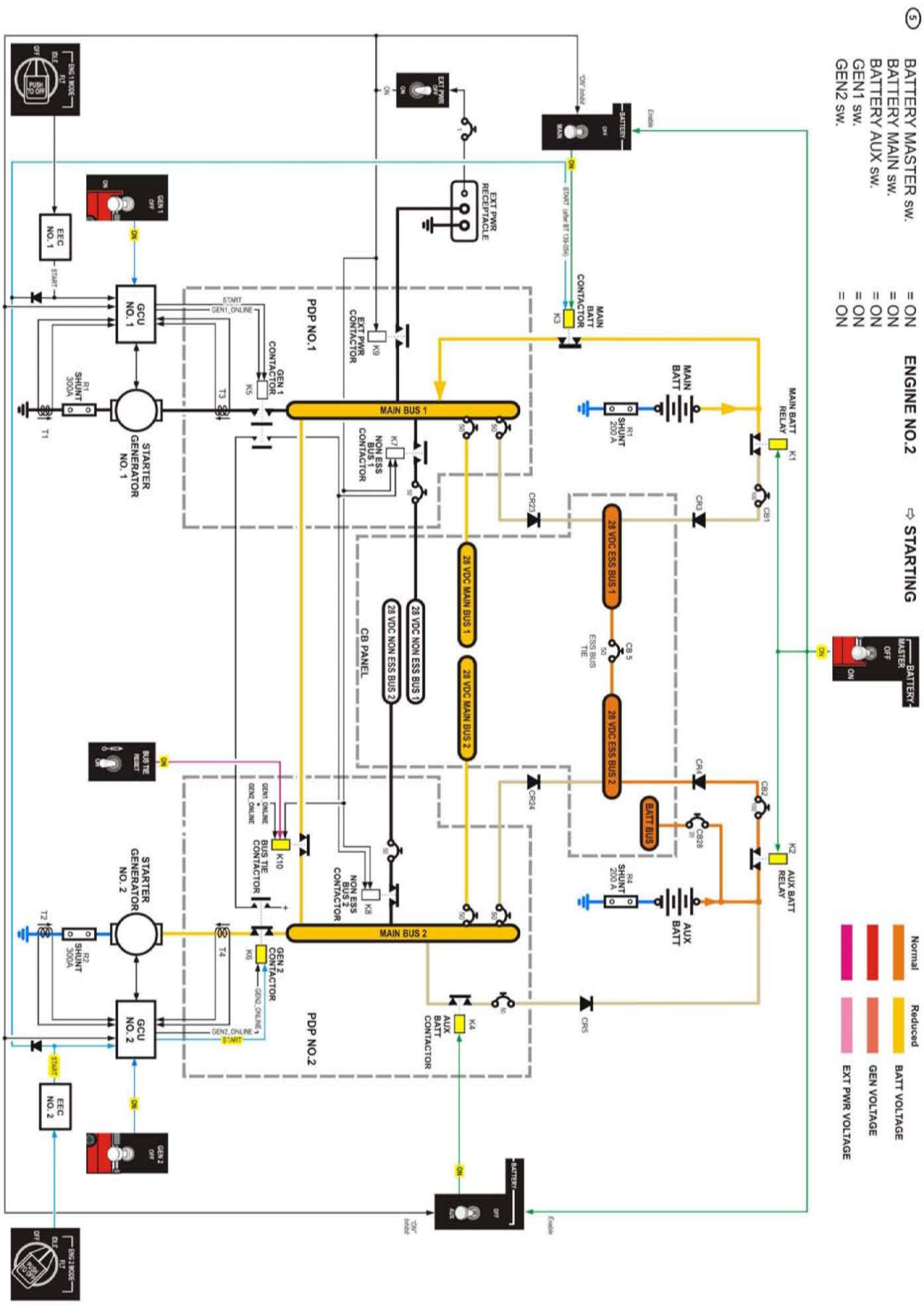
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STEP 4 – BUS TIE SWITCH ON



ICN-39-A-243000-G-A0126-00344-A-01-1

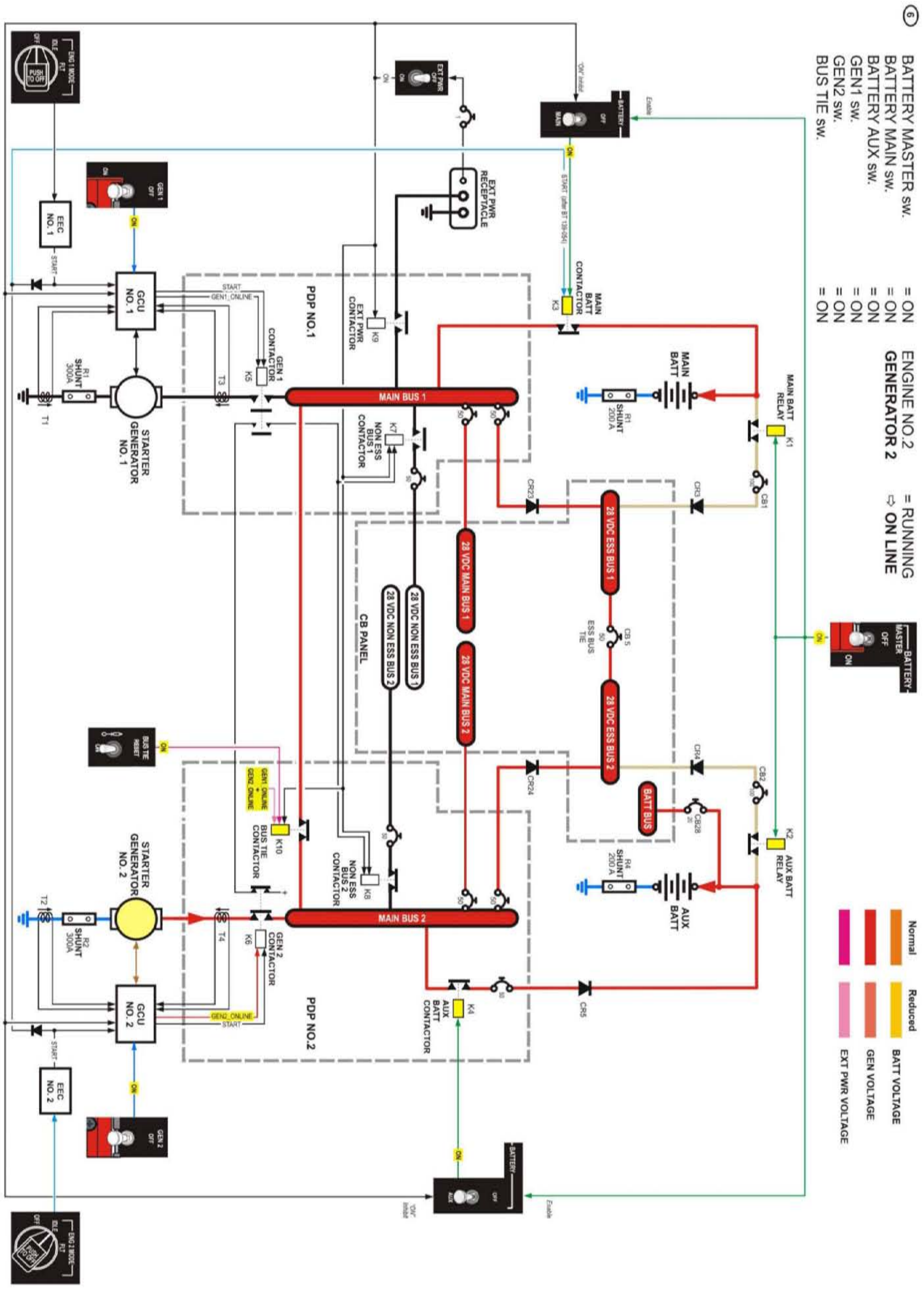
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STEP 5 – ENGINE NO.2 STARTING

ICN-39-A-243000-G-A0126-00345-A-01-1

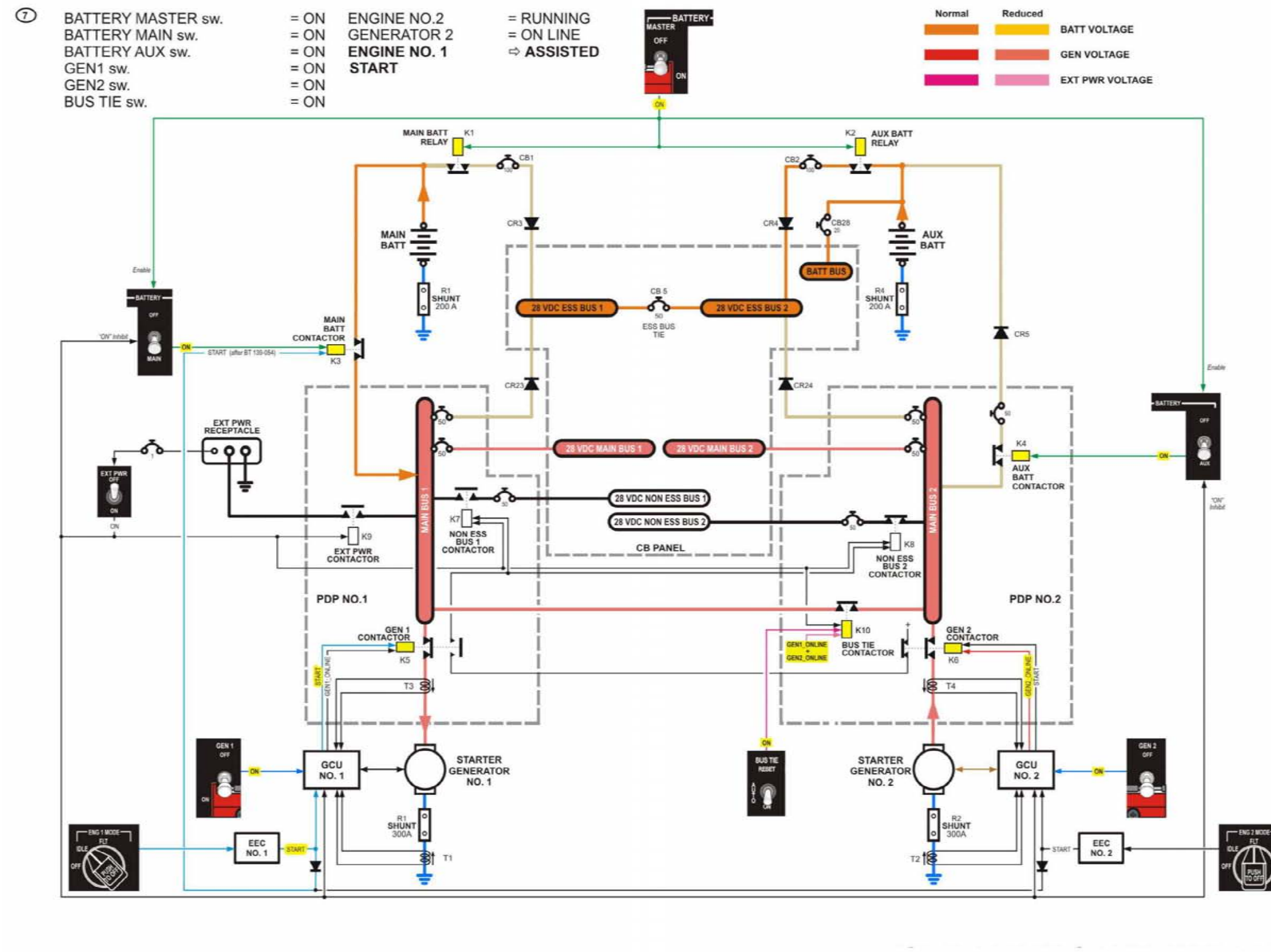
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STEP 6 – GEN 2 ON LINE

ICN-39-A-243000-G-A0126-00346-A-01-1

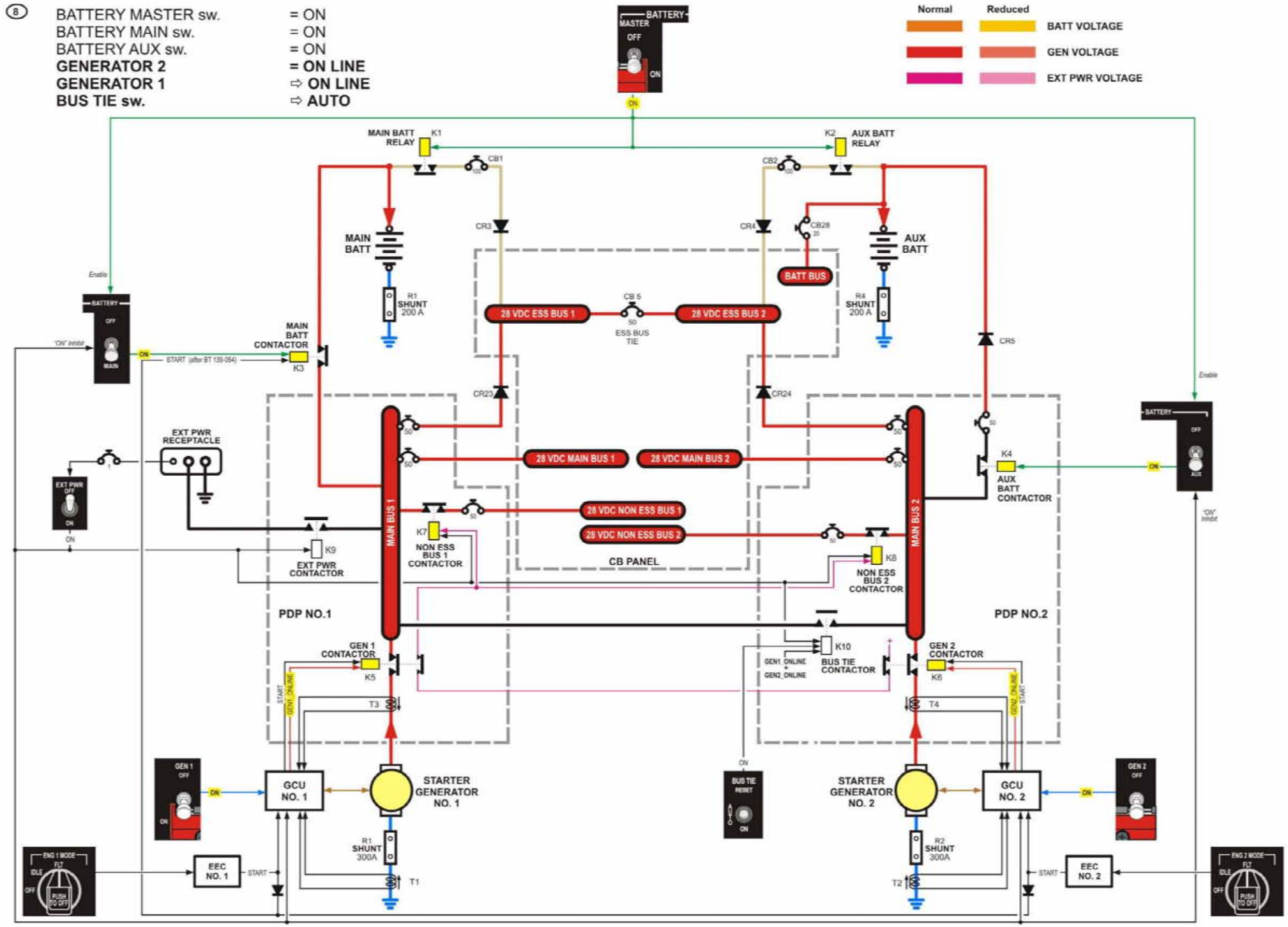
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STEP 7 – ENGINE NO.1 ASSISTED START

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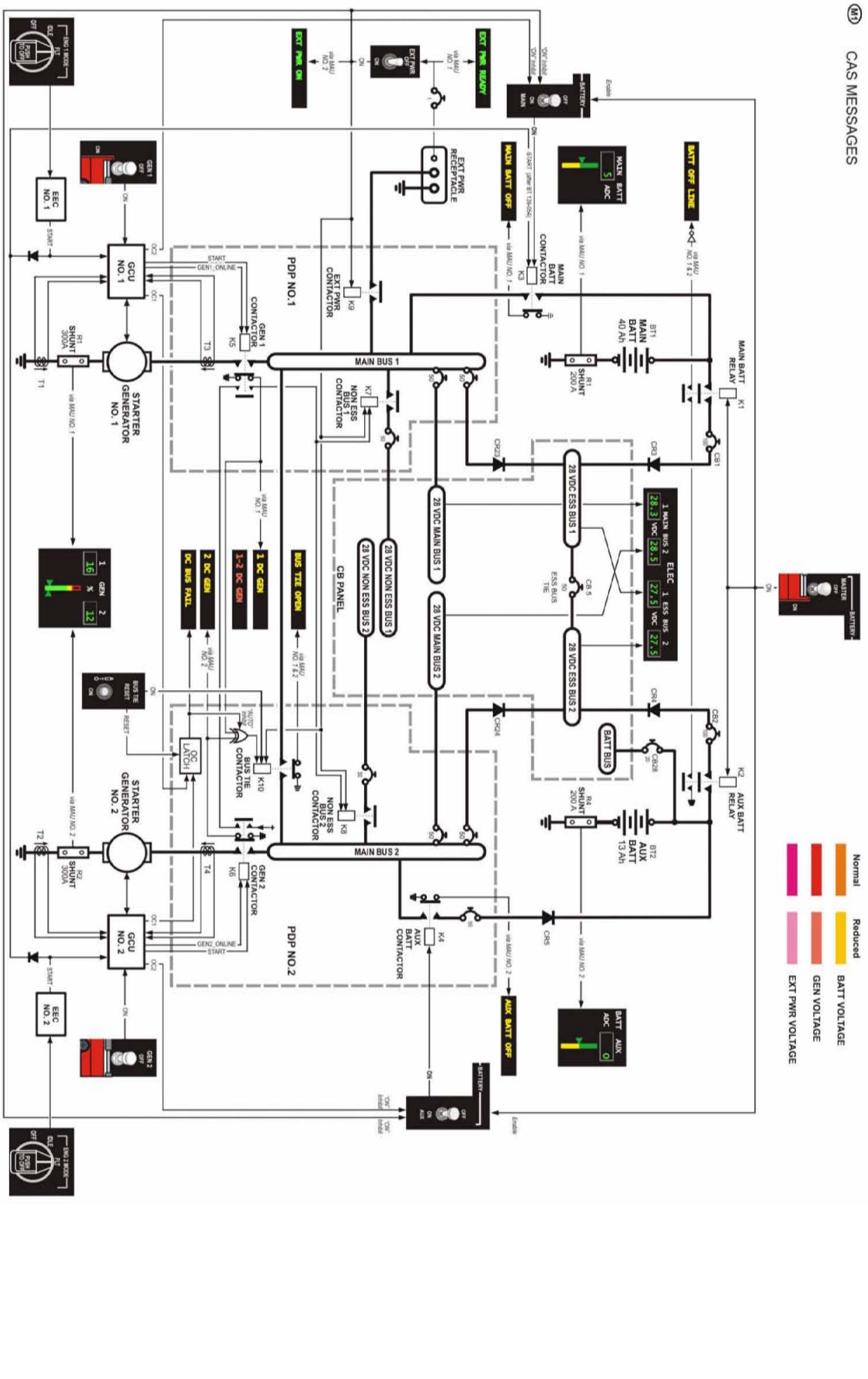


ICN-39-A-243000-G-A0126-00348-A-01-1

STEP 8 – GEN 1 AND GEN 2 ON-LINE AND BUS TIE SWITCH AUTO

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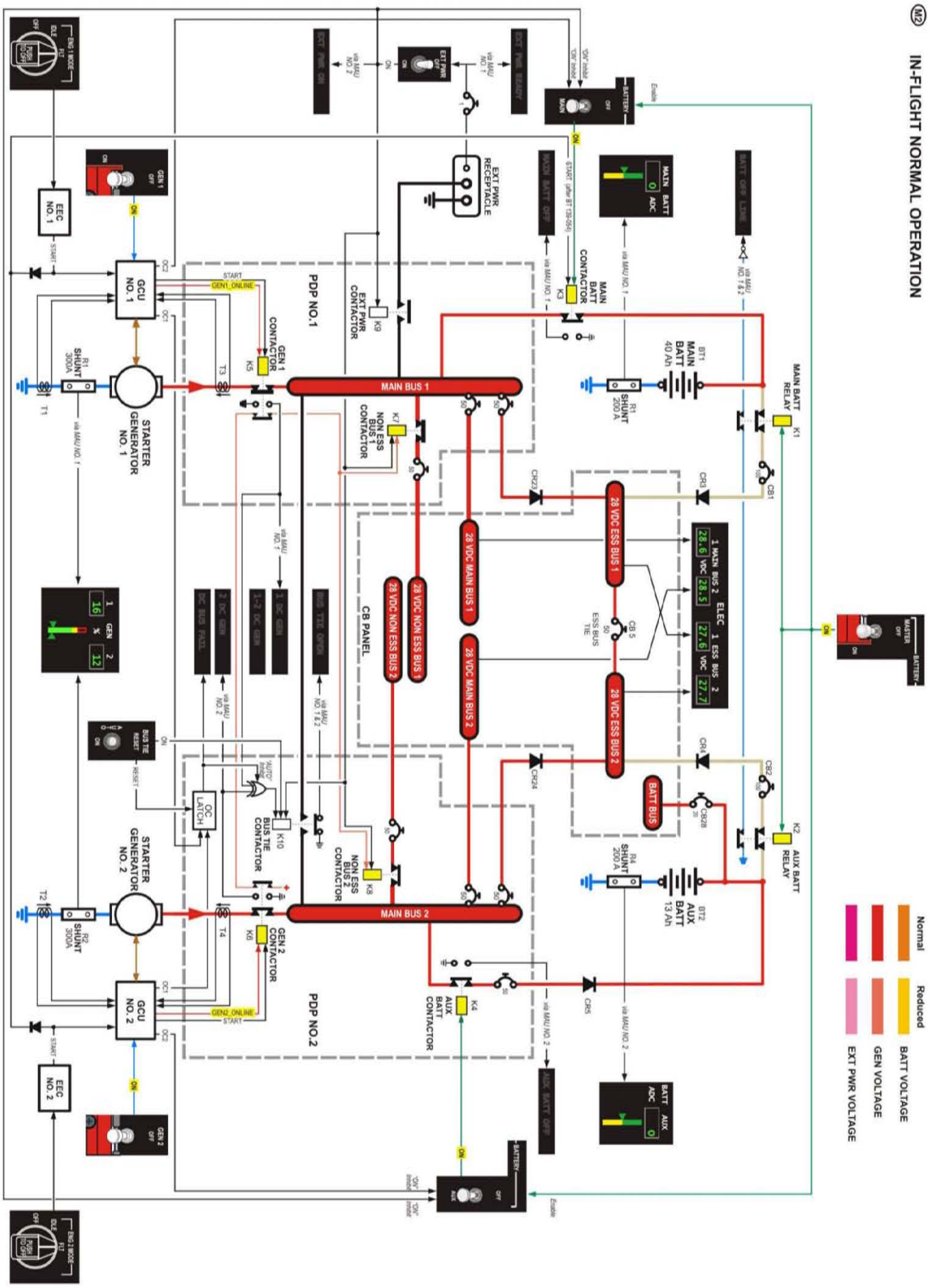
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M1 – CAS MESSAGES

ICN-39-A-243000-G-A0126-00349-A-01-1

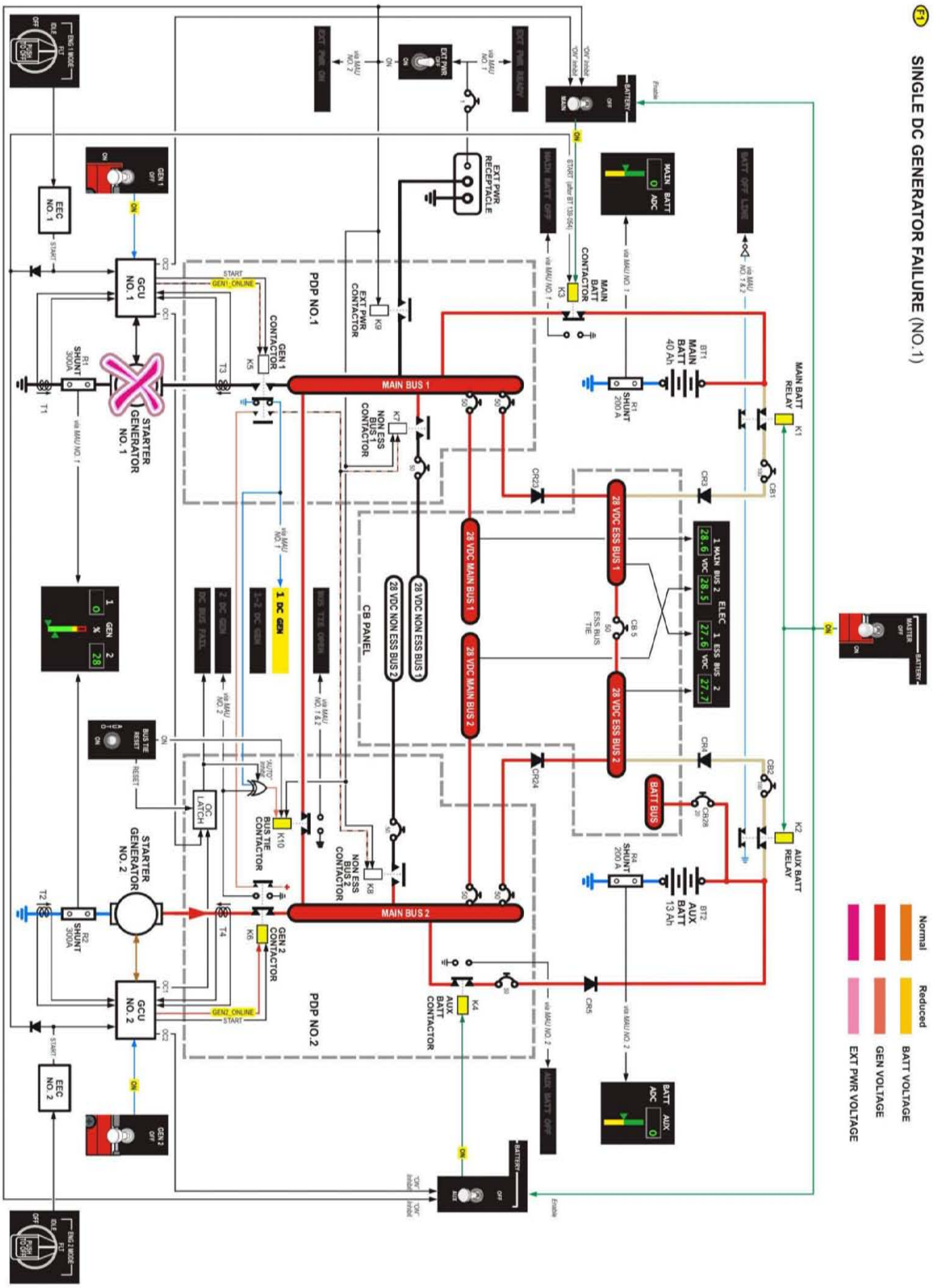
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M2 – IN FLIGHT NORMAL OPERATION

ICN-39-A-243000-G-A0126-00350-A-01-1

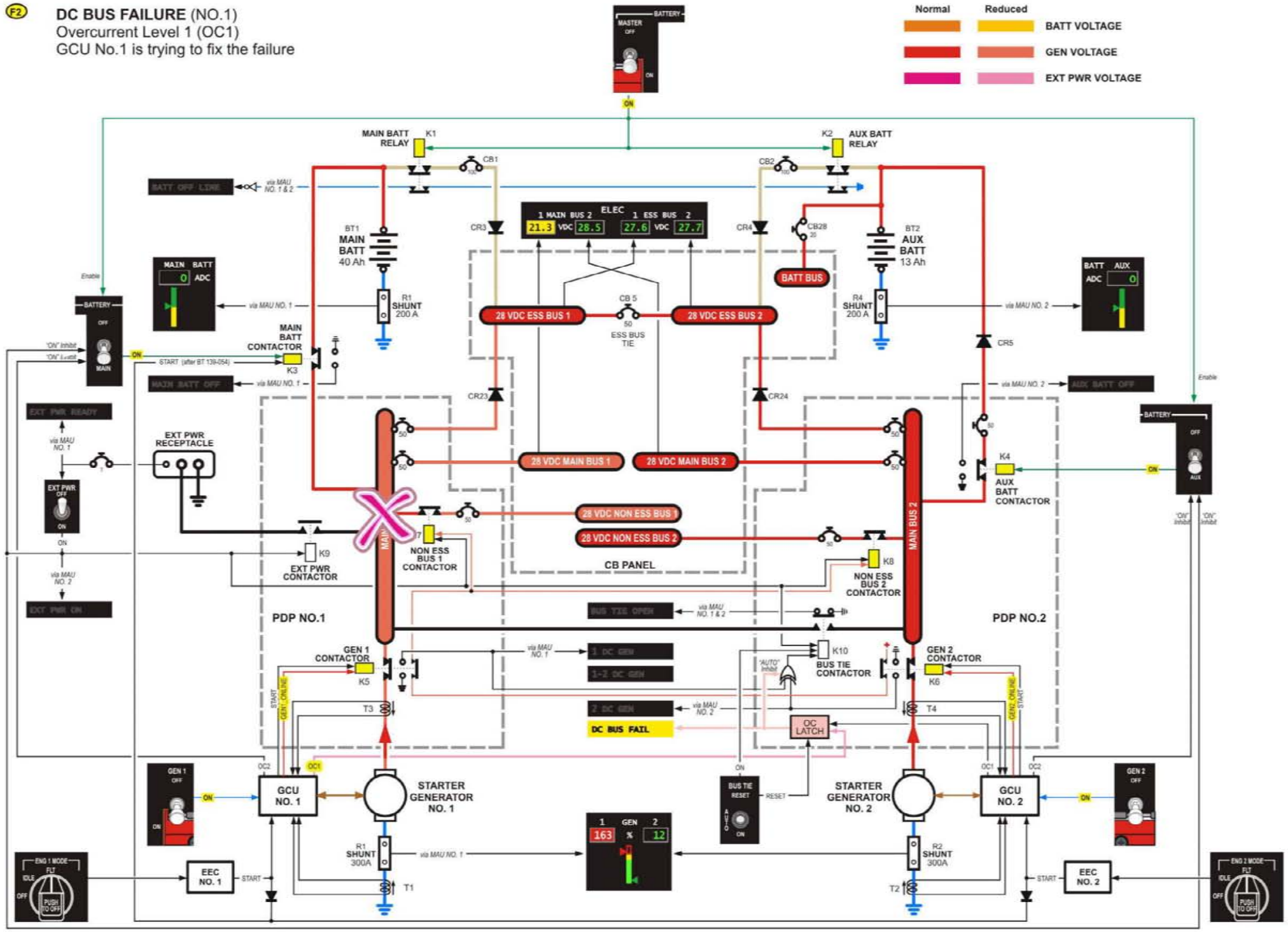
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F1 – SINGLE DC GENERATOR FAILURE (NO.1)

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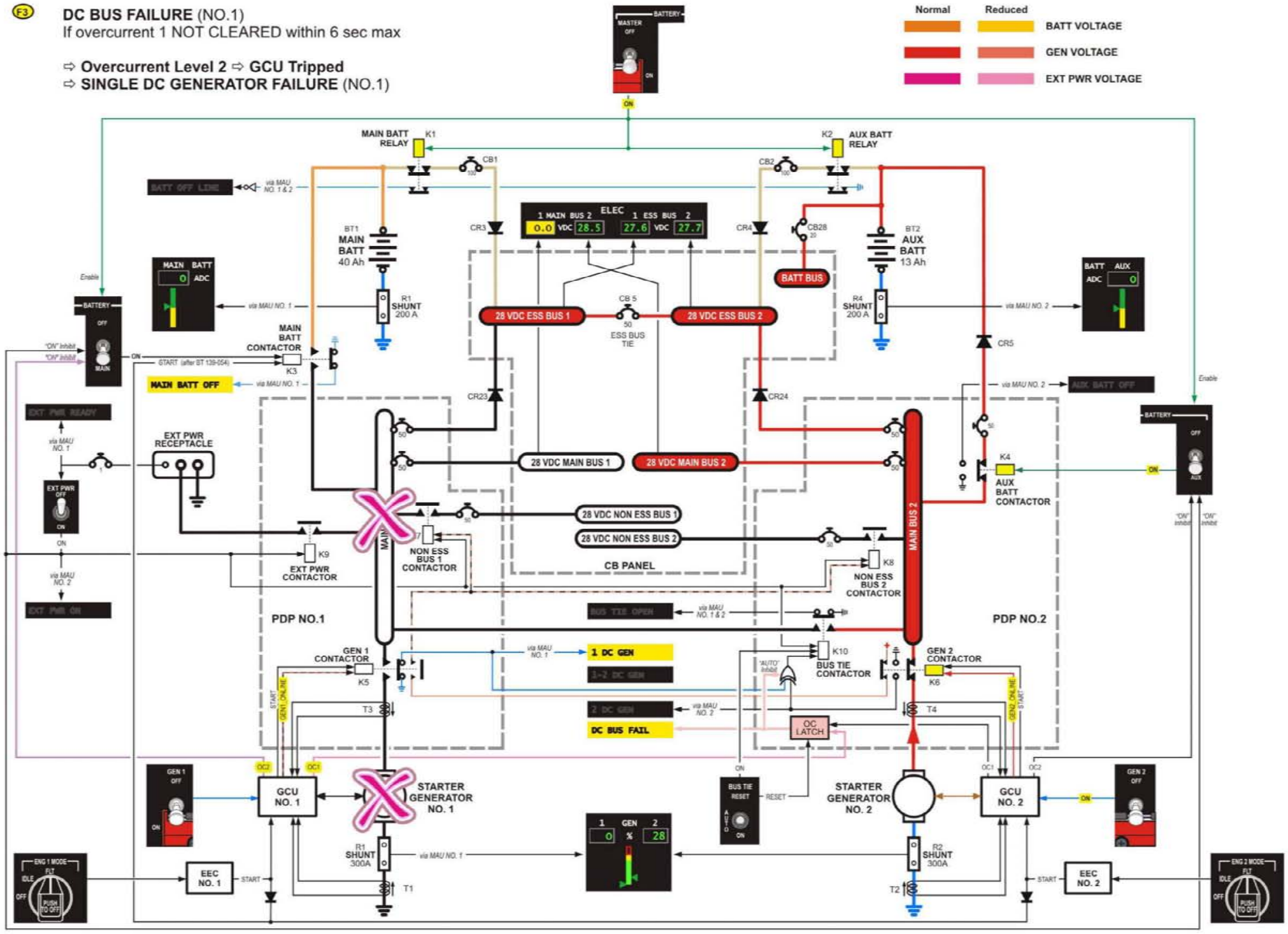


ICN-39-A-243000-G-A0126-00352-A-01-1

F2 – DC BUS FAILURE (NO.1) – OVERCURRENT LEVEL 1

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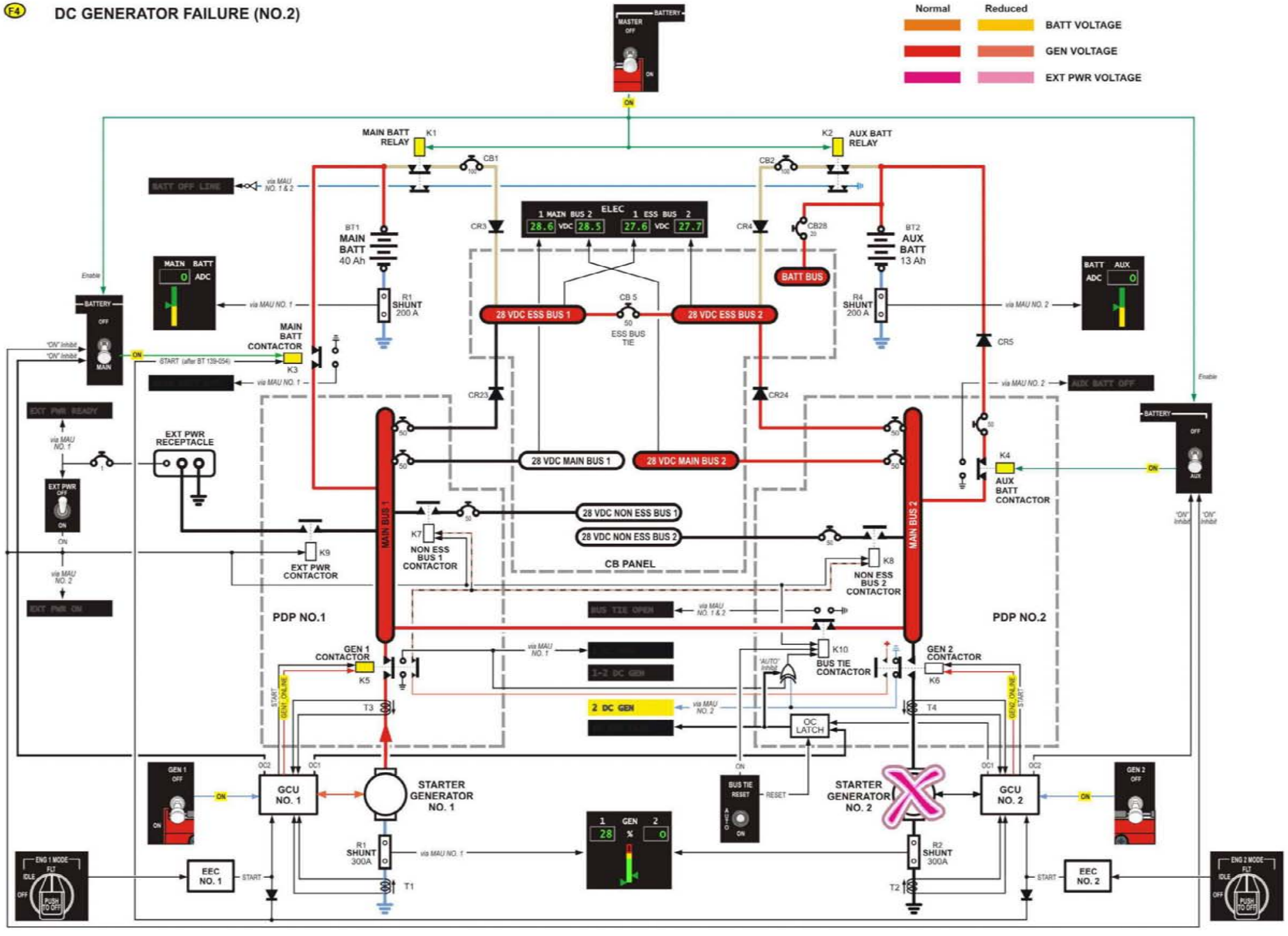


ICN-39-A-243000-G-A0126-00353-A-01-1

F3 – DC BUS FAILURE (NO.1) – IF OVERCURRENT NOT CLEARED WITHIN 6 SEC MAX

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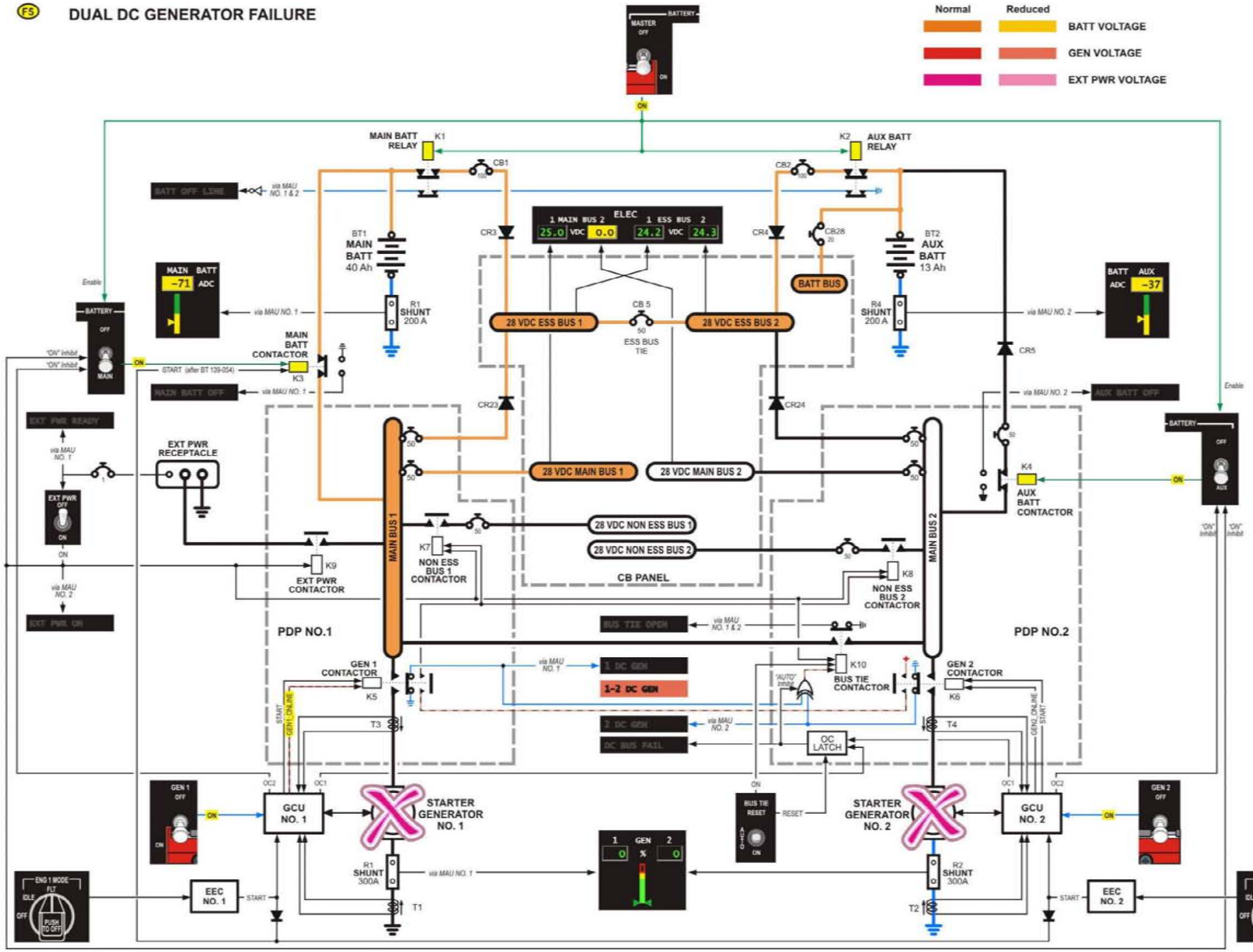
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ICN-39-A-243000-G-A0126-00354-A-01-1

F4 – DC GENERATOR FAILURE (NO.2)

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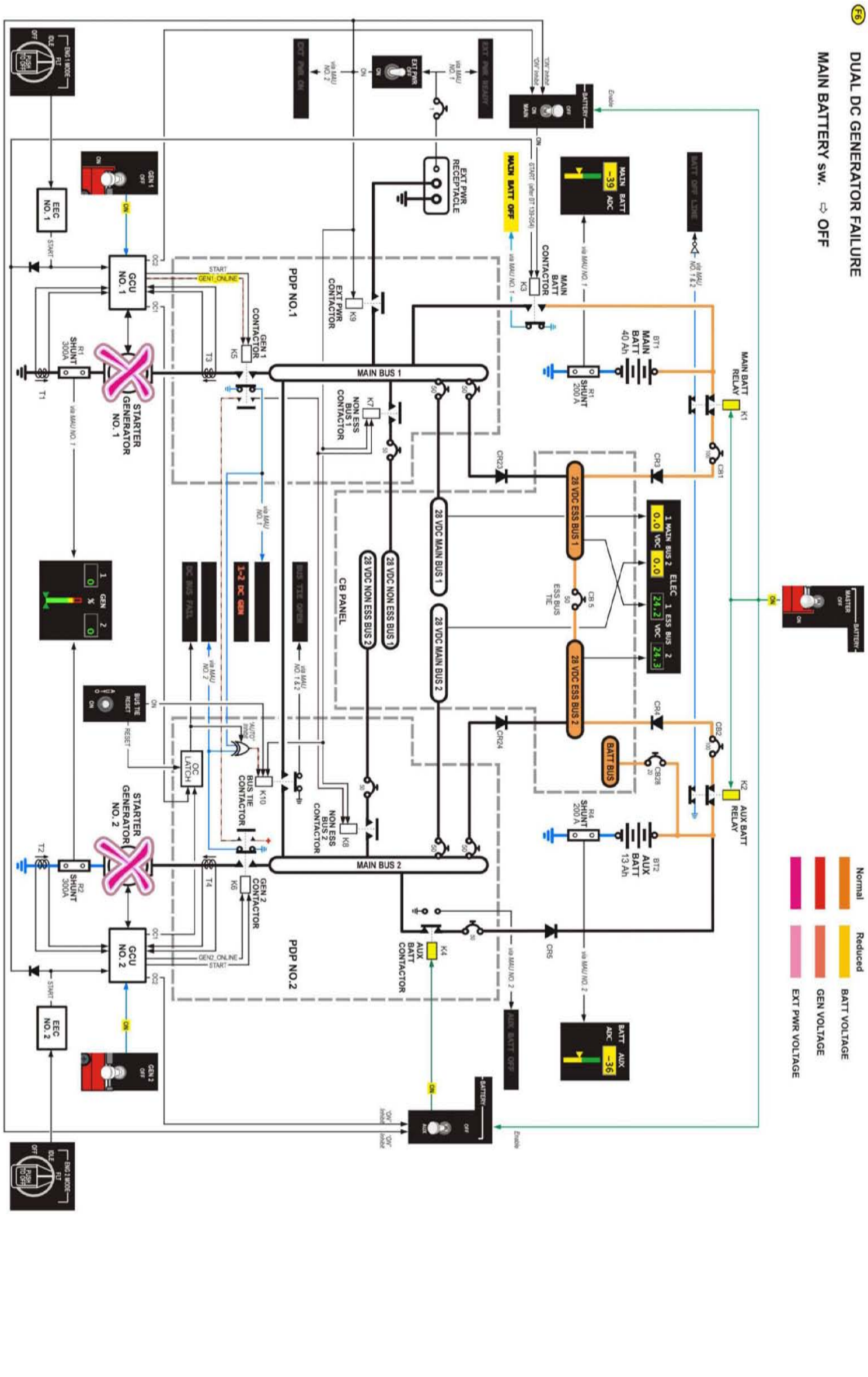


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F5 – DUAL DC GENERATOR FAILURE

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F6 – DUAL DC GENERATOR FAILURE – MAIN BATTERY SWITCH OFF

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SECTION 40 - EXTERNAL POWER

DESCRIPTION

The external power system includes these primary components:

- External power receptacle
- External power receptacle door switch
- External power control switch (S5).

EXTERNAL POWER RECEPTACLE

The external power receptacle is contained in a housing with an access door. It is installed on the right side of the helicopter nose.
Two screws and two washers attach the external power receptacle to the nose structure.
The external power receptacle has three pins. They are a large positive pin, a large ground pin and a small pin for the control signal input.

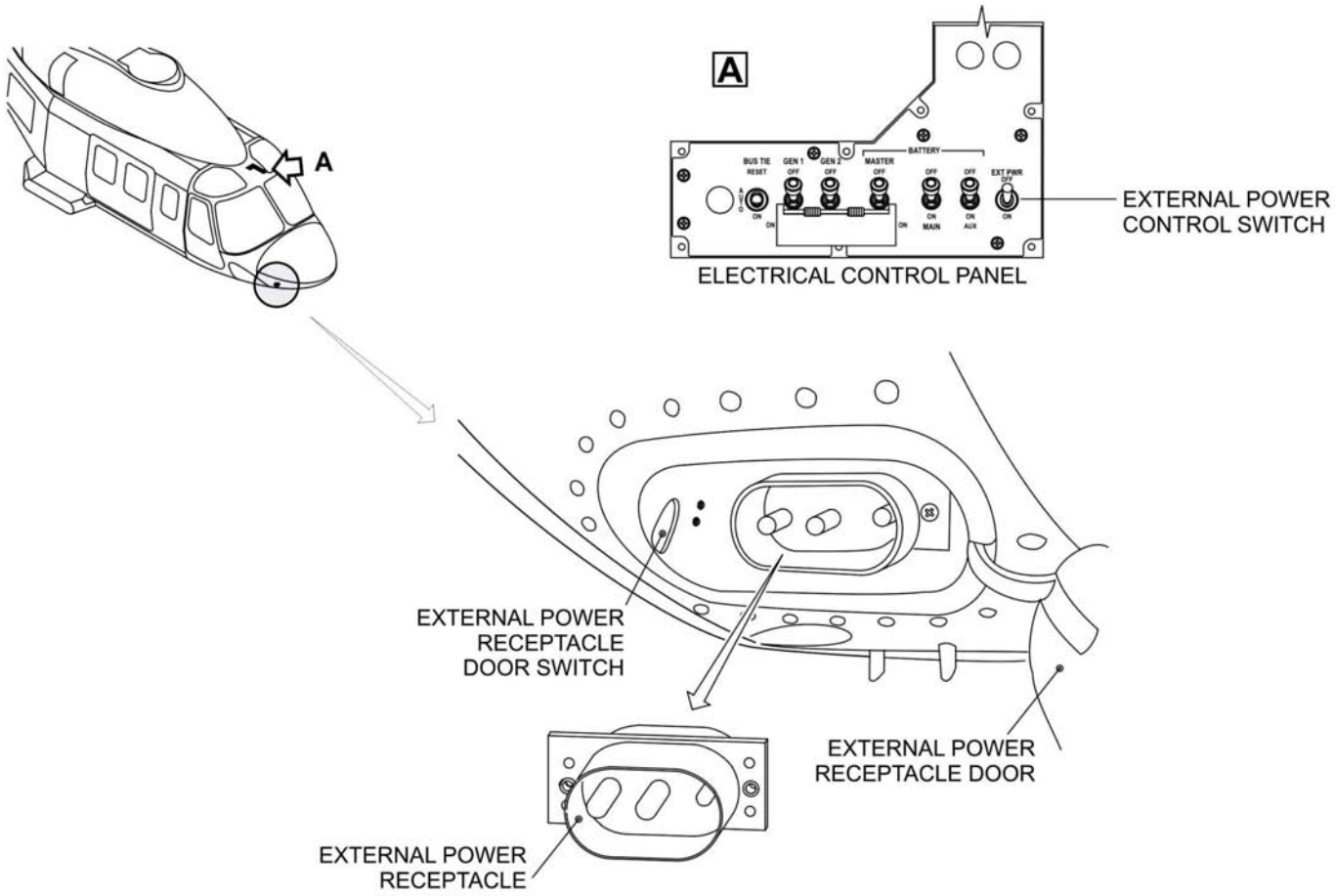
EXTERNAL POWER RECEPTACLE DOOR SWITCH

The external power receptacle door has a switch (S22). Two screws and two washers attach the switch to the nose structure.

EXTERNAL POWER CONTROL SWITCH

The external power control switch (S5) (EXT PWR) is installed on the electrical control panel on the overhead console.
It is a two position switch with the two selection that follow:

- OFF (up)
- ON (down).



ICN-39-A-244000-G-A0126-00371-A-01-1

EXTERNAL POWER SYSTEM PRIMARY COMPONENTS

OPERATION

External Power Receptacle

The external power receptacle connects an external power source to the helicopter electrical power generation system.
When the external power source is connected and operates, it overrides the helicopter power generation system. This occurs because the control circuits of the Number 1 Power Distribution Panel (PDP1) isolate the batteries and the generators from the external power source.
The external power receptacle has three pins. They are:

- a large positive pin
- a large ground pin and
- a small pin for the control signal input.

The positive pin is connected to the external power contactor of the relay K9. When this contactor is energized, it supplies external DC power to the Number 1 main bus.
The contactor K9 is energized when the EXT PWR switch is set to ON.
The other large pin is connected to ground.
The PDP1 senses the connection of the small pin (AUX) when the external power plug is connected.
The control circuit contained in the PDP1 is on and operates:

- The battery isolate relay (K17)
- The power changeover relays (K13) and (K19)
- The advisory indicator (EXT PWR RDY)
- The EXT PWR switch
- The over-voltage and latching relays (K11) and (K21).

The relay K19 and the relay K13 send an overlap signal to the related bus contactor. Thus electrical power is supplied to the essential and main busses without black-outs also when the power source changes (from the external power source to the helicopter generator system).

External Power Receptacle Door Switch

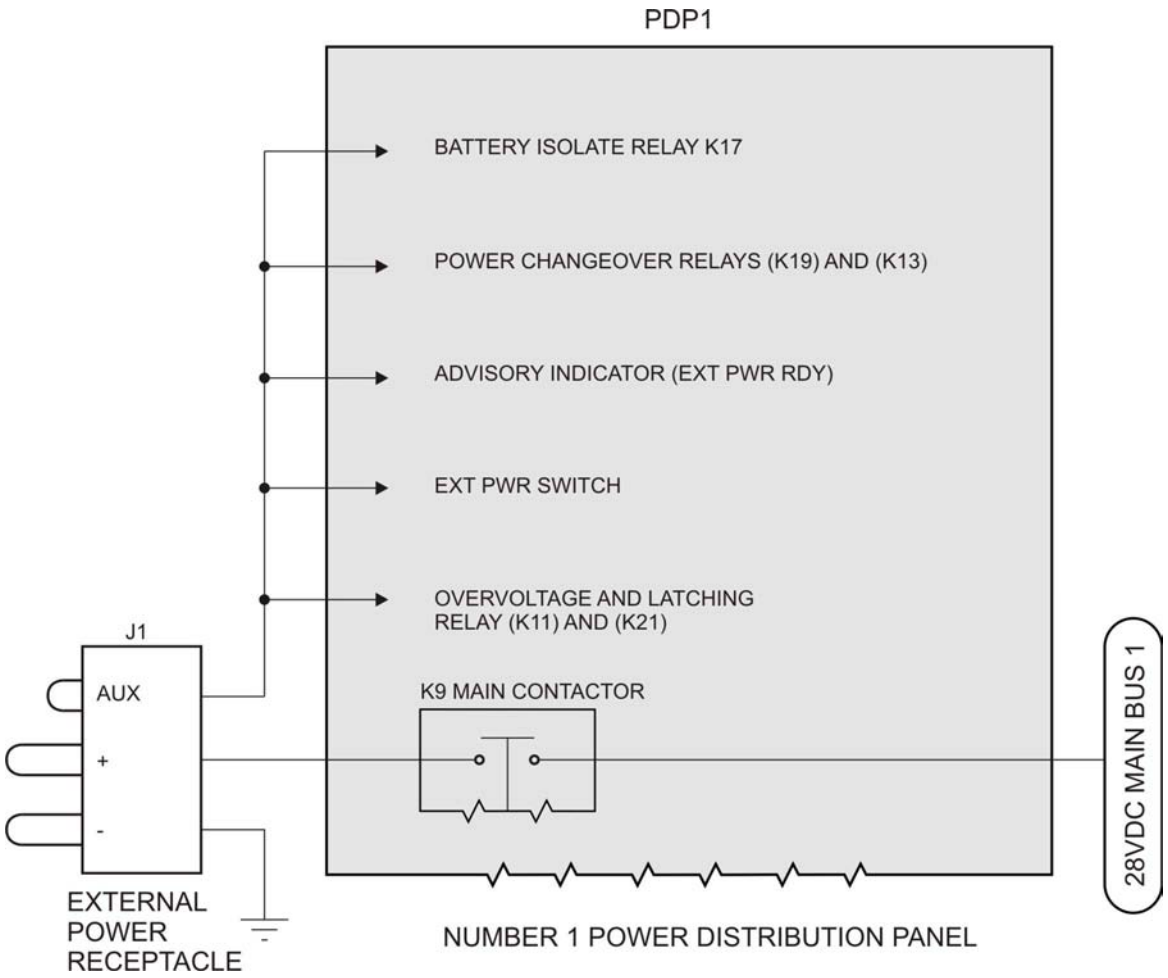
The external-power receptacle-door switch (S22) monitors the position of the external-power receptacle-access door. When the door is open, the EXT PWR DOOR caution is in view on the Multi Function Display (MFD) in the cockpit.

External Power Control Switch

The external power control switch (S5) (EXT PWR) is of the ON/OFF type.
When the switch is set to ON:

- If no external power over-voltage is sensed, the external power source is connected to the Number 1 main bus
- If no other electrical power source is connected to the DC power distribution system the non essential busses are energized
- The two main busses are connected
- All the helicopter loads are energized.

When the external power source operates, the batteries and generators are isolated from the main busses of the helicopter.
When the switch is set to OFF: the external power source does not energize the main busses of the helicopter. An advisory legend is in view in the cockpit if the plug of the external power source continues to be connected to the helicopter receptacle.



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EXTERNAL POWER RECEPTACLE – SCHEMATIC DIAGRAM

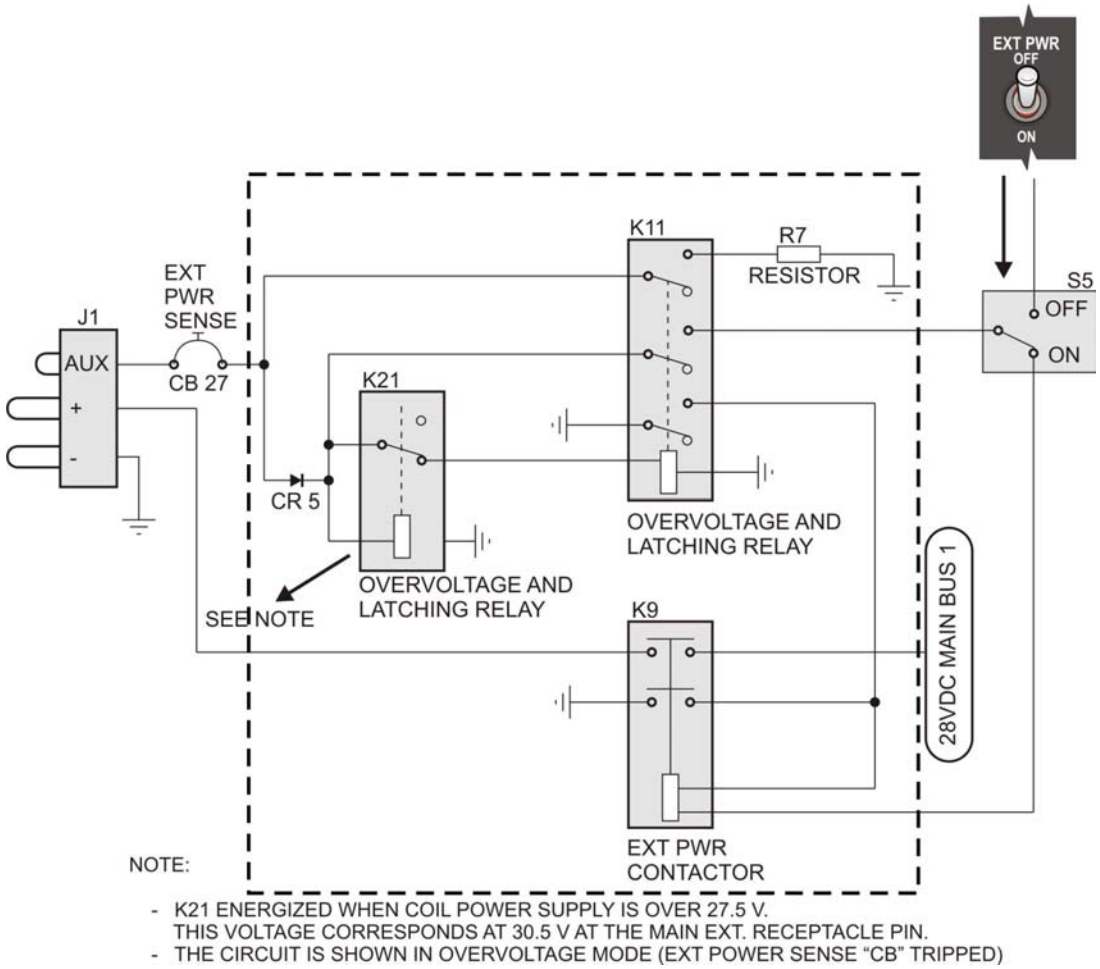
SYSTEM OPERATION

Main Power Connection

When you open the access door of the external power receptacle (to connect the electrical power system to the helicopter busses), the related switch sends a signal to the Number 1 Modular Avionic Unit (MAU1). This signal is used to cause the EXT PWR DOOR caution legend to come on in the MFD.
When the external power source is connected to the helicopter, the EXT PWR switch is supplied with 28 VDC power. The power is supplied through the EXT PWR SENSE circuit breaker and the contacts of the overvoltage trip relay K11. These contacts are usually closed.
At the same time the EXT PWR READY legend comes in view in the cockpit.
When the EXT PWR switch is moved from OFF to ON:

- The relay K9 becomes energized
- The large positive pin of the external power receptacle connects to the Number 1 main bus (MAIN BUS1)
- The EXT PWR ON legend comes in view on the MFD in the cockpit.

The relay K19 is of the time delay type. Its contacts, that are usually open, close when the control power input and the relay coil are energized. The relay does not release for 0.5 more seconds, and power continues to be supplied during that time.
When the relay K19 operates the contactors K7 and K8 of the non-essential feeder bus become energized. The two contactors connect the non-essential busses to the main busses.
At the same time the bus-tie contactor (K10) operates, and connects the two MAIN busses to each other.
The external power source thus energizes all the DC busses of the helicopter.



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MAIN POWER CONNECTION – SCHEMATIC DIAGRAM

Over-Voltage Protection

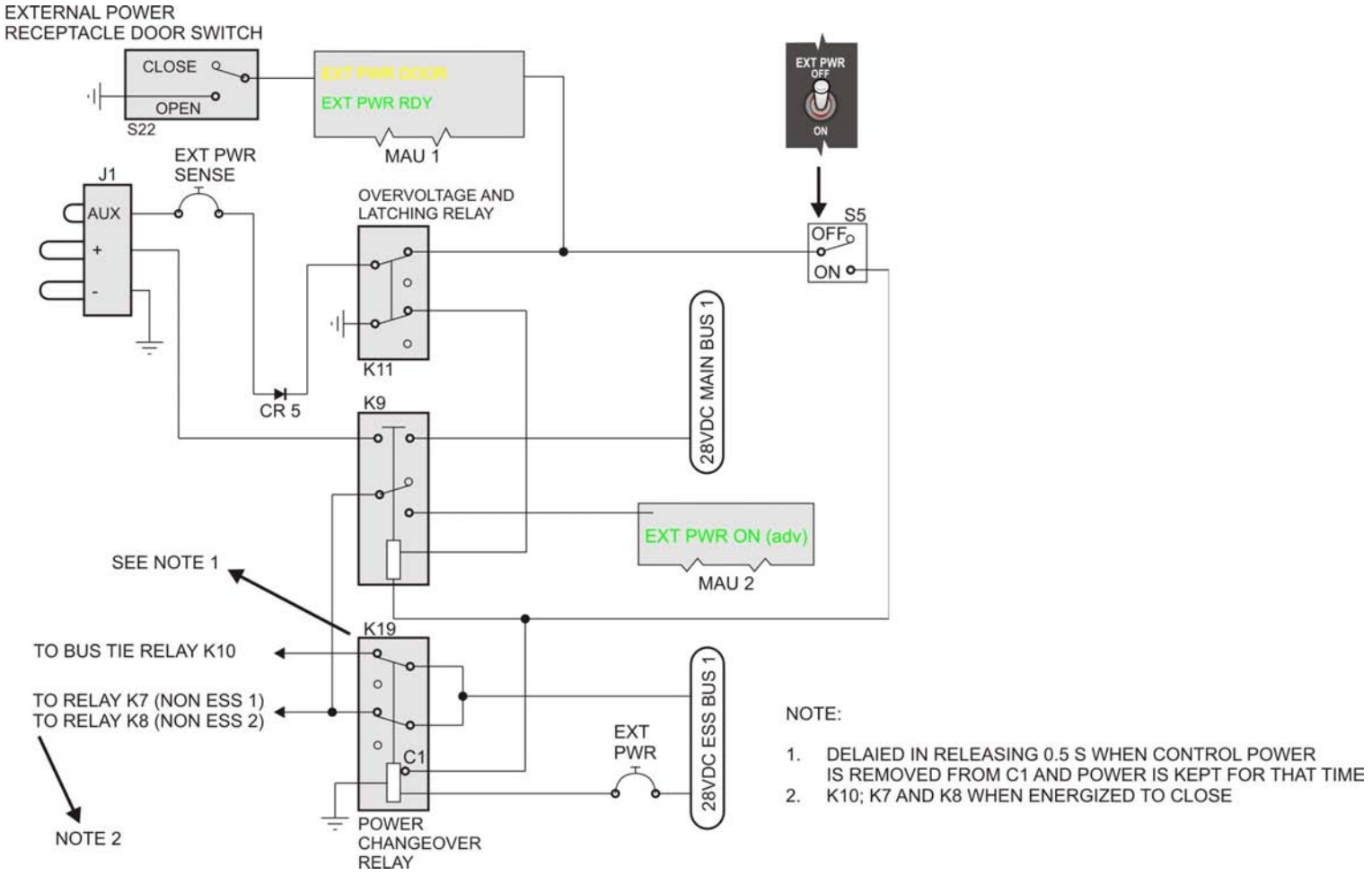
The relay K21 becomes energized when the nominal voltage of 30.5 V at the input pin of the external power receptacle is sensed. (The voltage threshold is 29.7 V).
The coil of relay K11 is energized because:

- The EXT PWR switch is ON
- The relay K21 is energized.

When the relay K11 is energized:

- The circuit of the EXT PWR switch is open
- The ground to the coil of relay K9 is disconnected
- The external power is removed from the main bus.

The relay K11 connects the load resistor R7 to the external power input, and this opens the related circuit breaker. The open circuit breaker prevents the supply of power from the external electrical power source.
To supply again electrical power from the external power source, let the voltage decrease, and close the circuit breaker.



ICN-39-A-244000-G-A0126-00374-A-01-1

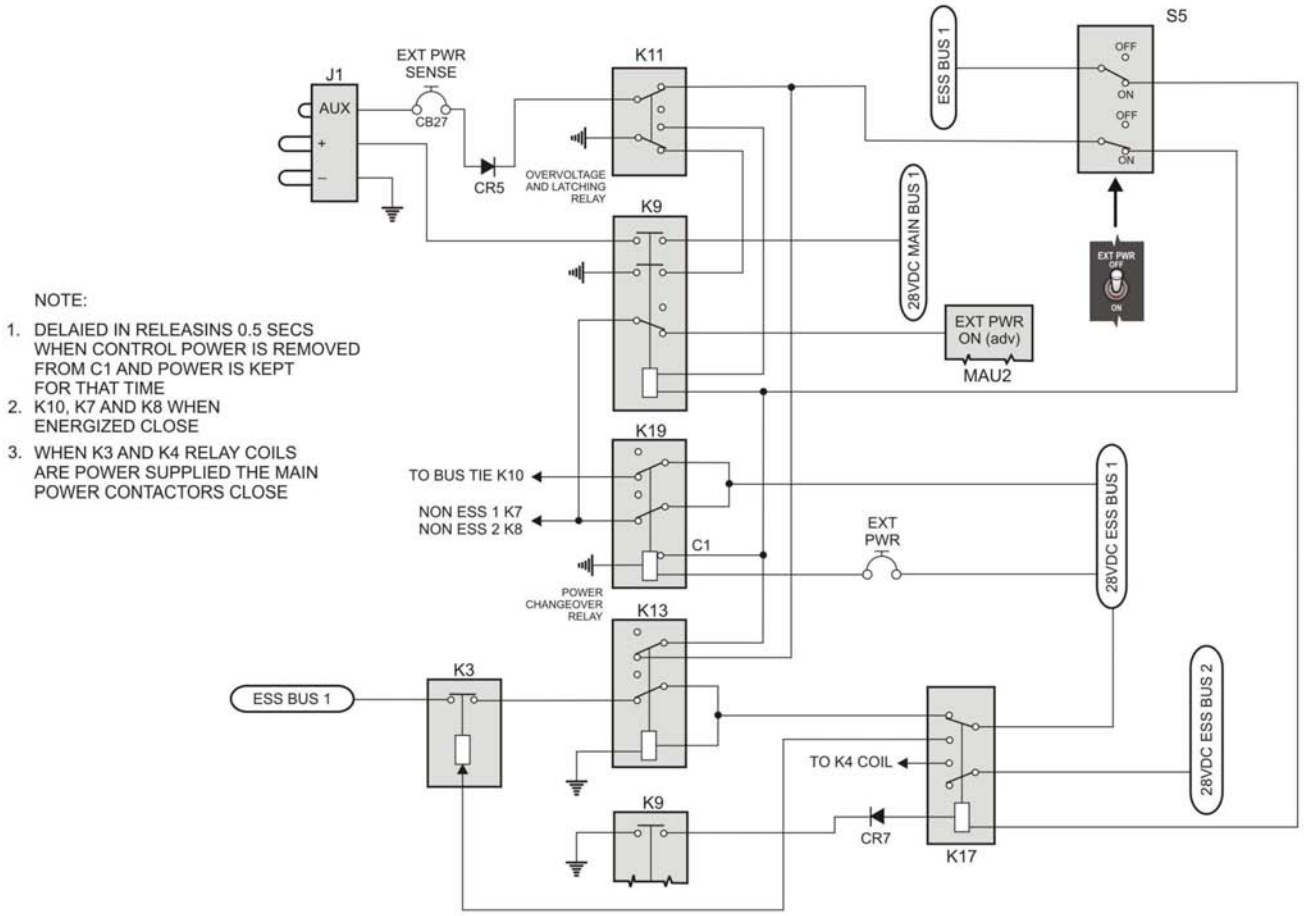
OVERVOLTAGE PROTECTION – SCHEMATIC DIAGRAM

SWITCHING FROM EXT PWR TO HELICOPTER POWER

The initial conditions are:

- EXT PWR switch ON
- BATT MASTER switch ON
- MAIN BATT switch ON (K1 energized and main battery connected to Number 1 essential bus)
- AUX BATT switch ON (K2 energized and auxiliary battery connected to Number 2 essential bus).

The open contacts of the relay K17 (energized) prevent the operation of the main battery relays K3 and K4.
At the same time, the closed contact of the relay K17 energize the relay K13 through the Number 1 essential bus.
Relay K13 receives voltage from the essential bus through a contact of relay K3 that is usually closed.
It makes a full circuit between the external power source and the energized coil of relay K9. It also controls the voltage to the control pin of relay K19.
When the EXT PWR switch is at OFF, the relay K 13, which is connected, operates the relay K9 and the time-delay relay K19. Thus, the external power continues to be supplied to the bus.
At the same time, power is removed from the relay K17, and voltage is supplied to the relays K3 and K4.
When the relays K3 and K4 are energized, each battery is connected to the related main bus. The auxiliary contact of the relay K3, which is usually closed, opens.
The latching voltage from the coil of relay K13 is removed. And power is removed also from the relays K9 and K19.
The relay K19 stays energized for approx. 0.5 s after the relay K13 releases. It keeps the busses connected during that period. Thus power from the batteries is supplied to all the busses after the external power is removed and until the generators are on-line.



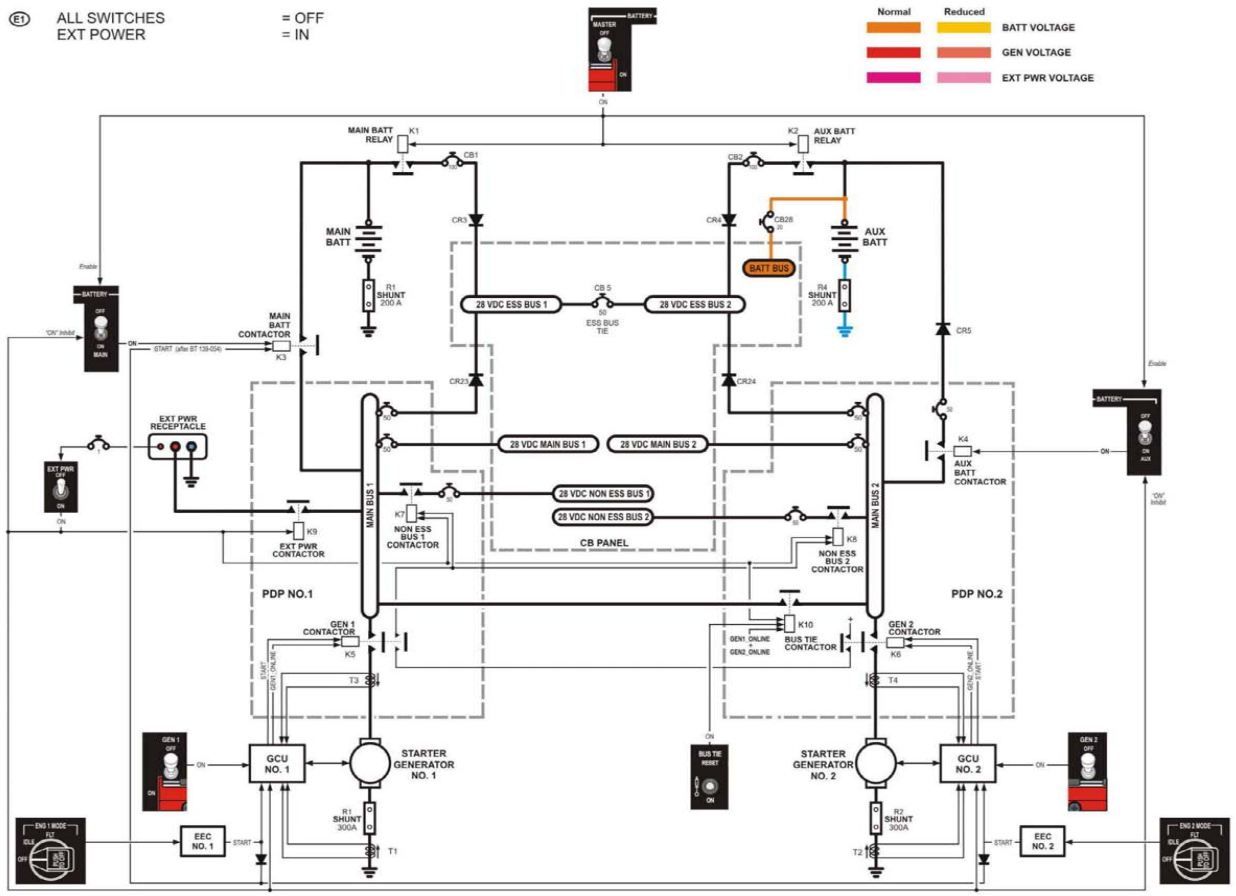
ICN-39-A-244000-G-A0126-00375-A-01-1

SWITCHING FROM EXT PWR TO HELICOPTER POWER

EXTERNAL POWER STARTING

Step E1 – External Power Available At Receptacle

When an external power voltage is available at the external power receptacle, it is checked for reverse polarity and overvoltage but it is not automatically connected to the aircraft distribution bus bars.



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STEP E1 – EXTERNAL PWR AVAILABLE AT THE RECEPTACLE

(Printed A3 format at the end of data module)

STEP E2 – CONNECTION OF THE EXTERNAL POWER

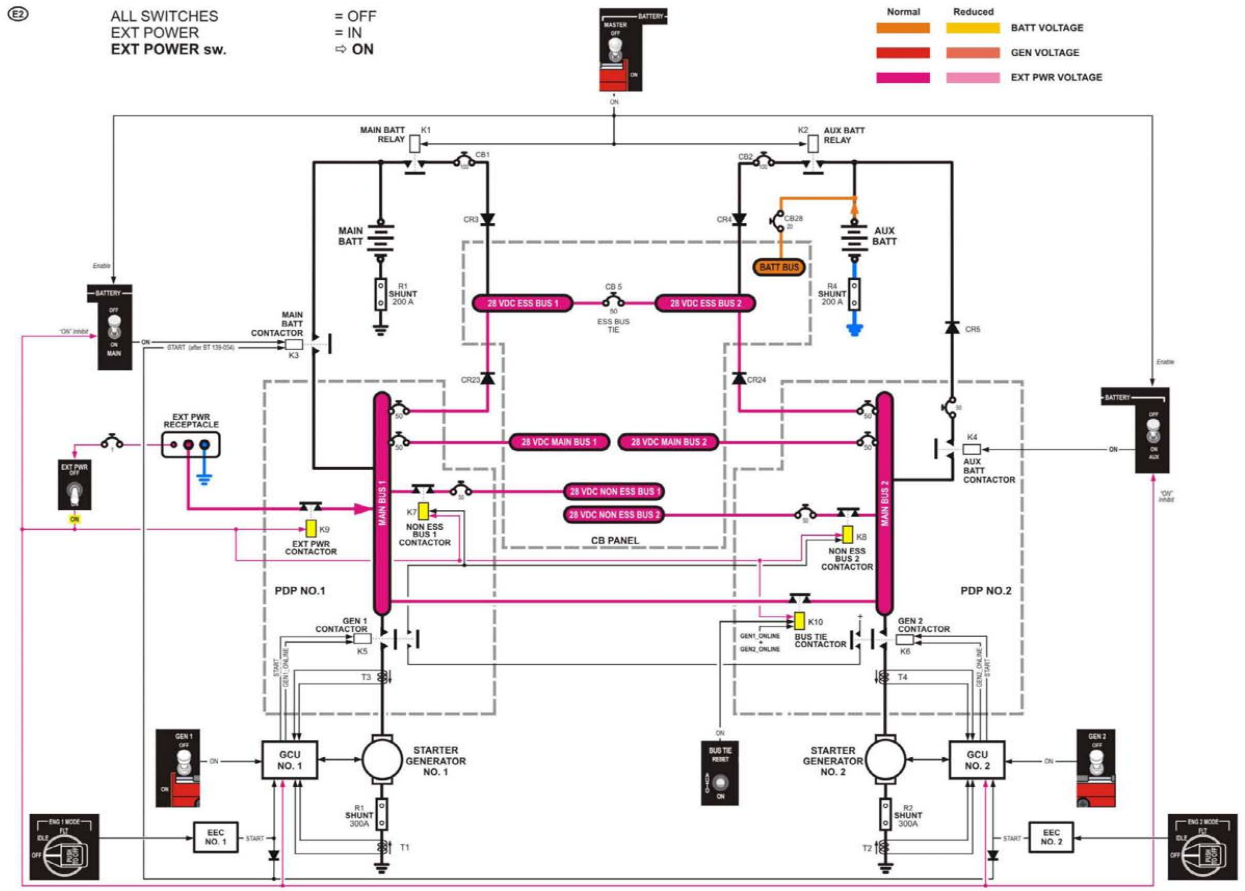
The EXT PWR switch on the Electrical control panel allows the pilot to control the connection of the external power source to the distribution bus bars. When a valid external power is available at the receptacle and the EXT PWR switch is moved to ON:

- EXT PWR contactor closes
- BUS TIE contactor automatically closes
- NON ESS BUS 1 & NON ESS BUS 2 contactors close
- MAIN and AUX battery connection circuits to MAIN BUS 1 and 2 are inhibited to prevent recharging of the batteries from external power
- Both GCU's are fed with a control signal that prevents generators to be put on-line while external power is feeding the aircraft loads

The external power supplies all distribution bus bars (ESS, MAIN and NON ESS).

NOTE A

Battery bus is supplied by the AUX battery.



ICN-39-A-244000-G-A0126-00377-A-01-1

STEP E2 – CONNECTION OF THE EXTERNAL POWER

(Printed A3 format at the end of data module)

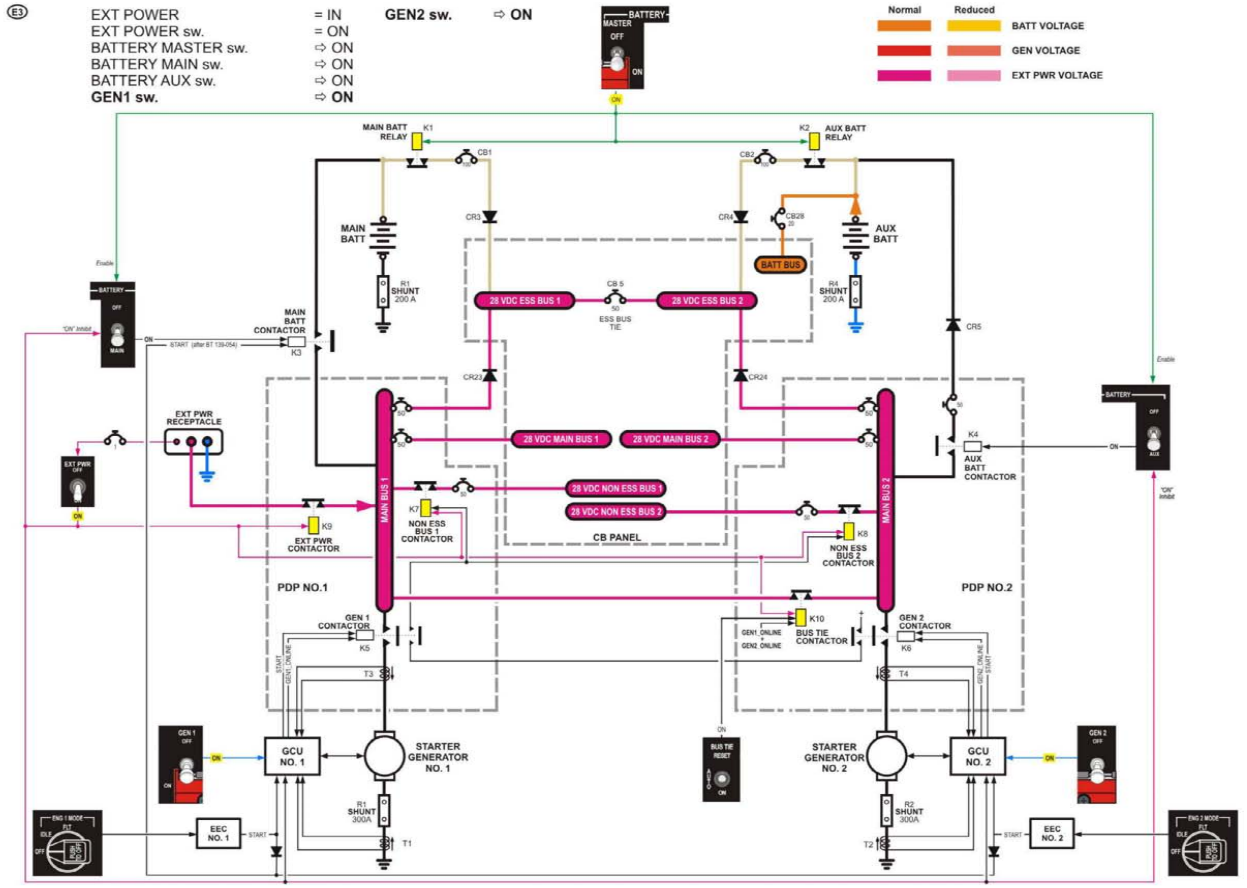
STEP E3 – BATTERIES AND EXT PWR STARTING

The following diagrams represent the sequence of actions for a normal battery starting of the engines using the external power. In this example engine no. 1 is started first. Refer to RFM – Section 2 for Normal procedures.
Steps 1 thru 3 discussed for Battery Starting are to be performed also in case of External Power Starting. When EXT PWR switch is then moved to ON:

- MAIN BATT contactor and AUX BATT contactor open to prevent recharging of the batteries from external power

NOTE B

Diodes CR1 and CR2 prevent the current flow from the ESS BUS 1 and 2 to the MAIN and AUX batteries



ICN-39-A-244000-G-A0126-00378-A-01-1

STEP E3 – BATTERIES AND EXT PWR STARTING

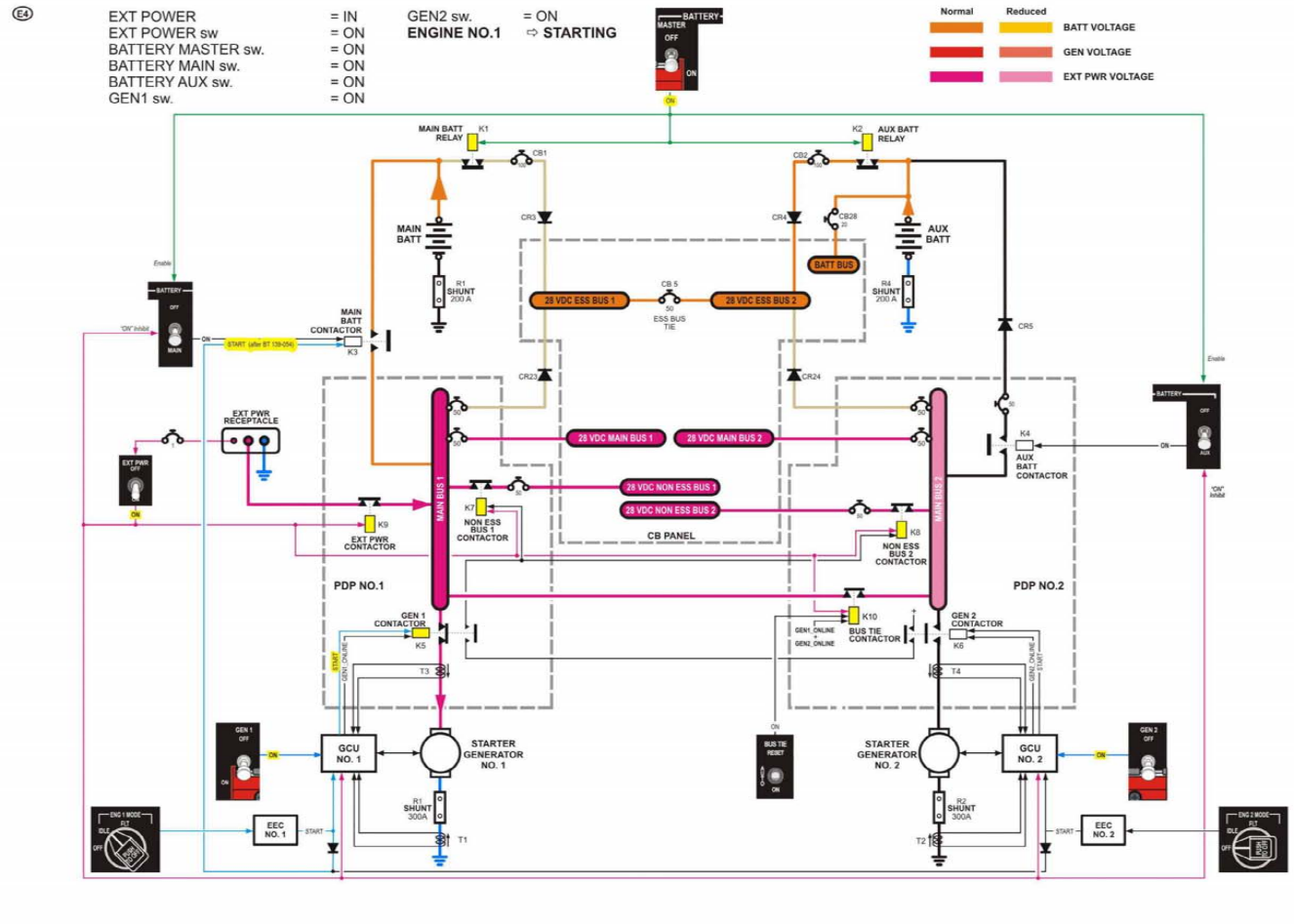
(Printed A3 format at the end of data module)

STEP E4 – ENGINE 1 STARTING

The Bollettino Tecnico 139-054 (Technical Bulletin 139-054) has introduced the concept of assisted start also when starting on External Power. During the start sequence (START control signal is active) the MAIN battery is connected to MAIN BUS 1 to allow a soft change-over of the starter power source in case the external power fails.

NOTE C

BUS TIE switch is to be moved to ON if engine no. 2 is started first.



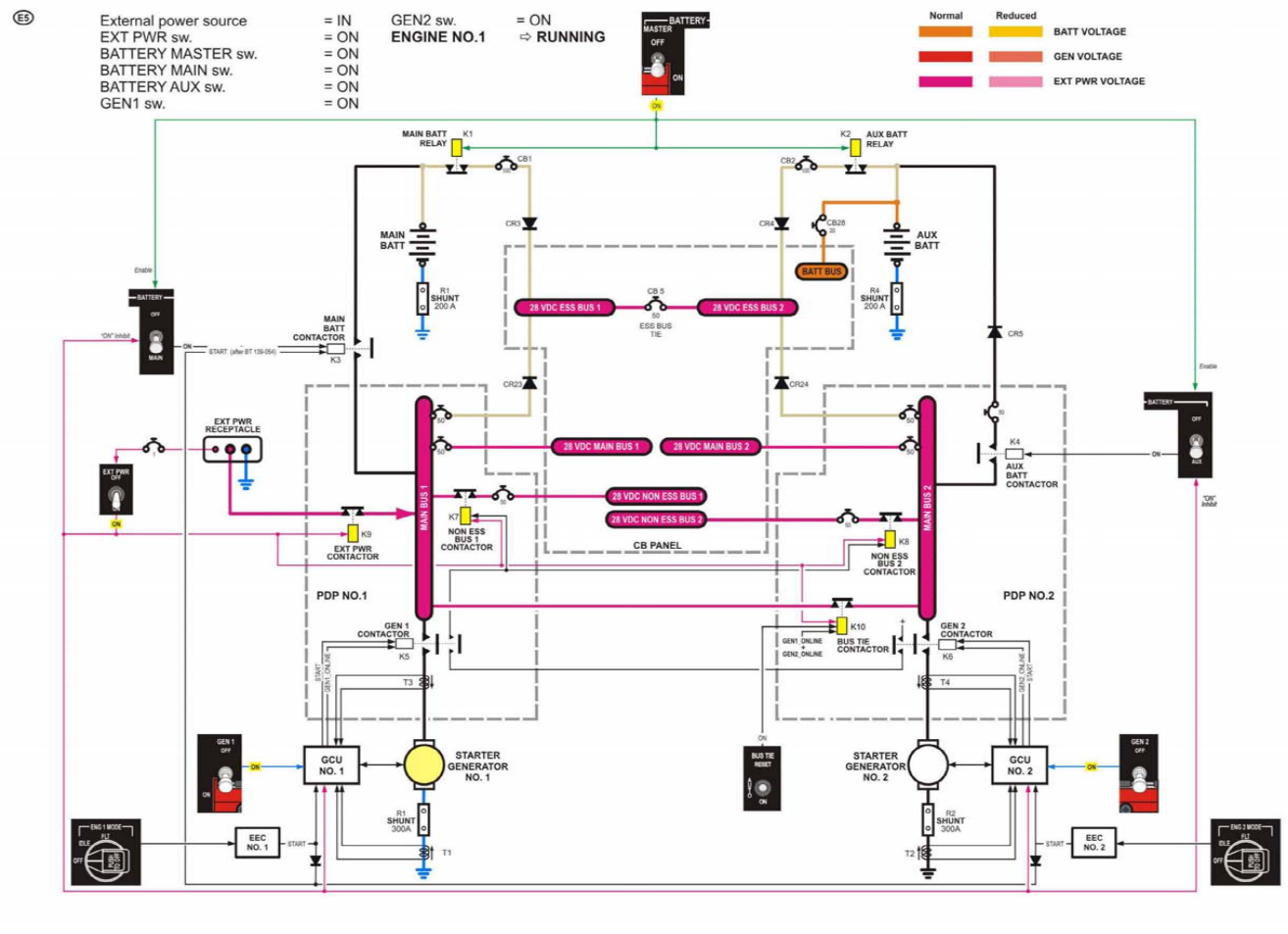
ICN-39-A-244000-G-A0126-00379-A-01-1

STEP E4 – ENGINE 1 STARTING

(Printed A3 format at the end of data module)

STEP E5 – ENGINE RUNNING

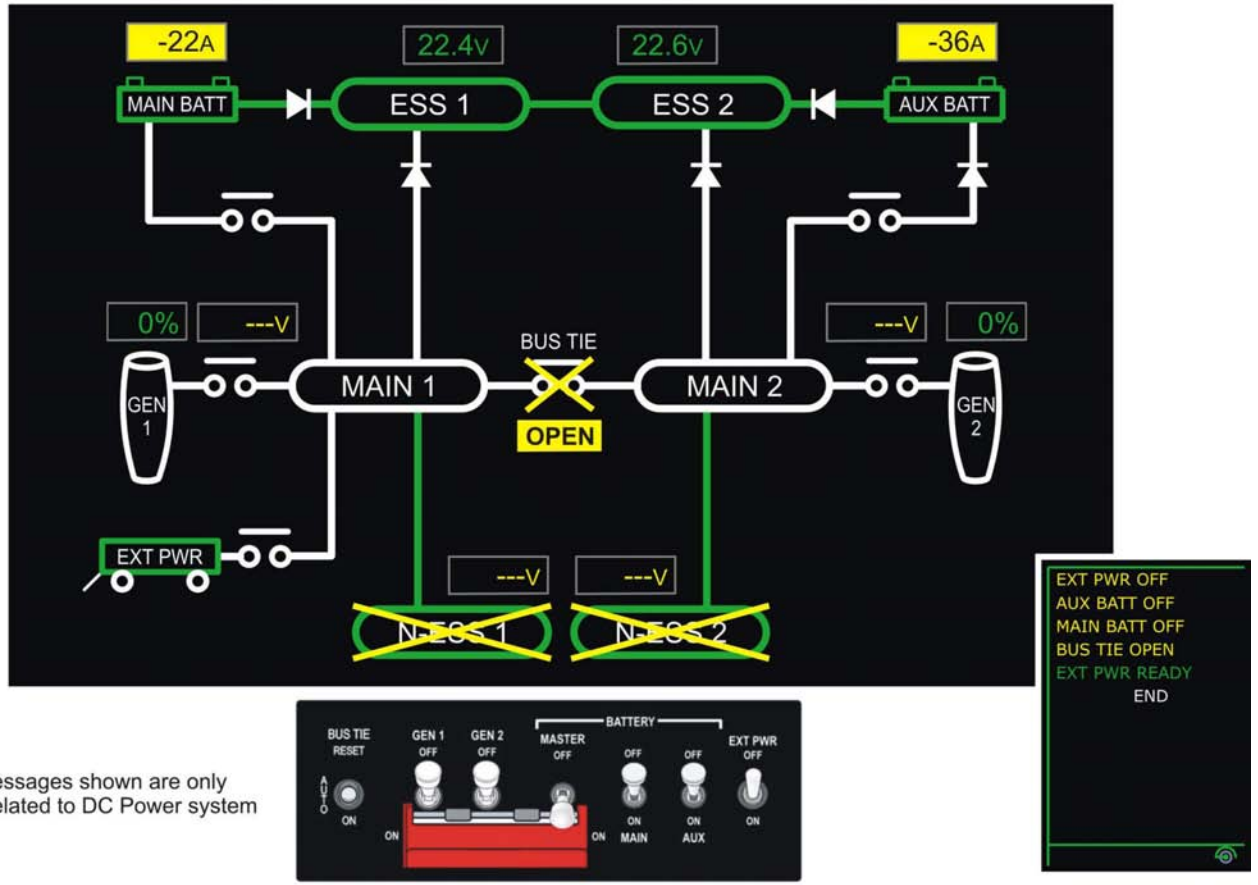
After engine starting, Generators remain off-line until the EXT PWR switch is returned to OFF.



ICN-39-A-244000-G-A0126-00380-A-01-1

STEP E5 – ENGINE RUNNING

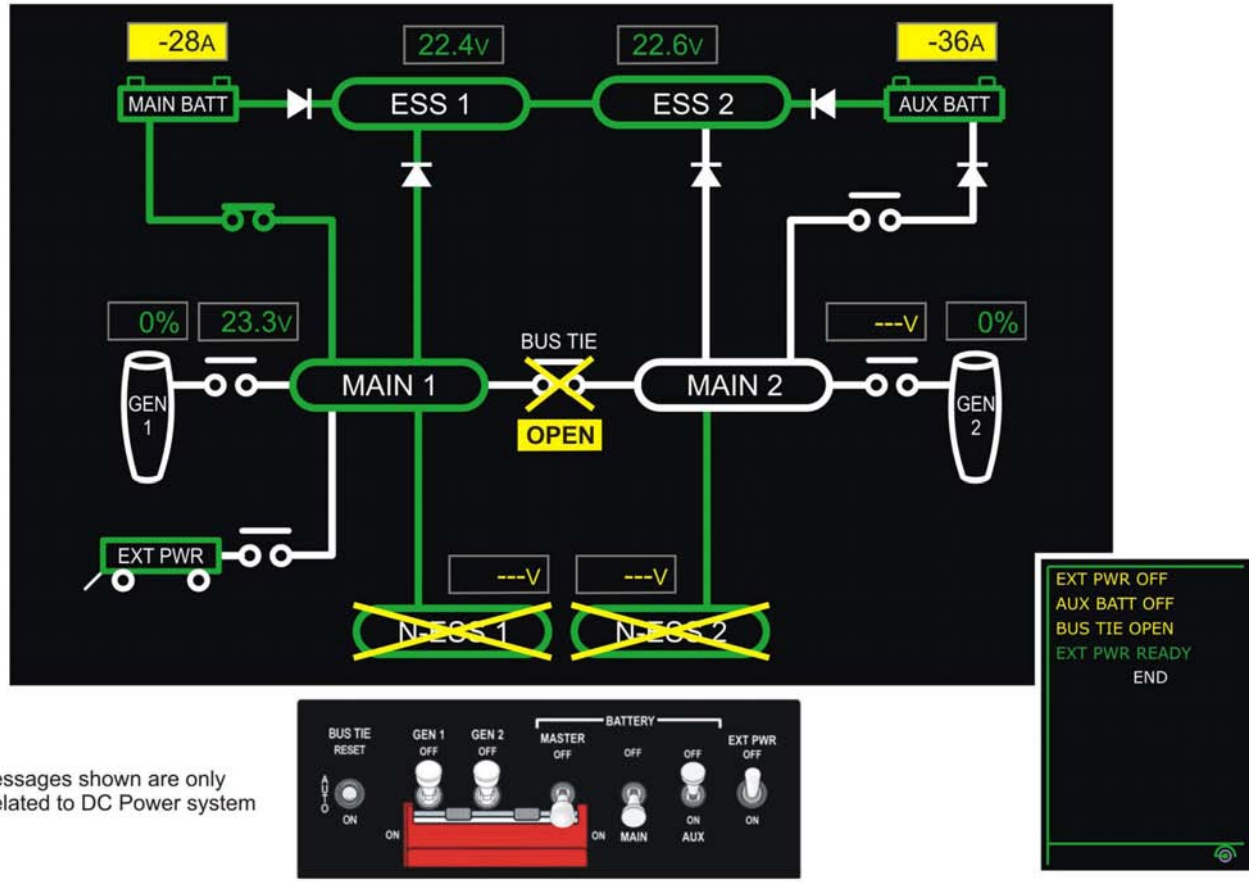
(Printed A3 format at the end of data module)



Note: CAS Messages shown are only those related to DC Power system

ICN-39-A-244000-G-A0126-00381-A-01-1

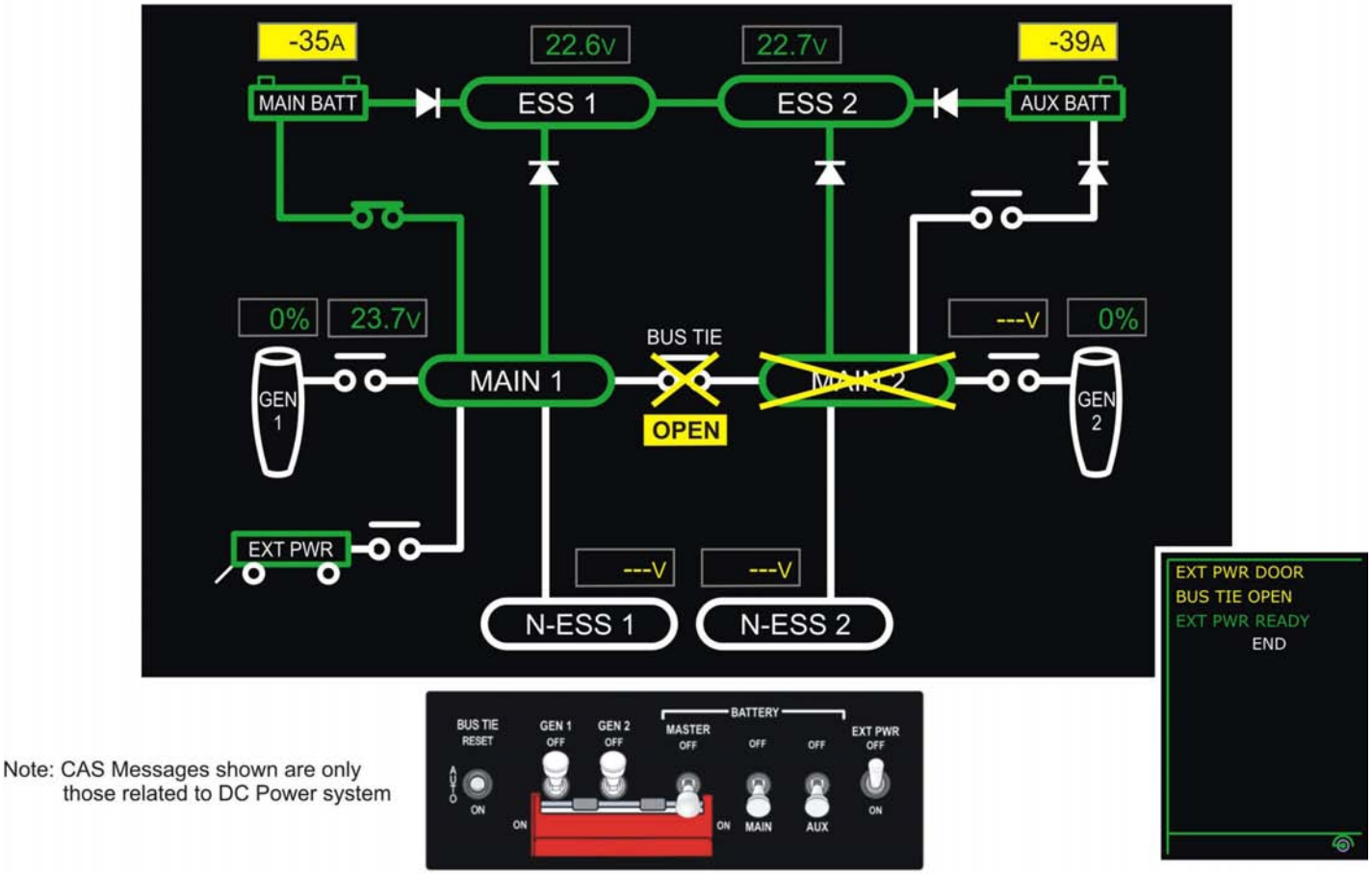
DC POWER SYNOPTIC – BATT MASTER = ON, EXT PWR = READY



Note: CAS Messages shown are only those related to DC Power system

ICN-39-A-244000-G-A0126-00382-A-01-1

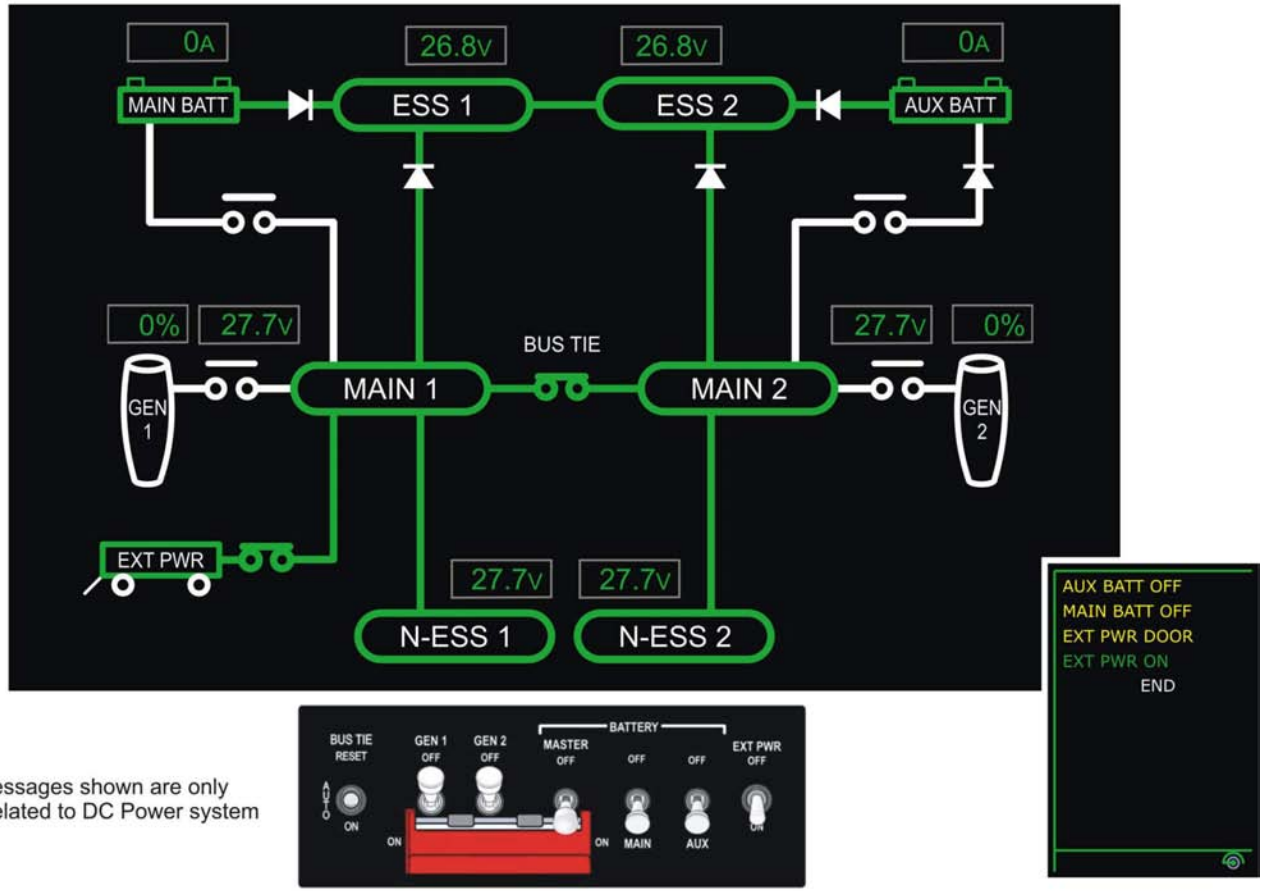
DC POWER SYNOPTIC – BATT MASTER + MAIN = ON, EXT PWR = READY



Note: CAS Messages shown are only those related to DC Power system

ICN-39-A-244000-G-A0126-00383-A-01-1

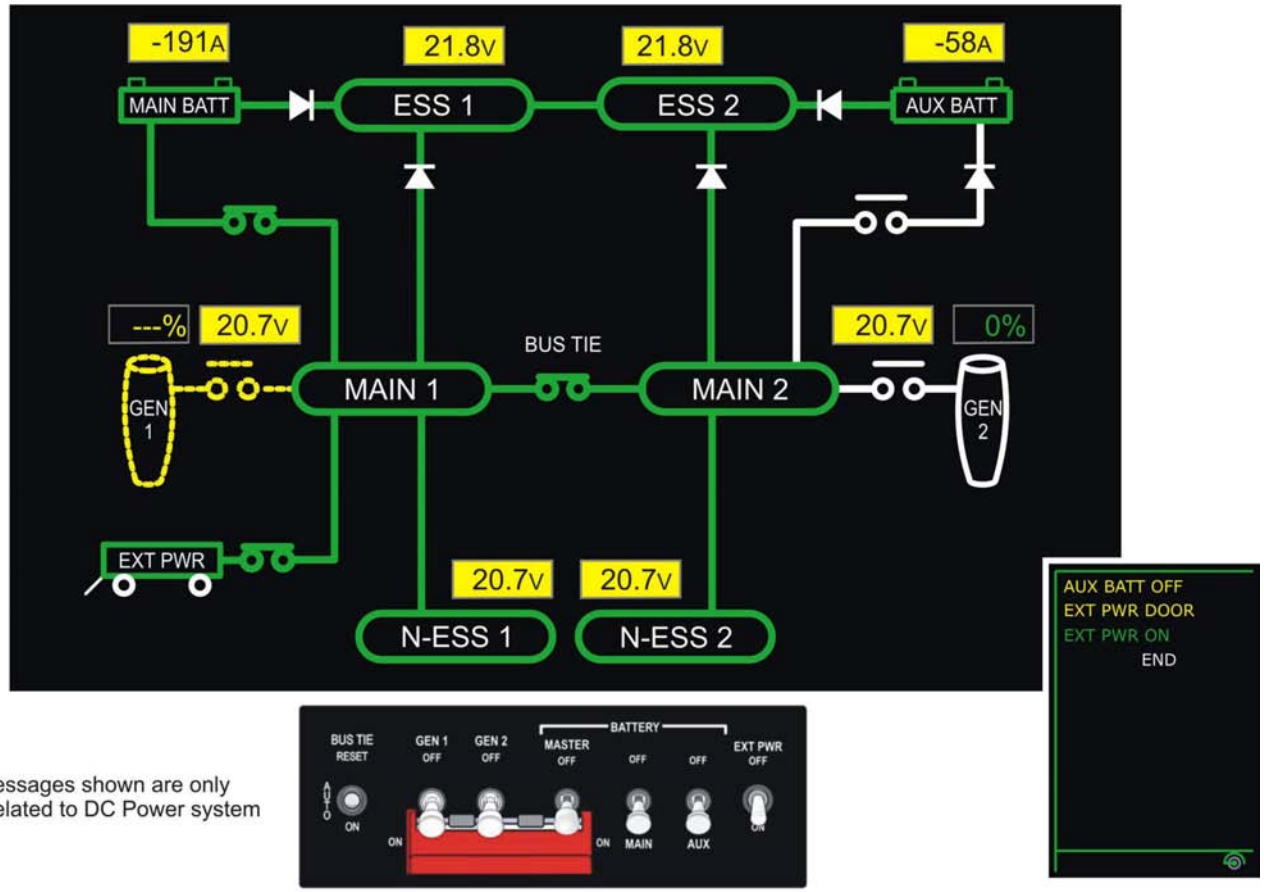
DC POWER SYNOPTIC – BATT MASTER + MAIN + AUX = ON, EXT PWR = READY



Note: CAS Messages shown are only those related to DC Power system

ICN-39-A-244000-G-A0126-00384-A-01-1

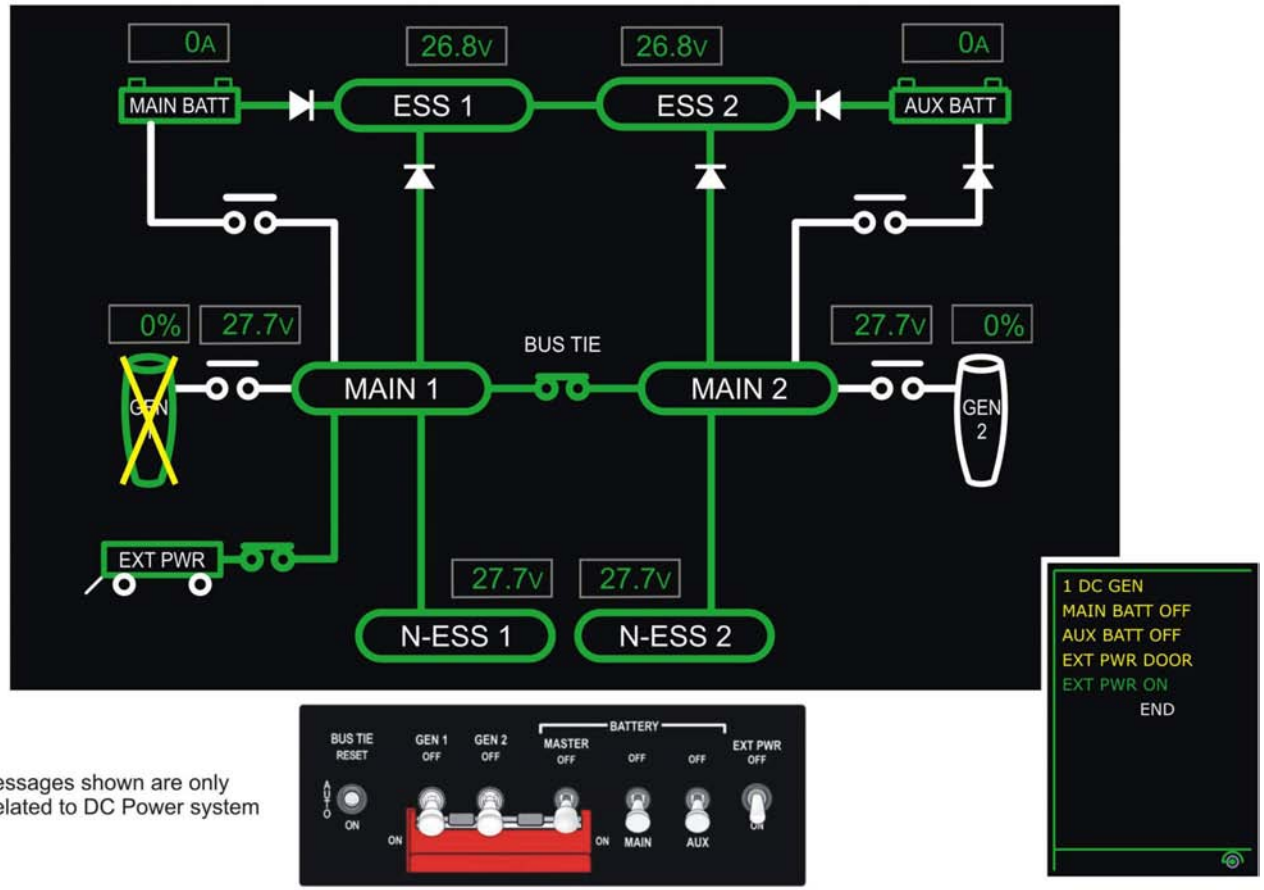
DC POWER SYNOPTIC – BATT + EXT PWR = ON



Note: CAS Messages shown are only those related to DC Power system

ICN-39-A-244000-G-A0126-00385-A-01-1

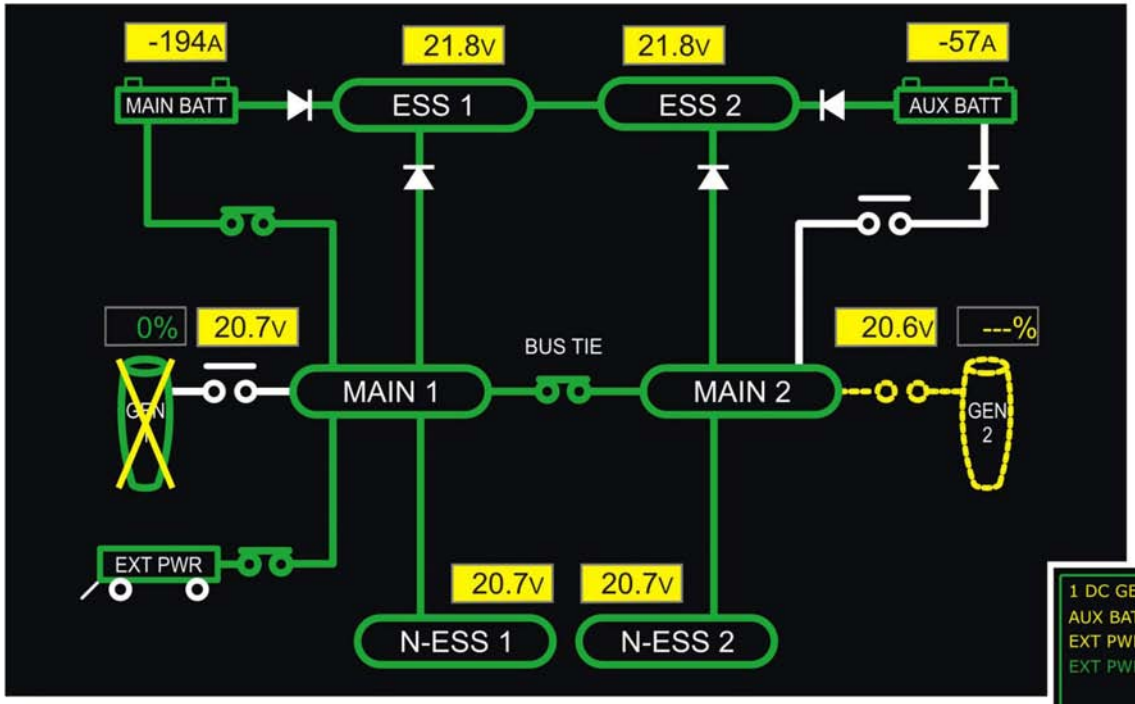
DC POWER SYNOPTIC – EXT PWR STARTING (ENGINE NO.1)



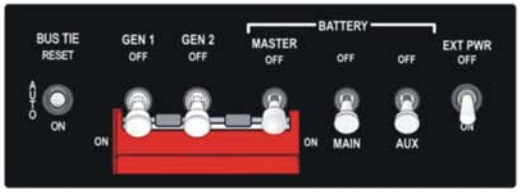
Note: CAS Messages shown are only those related to DC Power system

ICN-39-A-244000-G-A0126-00386-A-01-1

DC POWER SYNOPTIC – EXT PWR = ON, ENGINE NO.1 = RUNNING



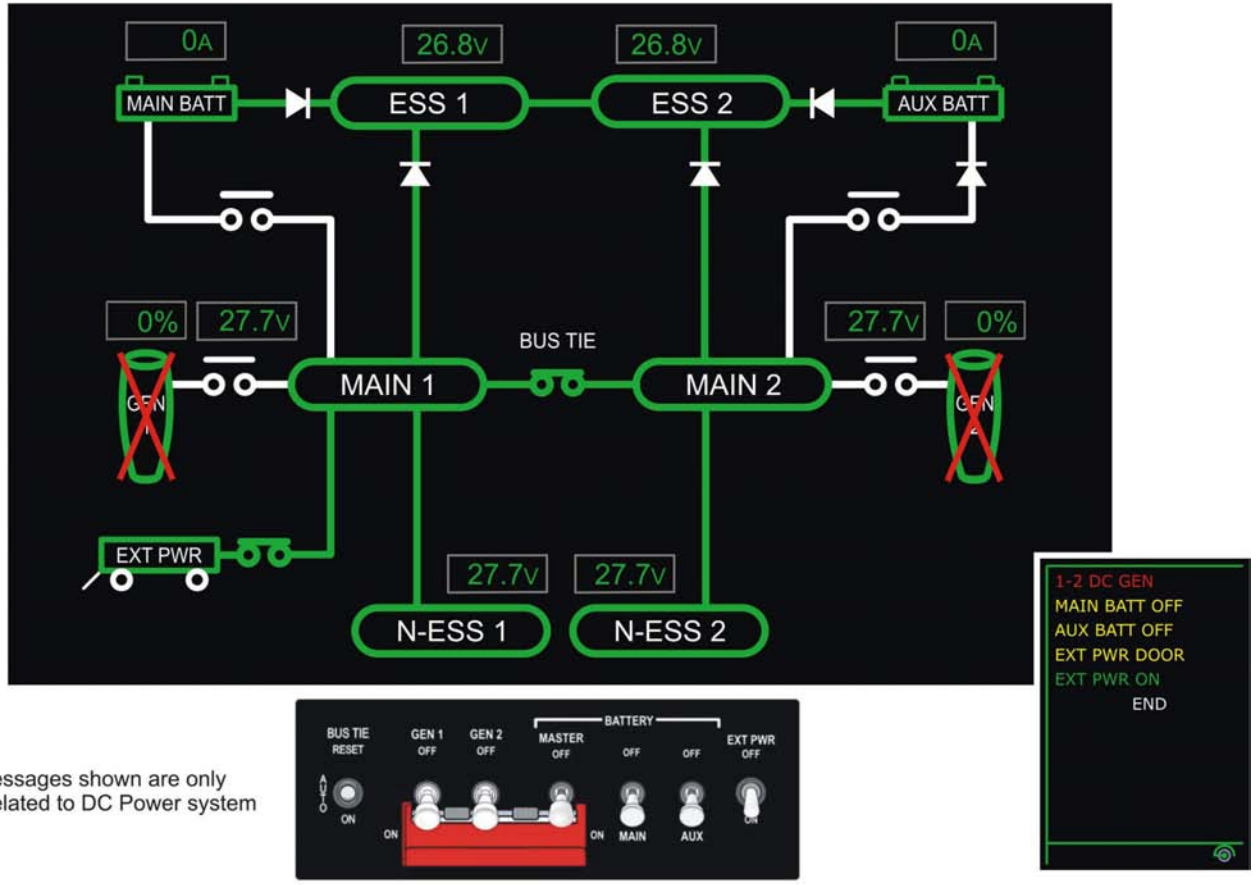




Note: CAS Messages shown are only those related to DC Power system

ICN-39-A-244000-G-A0126-00387-A-01-1

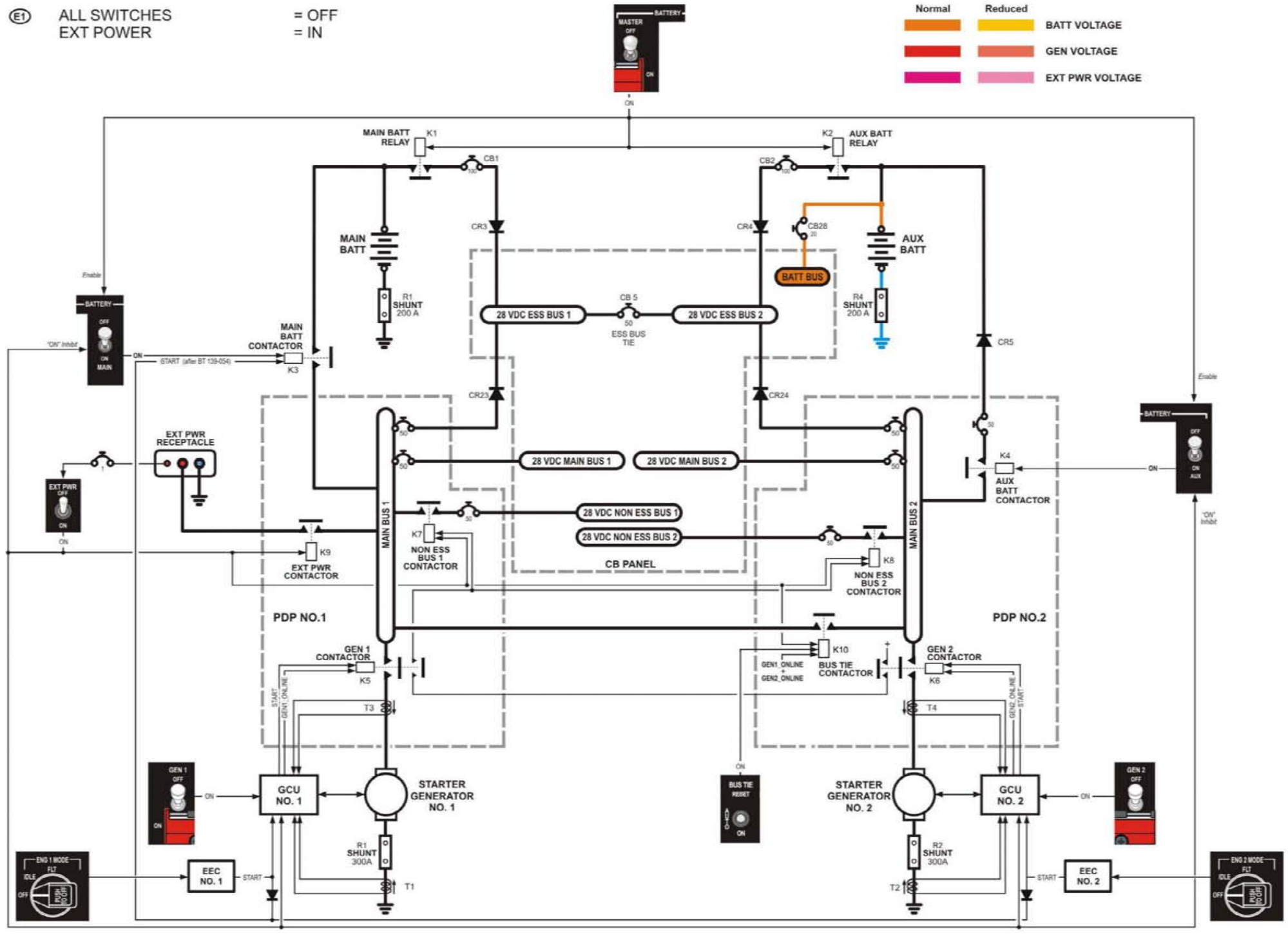
DC POWER SYNOPTIC – EXT PWR STARTING (ENGINE NO.2)



Note: CAS Messages shown are only those related to DC Power system

ICN-39-A-244000-G-A0126-00388-A-01-1

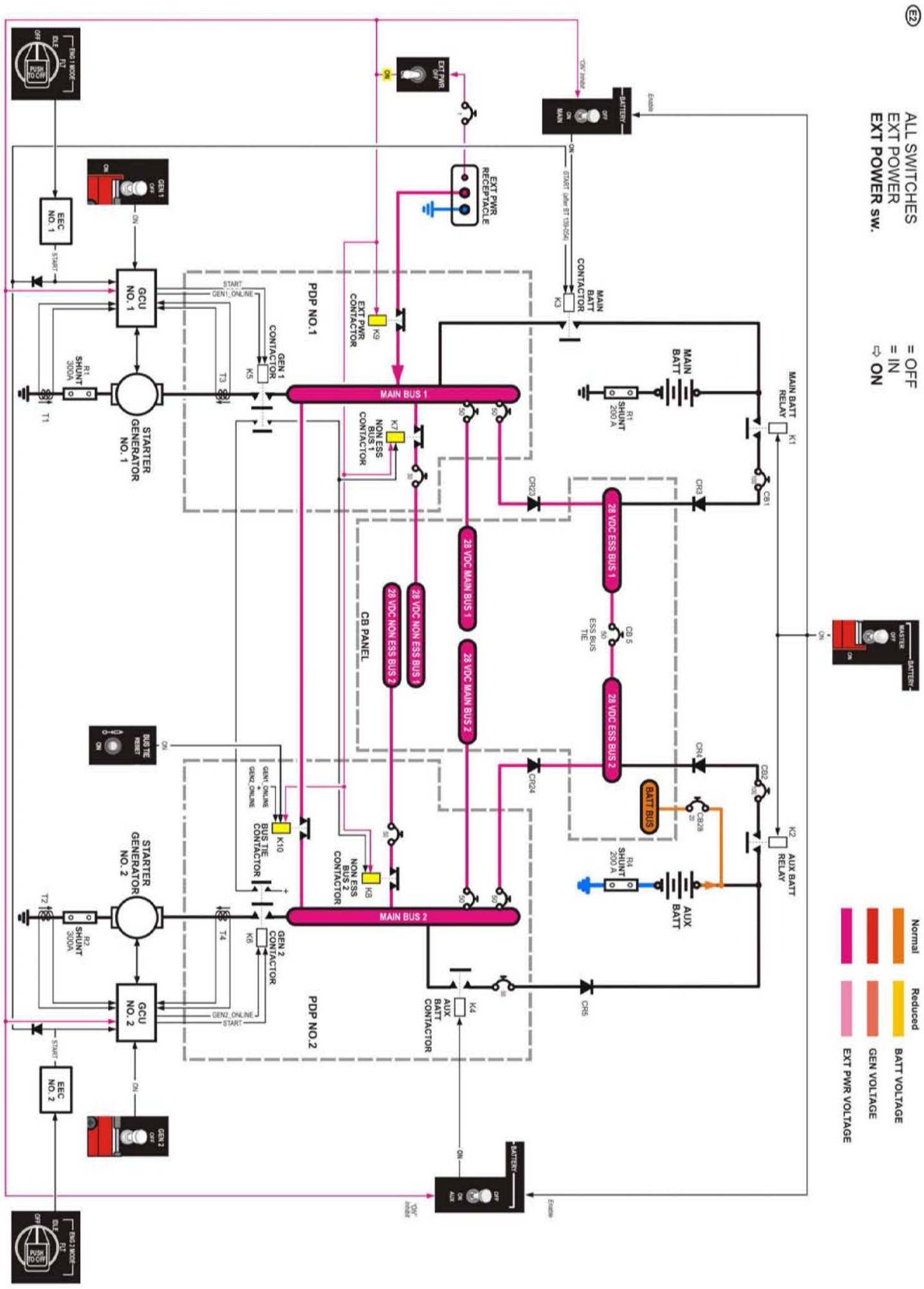
DC POWER SYNOPTIC – EXT PWR = ON, BOTH ENGINES RUNNING



ICN-39-A-244000-G-A0126-00376-A-01-1

STEP E1 – EXTERNAL PWR AVAILABLE AT THE RECEPTACLE

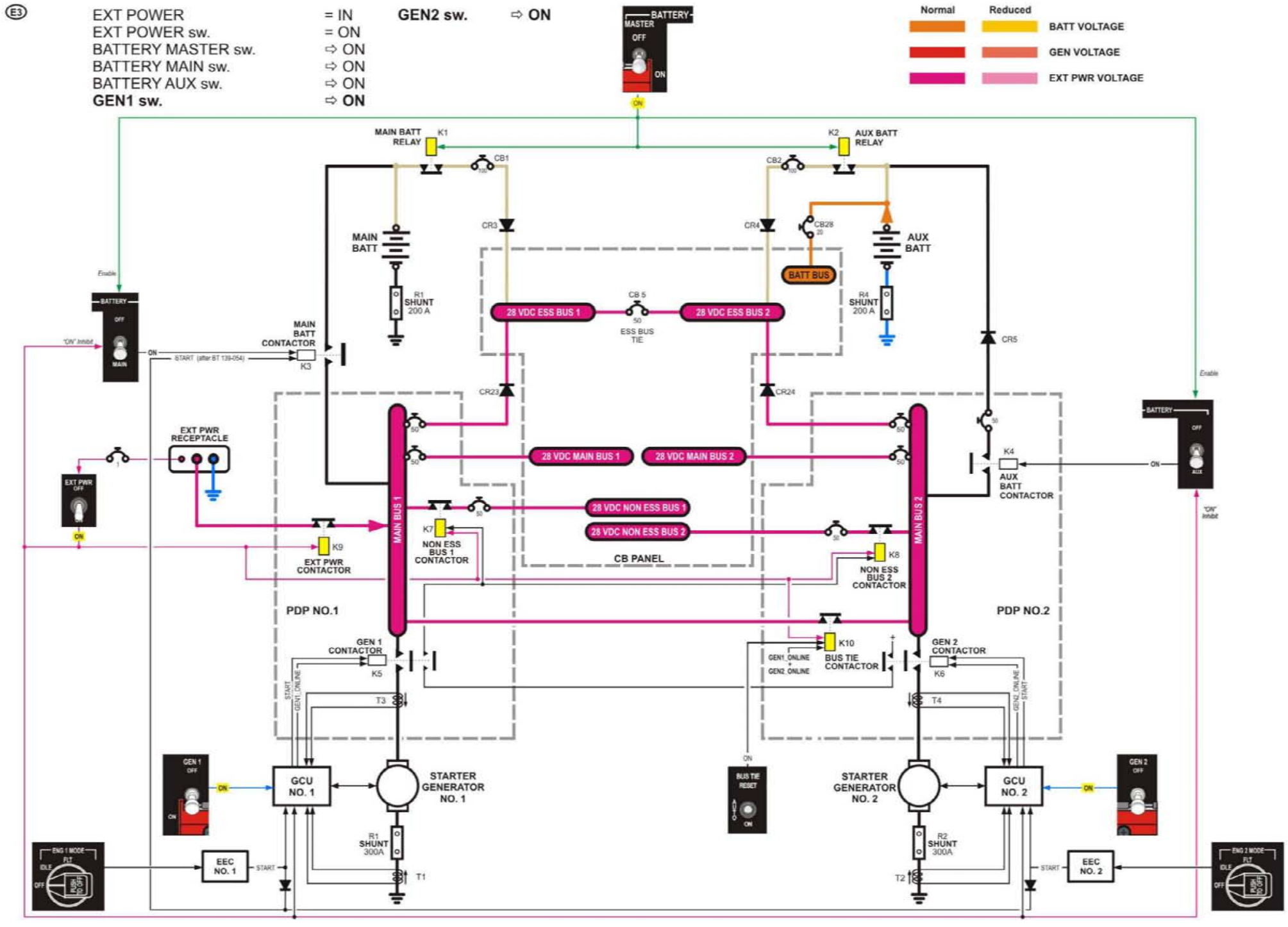
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STEP E2 – CONNECTION OF THE EXTERNAL POWER

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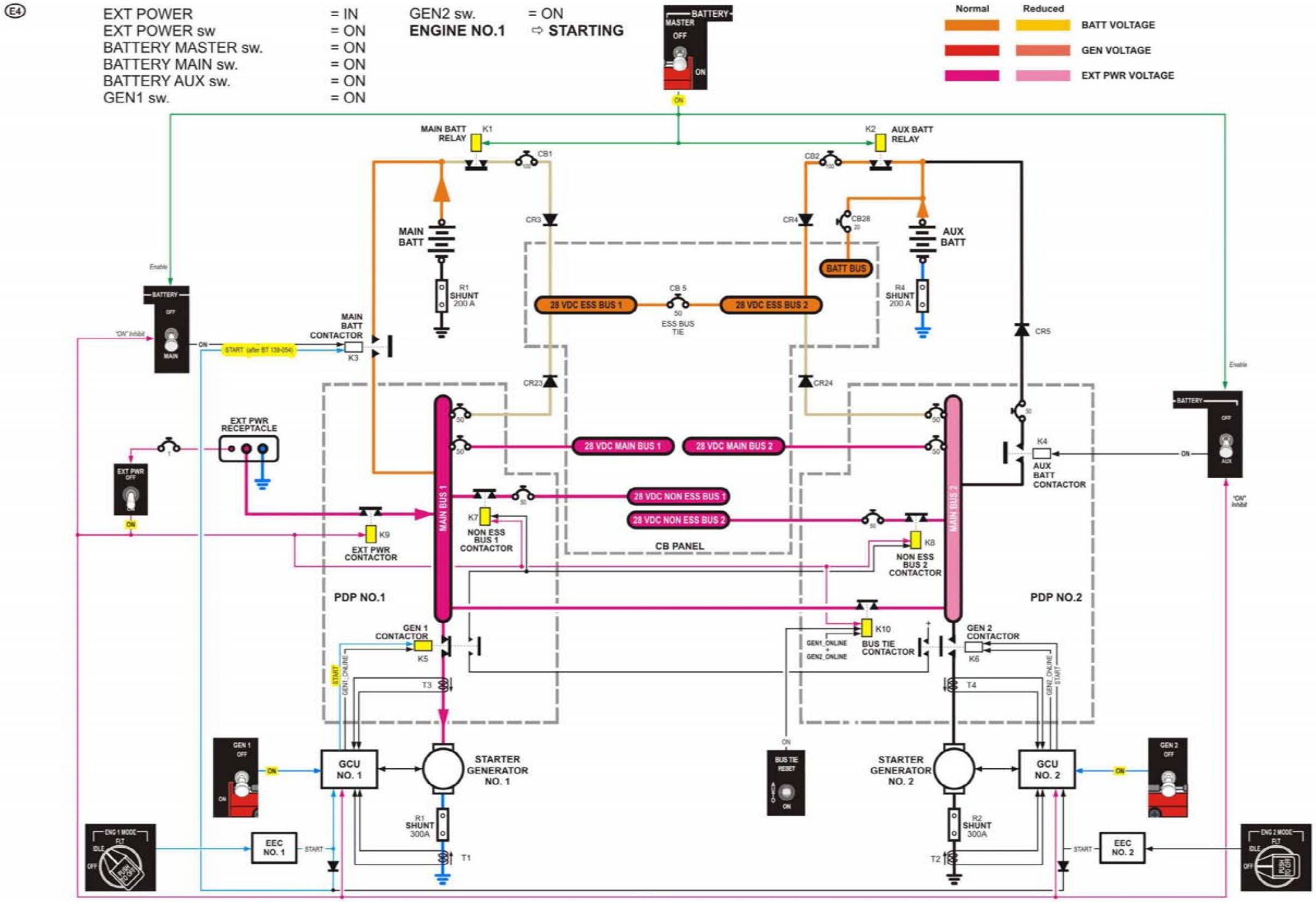
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STEP E3 – BATTERIES AND EXT PWR STARTING

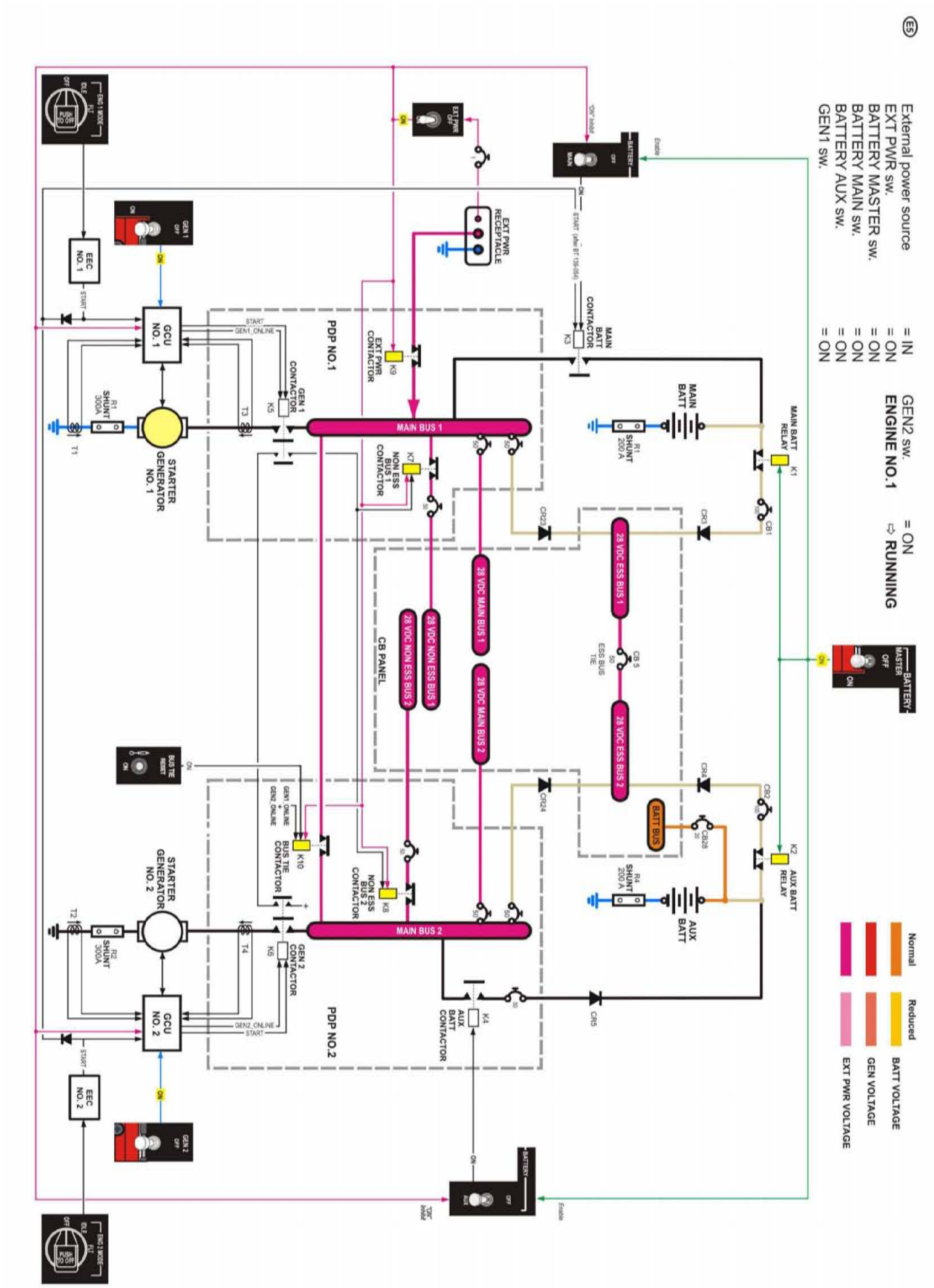
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ICN-39-A-244000-G-A0126-00379-A-01-1

STEP E4 – ENGINE 1 STARTING

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STEP E5 – ENGINE RUNNING

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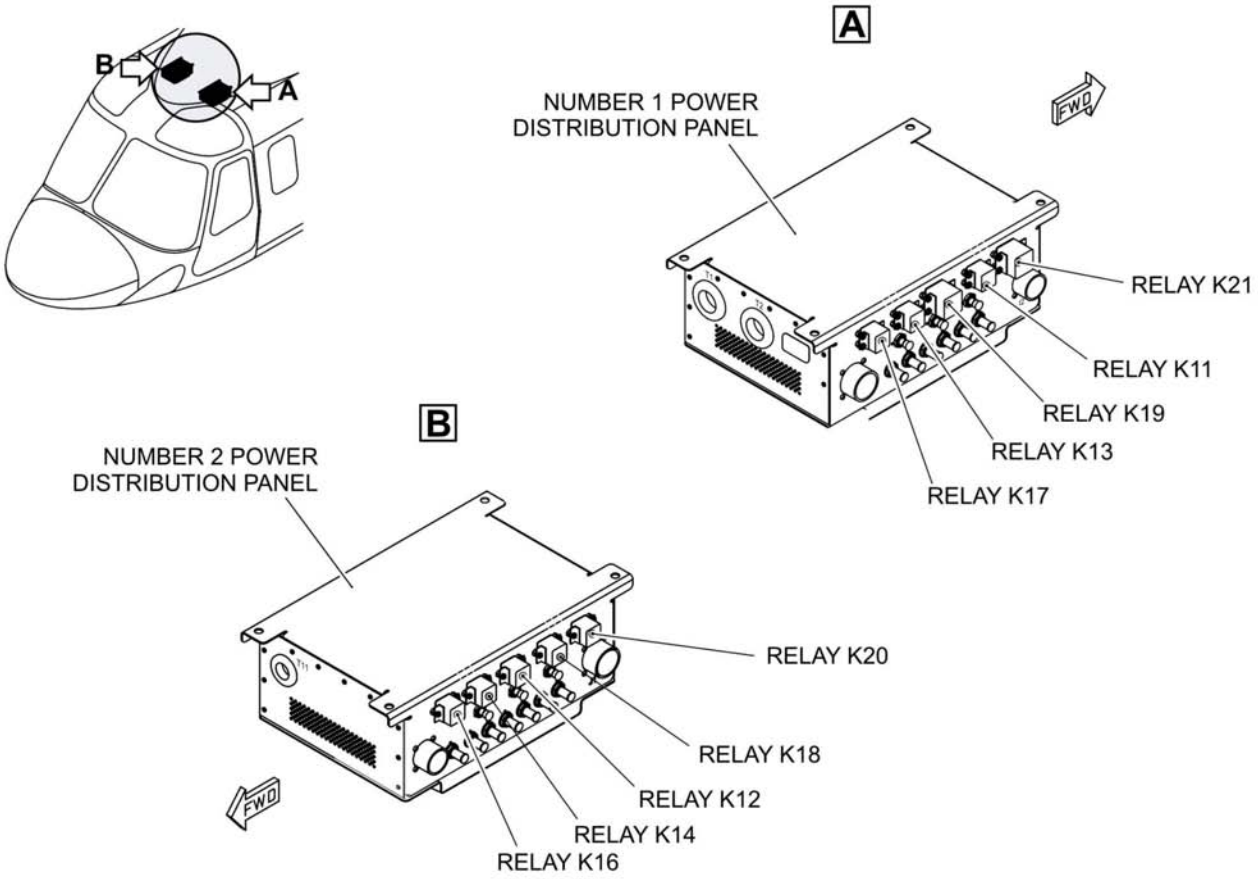
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SECTION 60 - DC ELECTRICAL LOAD DISTRIBUTION

DESCRIPTION

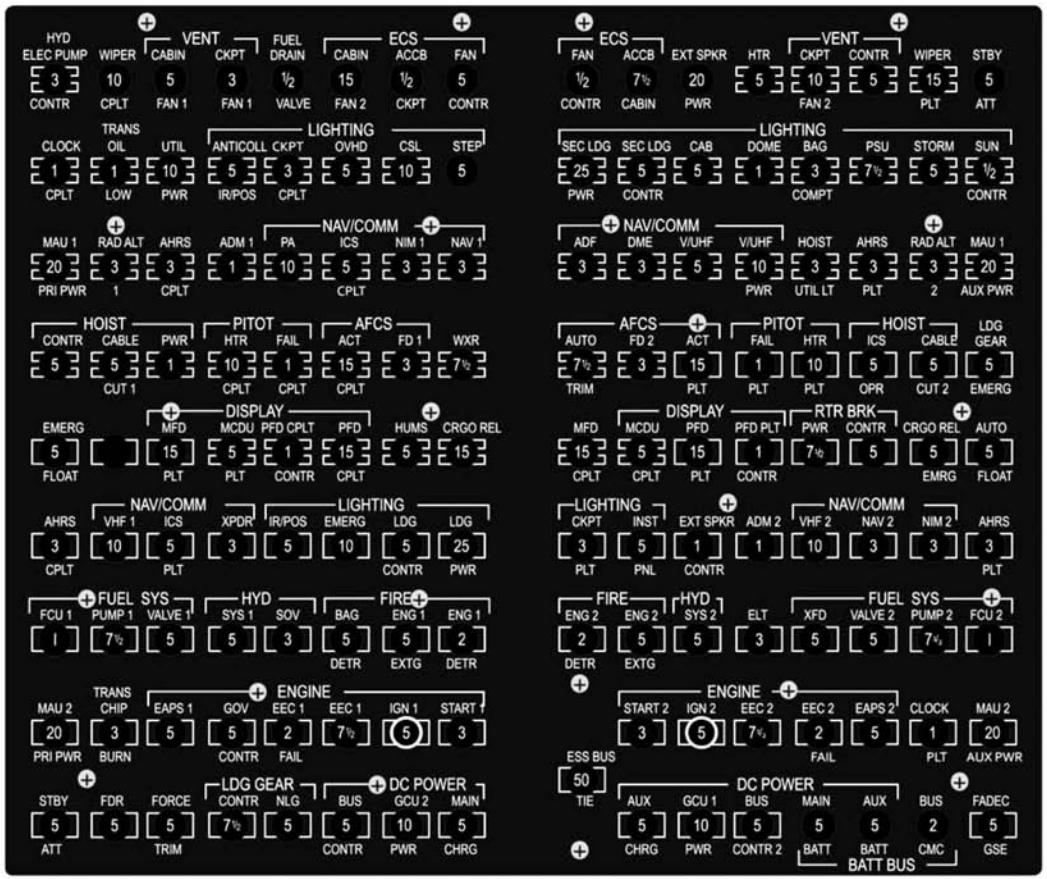
The DC electrical load distribution includes these primary components:

- Number 1 Power Distribution Panel (PDP1) (A3)
- Number 2 Power Distribution Panel (PDP2) (A4)
- Circuit breaker panel.



ICN-39-A-246000-G-A0126-00389-A-01-1

POWER DISTRIBUTION PANEL LOCATION



ICN-39-A-246000-G-A0126-02916-A-01-1

CIRCUIT BREAKERS PANEL

NUMBER 1 POWER DISTRIBUTION PANEL

The Number 1 power-distribution panel (A3) is mounted on the roof of the cabin, on the left side.
The PDP1 is an electronic box. It has an aluminium chassis and a cover. Ten screws attach the cover to the box body. The box also has two louvers, one on the front and one on the rear side. These louvers let the heat in the box be released, thus the internal temperature of the box does not become too high.
There are five openings on the front and rear sides of the box.
They have these labels: T1, T2, T8, T9 and T10.
They are used for the interface of the PDP1 with the power cables from the DC electrical power sources and the PDP2.
The PDP1 contains:

- The Number 1 DC current-transformer (T3) of the hot side of the output of the Number 1 generator
- The power relays of the Number 1 power-distribution system.

These power relays are:

- The Number 2 Generator Line Contactor (GCL2) (K5)
- The Number 1 contactor non-essential bus (K7)
- The external power contactor (K9)
- The Number 1 DC current-transformer.

The PDP1 also includes the power protection contactors and most protection devices.
Relay contactors and circuit breakers are installed on the front of the PDP1. Access to these relays and circuit breakers is easy. They let you monitor the status of the PDP1.
The relays are: K17, K13, K19, K11 and K21.
The circuit breakers are: CB3, CB7, CB9, CB13, CB15, CB17, CB43, CB47 and CB49.

NUMBER 2 POWER DISTRIBUTION PANEL

The installation and the construction of the Number 2 power-distribution panel (A4) is almost the same as that of the PDP1.
The PDP2 is mounted on the roof of the cabin, on the right side.
There are three openings on the left and right sides of the box: T1, T2 and T11.
They are used for the interface of the PDP2 with the power cables from:

- The DC electrical power sources
- The PDP1.

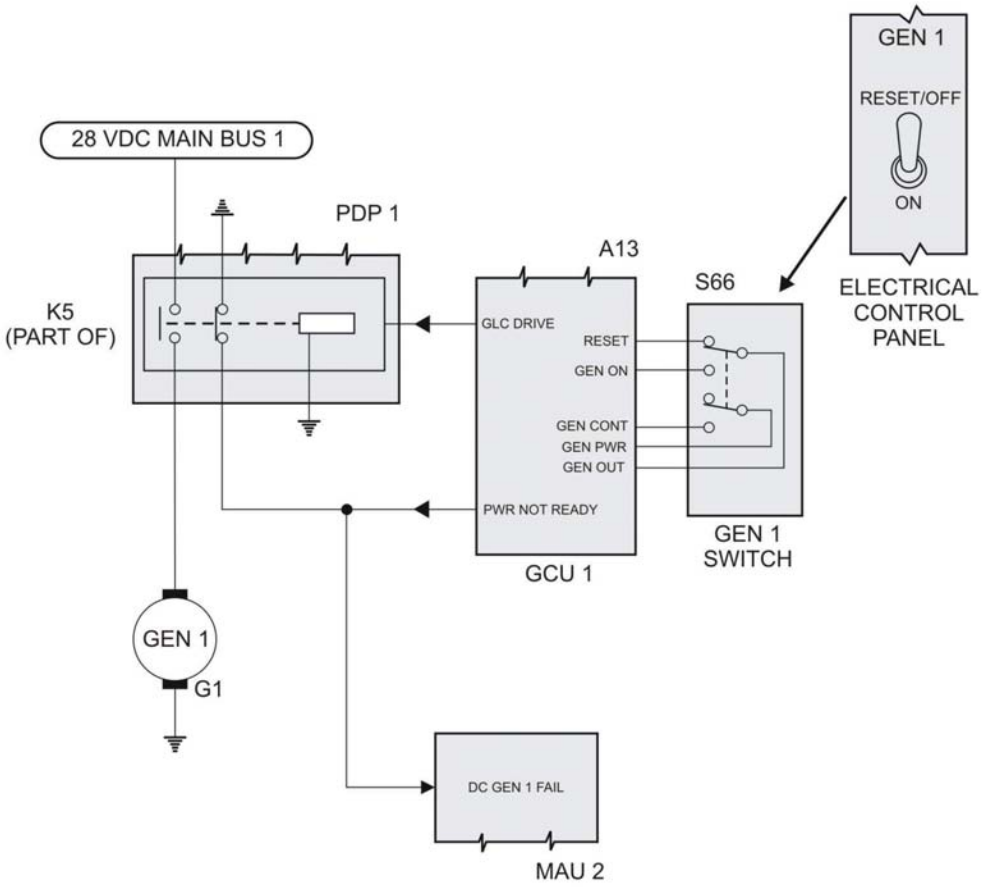
The PDP2 contains:

- The Number 2 DC current-transformer (T4) of the hot side of the output of the Number 2 generator
- The power relays of the Number 2 DC power-distribution system.

These power relays are:

- The Number 2 Generator Line Contactor (GCL2) (K6)
- The Number 2 contactor-non essential-bus (K8)
- The bus tie contactor (K10)
- The auxiliary battery contactor (K4)
- The Number 2 DC current-transformer.

The PDP2 also includes the power protection contactors and most protection devices.
Relay contactors and circuit breakers are installed on the front of the PDP2.
Access to these relays and circuit breakers is easy. They let you monitor the status of the PDP2.
The relays are: K12, K14, K16, K18 and K20.
The circuit breakers are: CB4, CB6, CB8, CB10, CB44, CB48 and CB50.



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POWER DISTRIBUTION PANEL SYSTEM SCHEMATIC

OPERATION

Number 1 Generator Line Contactor Relay

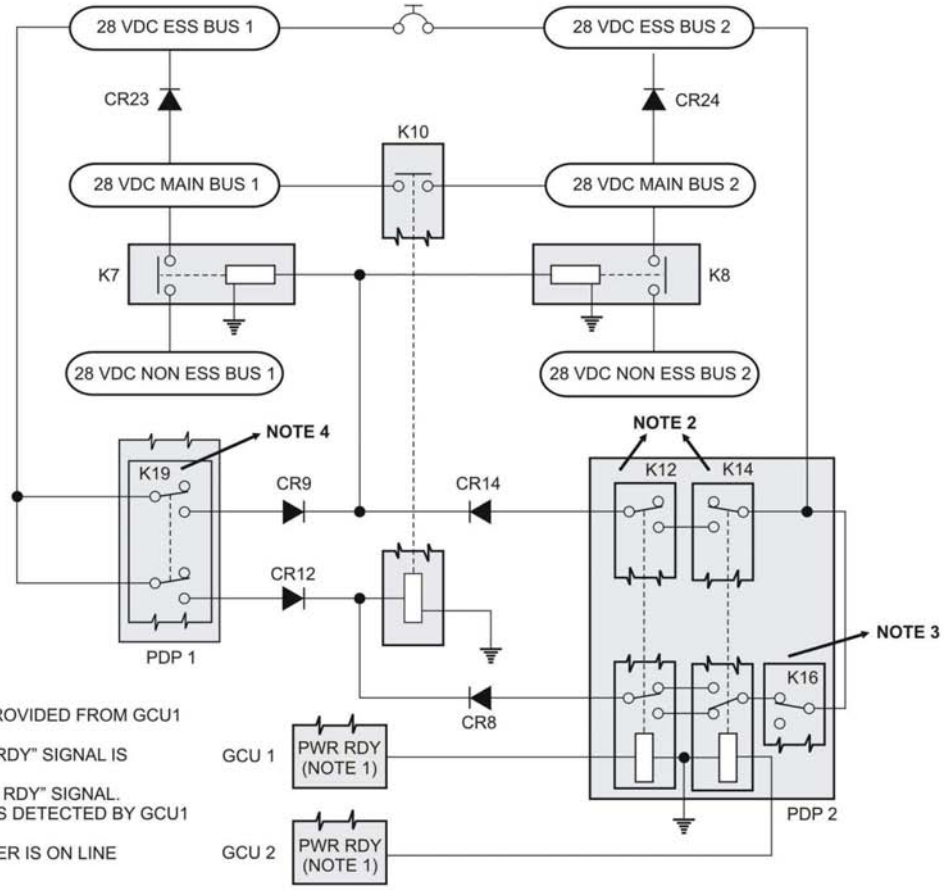
The Number 1 Generator Line Contactor-relay (GLC1) (K5) connects the generator (G1) to the Number 1 main bus (MAIN BUS1).
The relay K5 closes when energized by a drive signal supplied by the Number 1 Generator Control Unit (GCU1).

Number 2 Generator Line Contactor Relay

The Number 2 Generator Line Contactor-relay (GLC2) (K6) connects the generator (G2) to the Number 2 main bus (MAIN BUS2).
The relay K6 closes when energized by a drive signal supplied by the Number 2 Generator Control Unit (GCU2).

K7, K8 And Bus-Tie Contactor Relays

The relays K7, K8 and the Bus Tie Contactor relay (BTC) (K10) connect the main busses to the non essential busses and the two main busses to each other.



NOTE:

1. THE "POWER READY" 28 VDC OUTPUT SIGNAL IS PROVIDED FROM GCU1 AND GCU2.
   IF DC GENERATOR IS TAKEN "OFF-LINE" THE "PWR RDY" SIGNAL IS REMOVED.
2. K12 AND K14 ARE ENERGIZED THROUGH THE "PWR RDY" SIGNAL.
3. K16 OPENS WHEN AN OVERCURRENT CONDITION IS DETECTED BY GCU1 OR GCU2.
4. K19 CONTACTS CLOSE WHEN THE EXTERNAL POWER IS ON LINE (SEE 24-41-00-00A-042A-A).

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OPERATION – SCHEMATIC

Number 1 And Number 2 Generator Line Contactor Relays Logic Management

When the relays GLC1 (K5) and GLC2 (K6) are closed, they connect the generators to the related main busses.
The GCU1 and GCU2 transmit an output signal (GLC drive). This signal controls the 28 V dc power supply to the coil of relay K5 and to the coil of relay K6 as follows:

- The battery starts the Number 1 engine: K5 closed and K6 open
- The battery starts the Number 2 engine: K5 open and K6 closed
- The battery and the Number 1 generator start the Number 2 engine: K5 closed and K6 closed
- The battery and the Number 2 generator start the Number 1 engine: K5 closed and K6 closed
- The external power source starts the Number 1 engine: K5 closed and K6 open
- The external power source starts the Number 2 engine: K5 open and K6 closed
- Normal generation mode: K5 closed and K6 closed.

The relays K5 and K6 have many auxiliary contacts. One of them is used to monitor the conditions of the primary contact. It transmits a caution signal to the MAUs when the generator is not on – line.
The same caution signal is transmitted to the MAUs when the GCU stops the generator because of a malfunction.

Relays K7, K8 And Bus Tie Contactor Relay Management

When the relays K7, K8 and bus-tie contactor relay (K10) are energized (refer to Fig 2), they connect:

- The main busses to the non essential busses
- The two main busses to each other.

NORMAL OPERATION WITH BOTH GENERATORS ON-LINE

During normal operation, the GCU1 transmits a 28 V dc PWR RDY signal to the relay K12 installed in the PDP2.
The GCU2 transmits the same signal to the relay K14 installed in the PDP2.
When the two relays are energized, and the Number 2 essential bus is connected to the coils of relay K7 and K8 (through CR14):
The relays K7 and K8 become energized
The non essential busses are connected to the main busses.
If the GCU2 is not connected, the PWR RDY signal is removed from the relay K14. When power is removed from K14 the circuit of K7 and K8 is open.
If the GCU1 is not connected, the PWR RDY signal is removed from the relay K14. When power is removed from K14 the circuit of K7 and K8 is open.
When relays K12 and K14 are energized, the coil of relay K10 does not receive the 28 V dc power from the essential bus. The relay K10 is not energized and the main bus bars are disconnected from each other.
If the GCU1 or the GCU2 is not connected, the related K12 or K14 relay is not energized. 28 V dc power from the Number 2 essential bus is supplied to the coil of the relay K10. This supply is through the contact of relay K16, that is usually closed, and through the diode CR8. The relay K10 connects the main busses to each other.
If the GCU1 or GCU2 senses an overcurrent condition (interpole voltage current than 550 A) the contact of relay K16 opens. The essential bus is disconnected from the relay K10 that opens and disconnects the main busses from each other.

EXTERNAL POWER SOURCE ON-LINE

The relay K19 is energized and its contacts are closed when:

- The external power source is connected to the helicopter
- The EXT PWR switch is at ON.

As a result, the Number 1 essential bus supplies power to the coils of relays K7, K8 and K10 through the diodes CR9 and CR12.
When the relays K7, K8 and K10 become energized:

- The Number 1 non-essential bus is connected to the Number 1 main bus
- The Number 2 non essential bus is connected to the Number 2 main bus
- The main busses are connected to each other.

DC ESS BUS 1	DC ESS BUS 2
BAGGAGE FIRE DETECT	ADM 2
BUS CONTROL 1	AUTO FLOAT CONTROL
EAPS 1	AUX BATT CHARGE
EEC 1	BUS CONTROL 2
EEC 1 FAIL INDICATION	CARGO HOOK EMERG RELEASE
EMERG LIGHT	CLOCK PLT
EMERG FLOAT CONTROL (MANUAL)	COCKPIT PLT LIGHT
EMERG FLOAT POWER	EAPS 2
ENG 1 FIRE DETECTION	EEC 2
ENG 1 FIRE EXTINGUISHER	EEC 2 FAIL INDICATION
ENG 1 IGNITER & START	ELT
FCU 1	ENG 2 FIRE DETECTION
FDR / CVR	ENG 2 FIRE EXTINGUISHER
FUEL PUMP 1	ENG 2 IGNITER 6 START
FUEL SHUT OFF VALVE 1	EXT SPKR CONTROL
FORCE TRIM	FADEC GSE
GCU 2	FCU 2
ENG GOV CONTROL	FUEL PUMP 2
HYD SYS 1 & SOV 1 & 2	FUEL SHUT OFF VALVE 2
ICS PLT (PLT ICS IN BKUP MODE)	FUEL XFEED
LANDING LIGHT CONTROL	GCU 1
LANDING LIGHT POWER	HOIST CUTTER 2
LDG GEAR CONTROL	HOIST ICS OPERATOR
LDG GEAR INDICATION	HYD SYS 2
MAIN BATT CHARGE	INSTRUMENT PANEL ILLUM.
MAU 2 (PRI POWER)	LDG GEAR EMERG DOWN
MRC 1 (VHF 1)	LINEAR ACTUATOR 2
MRC 2 (XPDR)	MAU 2 (AUX POWER)
NLG STEERING LOCK	MRC 2 (NIM 2, VHF 2, NAV 2)
PLT MFD	PDF PLT CONTROL
STBY 1 ATT	PITOT 2 FAIL INDICATION
TAIL POS LIGHT	PITOT 2 HEAT
TRANS CHIP BURNER	PLT PFD
	RTR BRK

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SERVICES AVAILABLE ON ESSENTIAL BUS 1 AND 2

DC MAIN BUS 1

ADM 1
ANTI-COLLISION LIGHT
CARGO HOOK RELEASE
CLOCK CPLT
COCKPIT CPLT LIGHT
CPLT ICS
(CPLT ICS in back up mode)
CPLT PFD
CSL ILLUM
Fd1
HOIST POWER
HOIST CUTTER 1
HOIST CONTROL
HUMS
HYD ELEC PUMP
LINEAR ACTUATOR 1
MAU 1 (PRI POWER)
MCDU PLT
MRC 1 (NIM 1, NAV 1)
OVHD PANEL ILLUM
PA
PFD CPLT CONTROL
PITOT 1 FAIL INDICATION
PITOT HEAT 1
RAD ALT 1
UTIL POWER
W / RADAR
XMSN OIL LEVEL SENSOR

DC MAIN BUS 2

AUTO TRIM
BAGGAGE COMPT LIGHT
CABIN LIGHT
COCKPIT / CABIN HEATER
COCKPIT VENT (PLT)
CPLT MFD
DOME LIGHT
FD 2
HOIST LIGHT
MAU 1 (AUX POWER)
MCDU CPLT
MRC 2 (ADF & DME)
PLT W / WIPER
PSU
RAD ALT 2
SEARCH LIGHT CONTROL
SEARCH LIGHT POWER
STORM LIGHT
SUN LIGHT CONTROL
VENT CONTROL 2
V / UHF

DC NON ESS BUS 1

CABIN VENT
COCKPIT VENT (CPLT)
CPLT W / WIPER
ECS (COCKPIT)
FUEL DRAIN VALVE
STEP LIGHT

DC NON ESS BUS 1

ECS (CABIN)
EXT / SPKR POWER

ICN-39-A-246000-G-A0126-00394-A-01-1

SERVICES AVAILABLE ON MAIN BUS 1 AND 2

REF-DES	CB LABEL	BUS BAR	W/D SHT
CB3	STBY 1 ATT	ESS 1	6
CB4	LIGHTING STEP	NON ESS 1	1
CB5	LIGHTING POSN	ESS 1	1
CB6	LIGHTING ANTI COLL	MAIN 1	1
CB7	LIGHTING LDG PWR	ESS 1	6
CB8	LIGHTING SEC LDG CONTR	MAIN 2	3
CB9	LIGHTING LDG CONTR	ESS 1	6
CB10	LIGHTING SEC LDG PWR	MAIN 2	3
CB11	ENGINE EEC 1	ESS 1	6
CB12	FADEC GSE	ESS 2	4
CB13	ENGINE EEC 2	ESS 2	12
CB14	ENGINE EEC 2 FAIL	ESS 2	12
CB15	ENGINE GOV CONTR	ESS 1	6
CB16	ENGINE EEC 1 FAIL	ESS 1	6
CB17			
CB18	NAV/COMM NAV 2	ESS 2	13
CB19	NAV/COMM ADF	MAIN 2	4
CB20	NAV/COMM XPDR	ESS 1	13
CB21	NAV/COMM DME	MAIN 2	4
CB22	NAV/COMM NIM 2	ESS 2	12
CB23	NAV/COMM NAV 1	MAIN 1	1
CB24	NAV/COMM NIM 1	MAIN 1	1
CB25			
CB26			
CB29			
CB30	SUN CONTR	GND	20
CB31	LIGHTING PSU	MAIN 2	3
CB32			
CB33	LIGHTING BAG COMPT	MAIN 2	3
CB34	LIGHTING INST PNL	ESS 2	14
CB35	LIGHTING CSL	MAIN 1	16
CB36	LIGHTING OVHD	MAIN 1	16
CB37	LIGHTING DOME	MAIN 2	3
CB38	ADM 2	ESS 2	14
CB39	ADM 1	MAIN 1	2
CB40	NAV/COMM VHF 2	ESS 2	13
CB41	NAV/COMM VHF 1	ESS 1	1
CB42	LIGHTING CKPT PLT	ESS 2	11
CB43	LIGHTING CKPT CPLT	MAIN 1	11
CB44	NAV/COMM ICS PLT	ESS 1	14
CB46	ECS FAN CONTR	GND	19
CB47	ECS FAN CONTR	GND	19
CB48	AHRS PLT	ESS 2	13
CB49	AHRS CPLT	ESS 1	1
CB50	AHRS PLT	MAIN 2	14
CB51	AHRS CPLT	MAIN 1	2
CB52	ECS ACCB CABIN	NON ESS 2	19

REF-DES	CB LABEL	BUS BAR	W/D SHT
CB53	ECS ACCB CKPT	NON ESS 1	19
CB54	AFCS ACT PLT	ESS 2	5
CB55	AFCS ACT CPLT	MAIN 1	2
CB56	ECS CABIN FAN 2	NON ESS 1	19
CB57	LIGHTING CAB	MAIN 2	21
CB58			
CB59	HOIST PWR	MAIN 1	19
CB60	AFCS FD 1	MAIN 1	2
CB61	AFCS AUTO TRIM	MAIN 2	5
CB62	HOIST CONTR	MAIN 1	19
CB63	AFCS FD 2	MAIN 2	3
CB64	FORCE TRIM	ESS 1	7
CB65	HYD SYS 1	ESS 1	7
CB66	HYD SYS 2	ESS 2	14
CB67	HYD ELEC PUMP CONTR	MAIN 1	5
CB68	HYD SOV	ESS 1	7
CB69	HUMS	MAIN 1	21
CB70	BATT BUS MAIN BATT	BATT	17
CB71	DC PWR AUX CHRG	ESS 2	14
CB72	NAV/COMM ICS PLT	MAIN 1	2
CB73	CLOCK CPLT	MAIN 1	1
CB74	BATT BUS AUX BATT	BATT	17
CB75	DC POWER MAIN CHRG	ESS 1	8
CB76	CLOCK PLT	ESS 2	12
CB77	TRANS OIL LOW	MAIN 1	7
CB78	DISPLAY PFD PLT	ESS 2	12
CB79	DISPLAY PFD PLT CONTR	ESS 2	12
CB80	DISPLAY MFD	ESS 1	1
CB81	DISPLAY PFD CPLT	MAIN 1	1
CB82	DISPLAY PFD CPLT CONTR	MAIN 1	2
CB83	DC POWER BUS CONTR 1	ESS 1	8
CB84			
CB85	TRANS CHIP BURN	ESS 1	7
CB86	RAD ALT 1	MAIN 1	2
CB87	HOIST CABLE CUT 1	MAIN 1	19
CB88	HOIST CABLE CUT 2	ESS 2	20
CB89	WIPER PLT	MAIN 2	21
CB90	FUEL SYS VALVE 1	ESS 1	8
CB91	FUEL SYS PUMP 1	ESS 1	8
CB92	FUEL SYS PUMP 2	ESS 2	15
CB93	FUEL SYS VALVE 2	ESS 2	15
CB94	FUEL SYS XFD	ESS 2	15
CB95	FUEL SYS FCU 2	ESS 2	15
CB96	FUEL SYS FCU 1	ESS 1	8
CB97	FUEL DRAIN VALVE	NON ESS 1	16
CB98	FIRE BAG DET	ESS 1	20
CB99	FDR	ESS 1	6

ICN-39-A-246000-G-A0126-00395-A-01-1

CIRCUIT BREAKERS LIST (SHEET 1 OF 2)

REF-DES	CB LABEL	BUS BAR	W/D SHT
CB100	RTR BK PWR	ESS 2	15
CB101	RTR BK CONTR	ESS 2	14
CB102	WIPER CPLT	NON ESS 1	21
CB103	ENGINE START 1	ESS 1	6
CB104	ENGINE START 2	ESS 2	12
CB105	DISPLAY MCDU CPLT	MAIN 2	4
CB106	NAV/COMM	MAIN 1	21
CB107	ENGINE IGN 1	ESS 1	6
CB108	ENGINE IGN 2	ESS 2	12
CB109	MAU 1 PRI PWR	MAIN 1	10
CB110	MAU 2 PRI PWR	ESS 1	9
CB111	ENGINE EAPS 1	ESS 1	20
CB112	ENGINE EAPS 2	ESS 2	20
CB113			
CB114	CARGO REL EMERG	ESS 2	20
CB115	CARGO REL	MAIN 1	19
CB116	WXR	MAIN 1	21
CB117	MAU 1 AUX PWR	MAIN 2	5
CB118	MAU 2 AUX PWR	ESS 2	15
CB119	FIRE ENG 1 DETR	ESS 1	9
CB120	FIRE ENG 2 DETR	ESS 2	15
CB121	FIRE ENG 2 EXTG	ESS 2	9
CB122	FIRE ENG 1 EXTG	ESS 1	15
CB125	HTR	MAIN 2	16
CB126	LDG GEAR CONTR	ESS 1	9
CB127	LDG GEAR EMERG	ESS 2	9
CB128	DC POWER GCU 1 PWR	ESS 2	9
CB129	DC POWER GCU 2 PWR	ESS 1	14
CB130	DC PWR BUS CONTR 2	ESS 2	14
CB131	VENT CONTR	MAIN 2	5
CB132	DISPLAY MCDU PLT	MAIN 1	2
CB133	BATT BUS MAU CMC	BATT	17
CB134	LDG GEAR MLG	ESS 1	9
CB135	VENT CABIN FAN 1	NON ESS 1	16
CB136	PITOT FAIL PLT	ESS 2	12
CB137	PITOT FAIL CPLT	MAIN 1	2
CB138	PITOT HTR PLT	ESS 2	18
CB139	PITOT HTR CPLT	MAIN 1	18
CB140			
CB141	EMERG FLOAT	ESS 1	20
CB142	AUTO FLOAT	ESS 2	20
CB143	RAD ALT 2	MAIN 2	21
CB144	VENT CKPT FAN 1	NON ESS 1	16
CB145	VENT CKPT FAN 2	MAIN 2	5
CB147	LIGHTING STORM	MAIN 2	21
CB148	UTIL PWR	MAIN 1	21
CB150	LIGHTING EMERG	ESS 1	20

REF-DES	CB LABEL	BUS BAR	W/D SHT
CB155	STBY 2 ATT	MAIN 2	4
CB156	HOIST UTIL LT	MAIN 2	4
CB157	HOIST ICS	ESS 2	20
CB158	DISPLAY MFD CPLT	MAIN 2	4

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CIRCUIT BREAKERS LIST (SHEET 2 OF 2)

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CHAPTER

25

EQUIPMENT/FURNISHING

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SECTION 00 - EQUIPMENT/FURNISHINGS

GENERAL DESCRIPTION

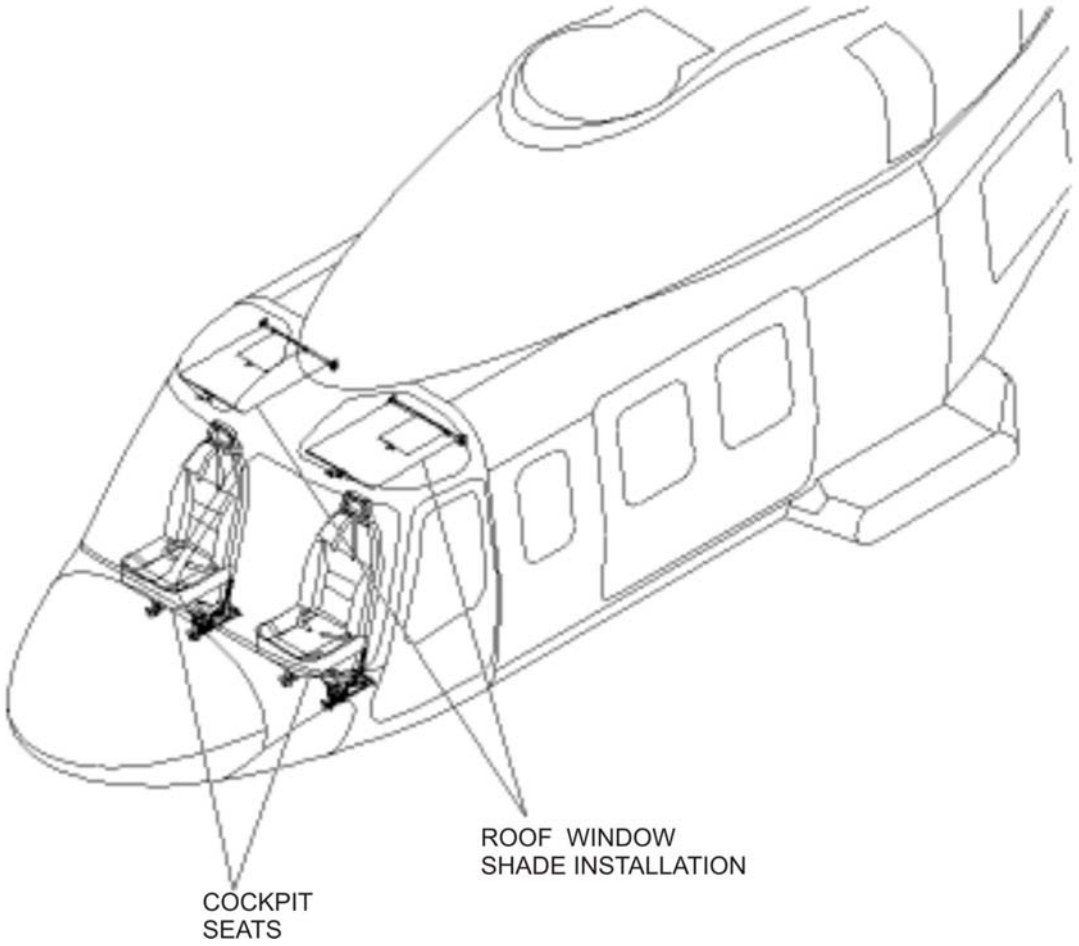
The equipment/furnishings system includes the sub-systems that follow:

- The flight compartment;
- The passenger compartment;
- The cargo compartment;
- The emergency;
- The cabin furnishings (optional);
- The insulation and lining;
- The aerial delivery (optional).

FLIGHT COMPARTMENT

Flight compartment is composed by:

- The cockpit seats;
- The roof window shade installation.



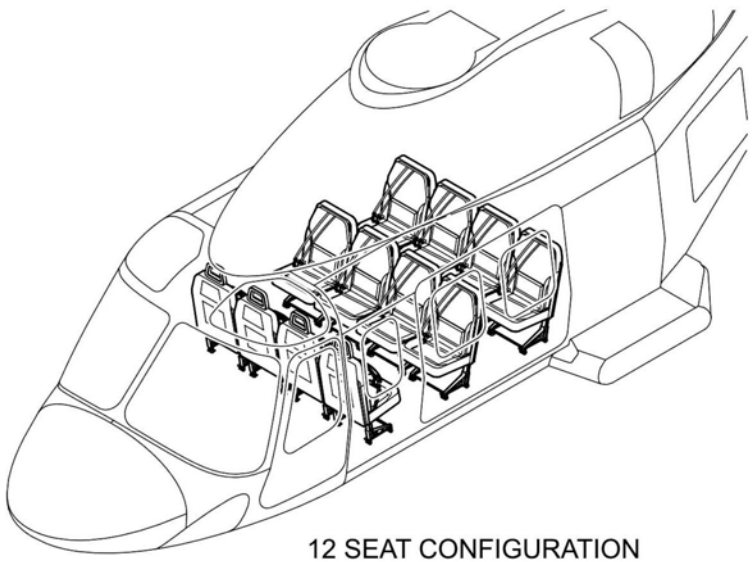
ICN-39-A-250000-G-A0126-00014-A-01-1

COCKPIT SEATS AND ROOF WINDOW INSTALLATION

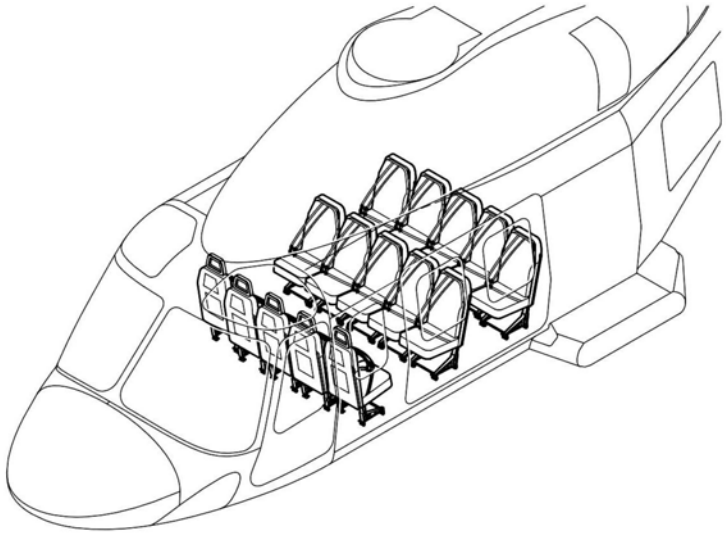
PASSENGER COMPARTMENT

The passenger compartment can have two configuration, 12 seat or 15 seat.
Both configuration have the same primary components:

- The forward cabin seats;
- The aft cabin seats;
- The safety belts;
- The headrests.



12 SEAT CONFIGURATION



15 SEAT CONFIGURATION

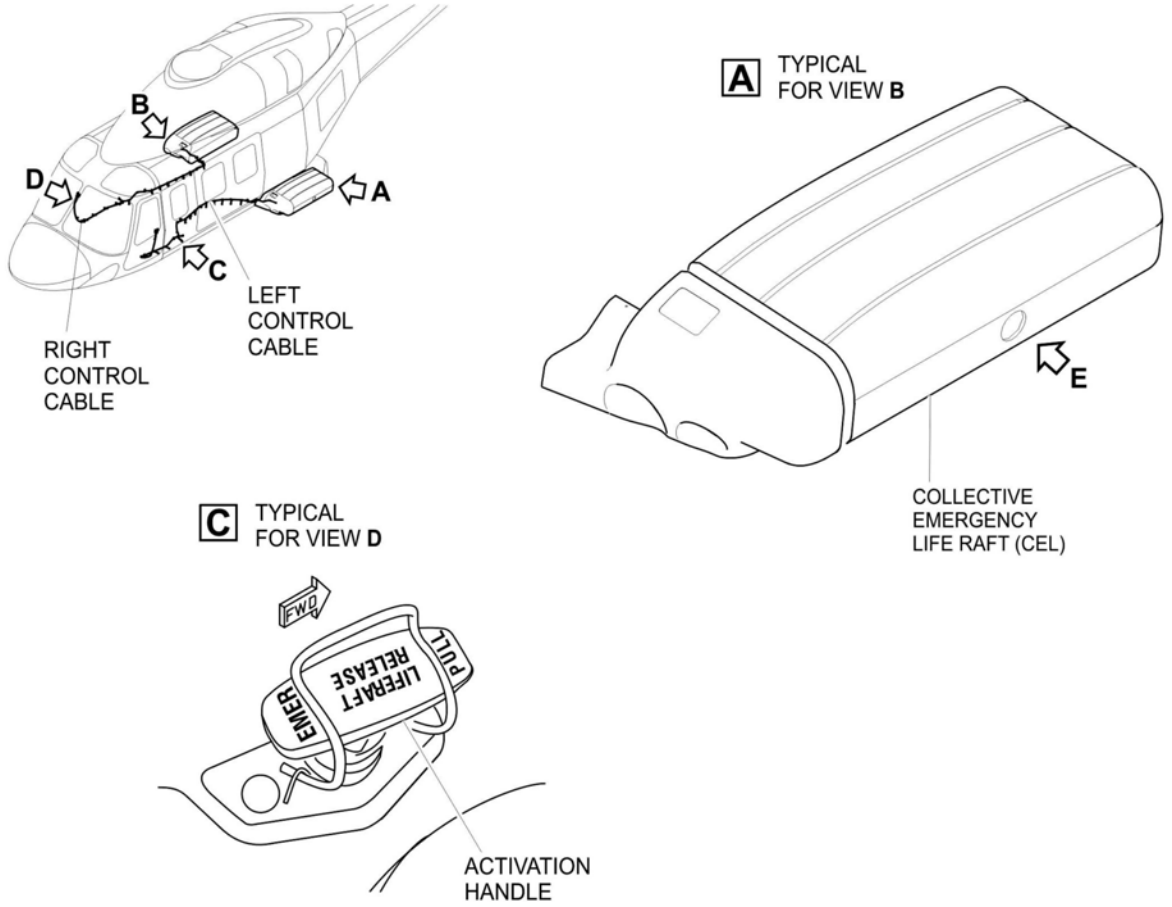
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PASSENGER COMPARTMENT CONFIGURATION

EMERGENCY

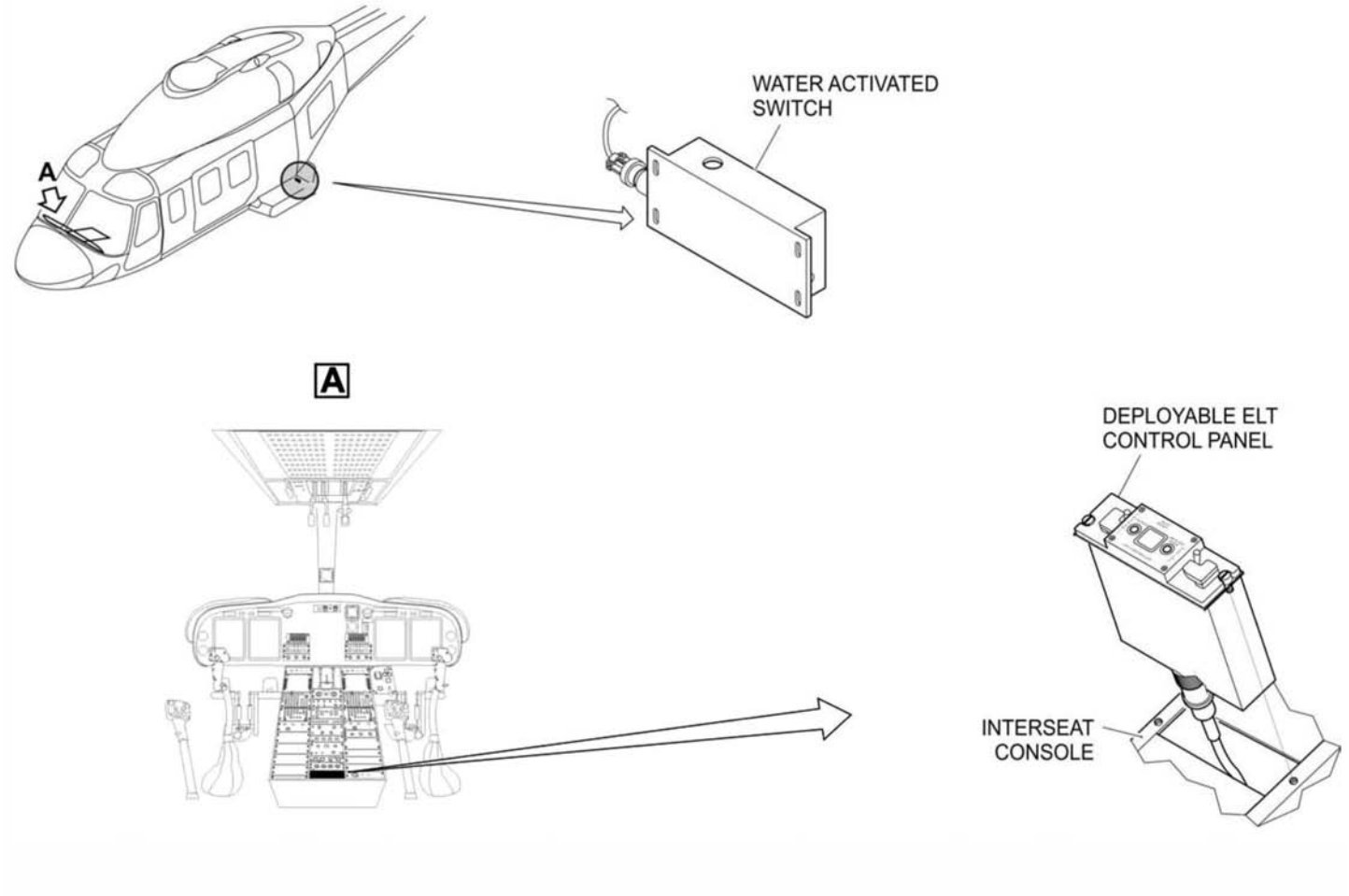
The emergency equipment is composed by:

- The emergency locator transmitter (ELT) system (optional);
- The life raft installation (optional);
- The first aid kit (optional);
- The deployable emergency locator transmitter (ELT) system (optional);
- The stand-alone emergency locator transmitter (ELT) installation (optional);
- The flashlight installation;
- The life jacket bag installation (optional).



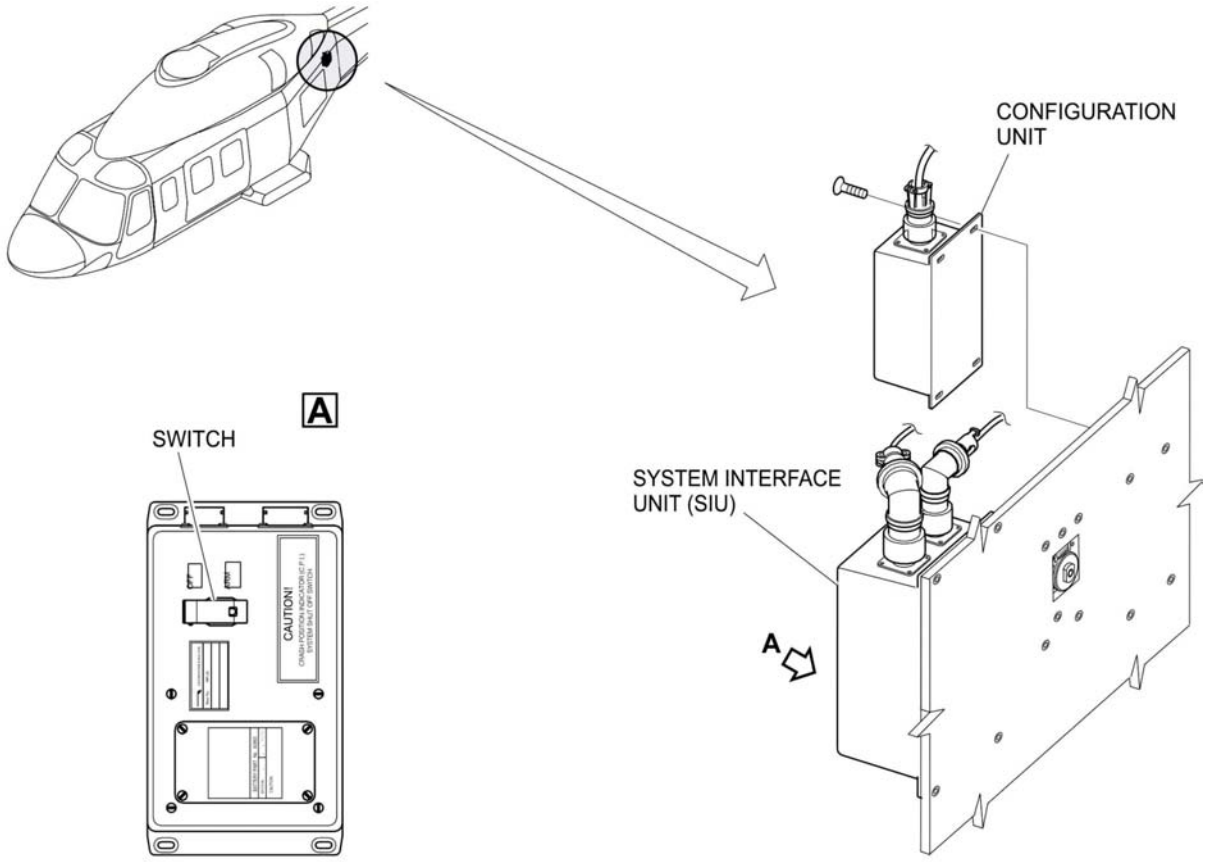
ICN-39-A-250000-G-A0126-00016-A-01-1

LIFE RAFT INSTALLATION



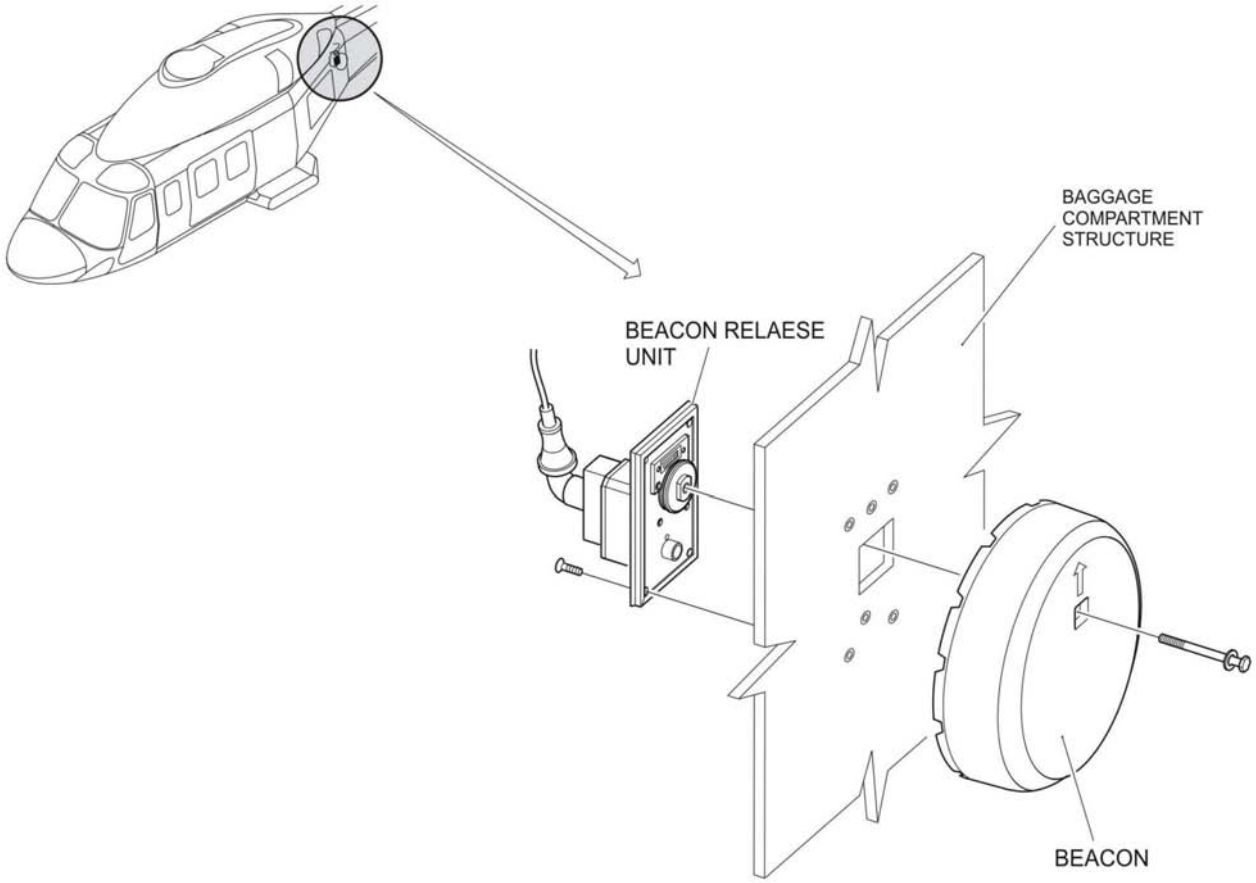
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EMERGENCY LOCATOR TRANSMITTER SYSTEM (SHEET 1 OF 3)



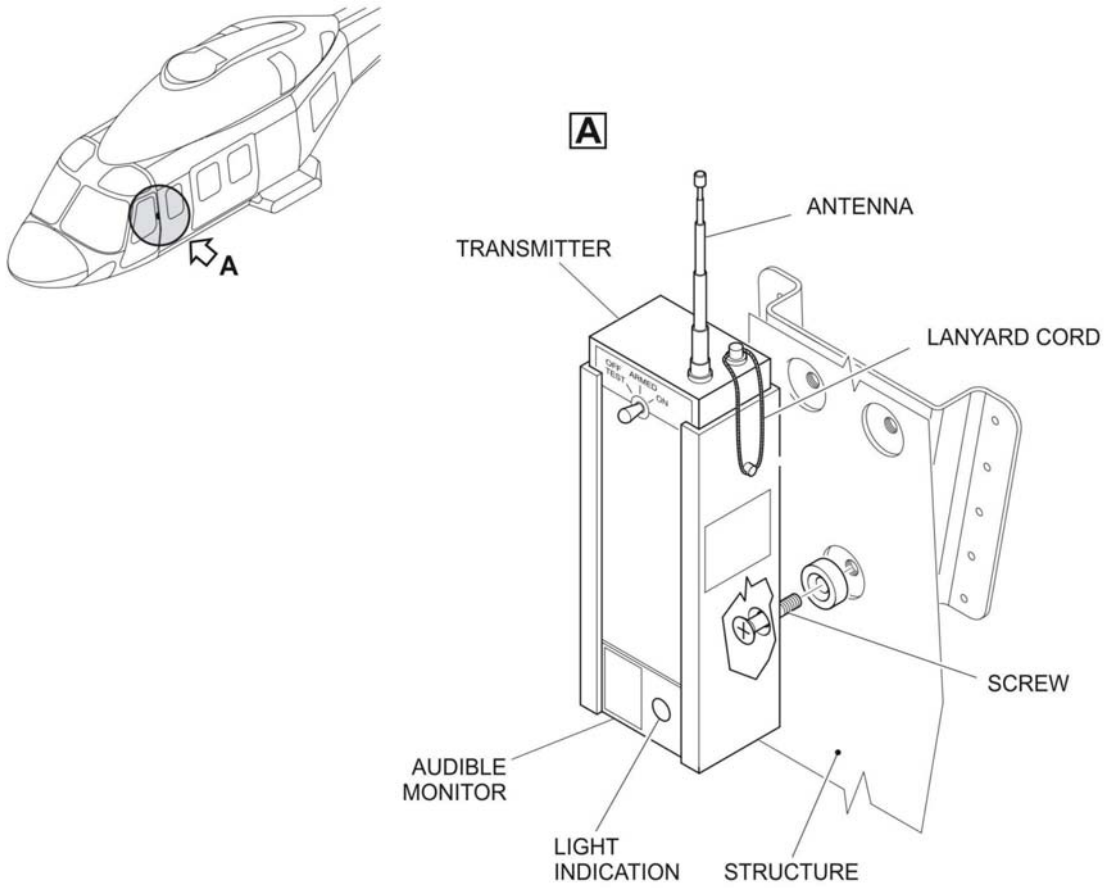
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EMERGENCY LOCATOR TRANSMITTER SYSTEM (SHEET 2 OF 3)



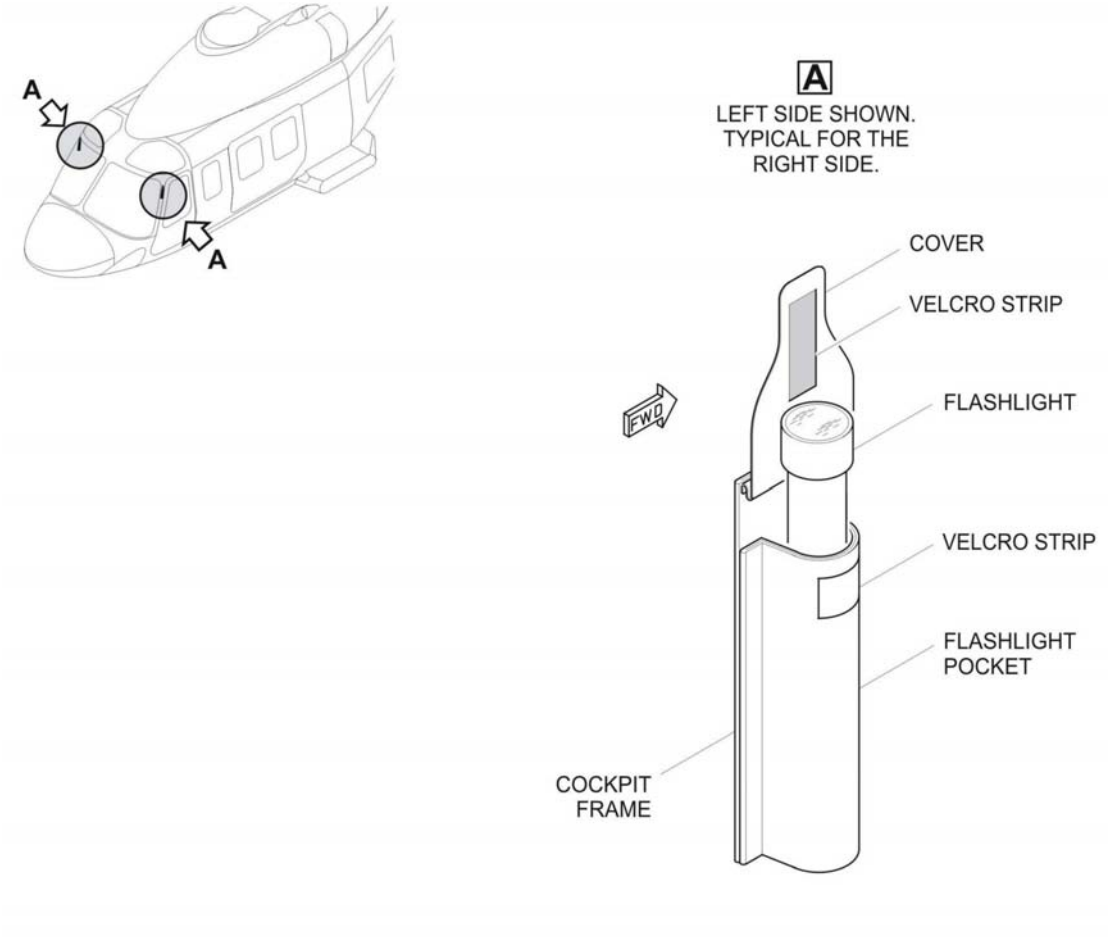
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EMERGENCY LOCATOR TRANSMITTER SYSTEM (SHEET 3 OF 3)



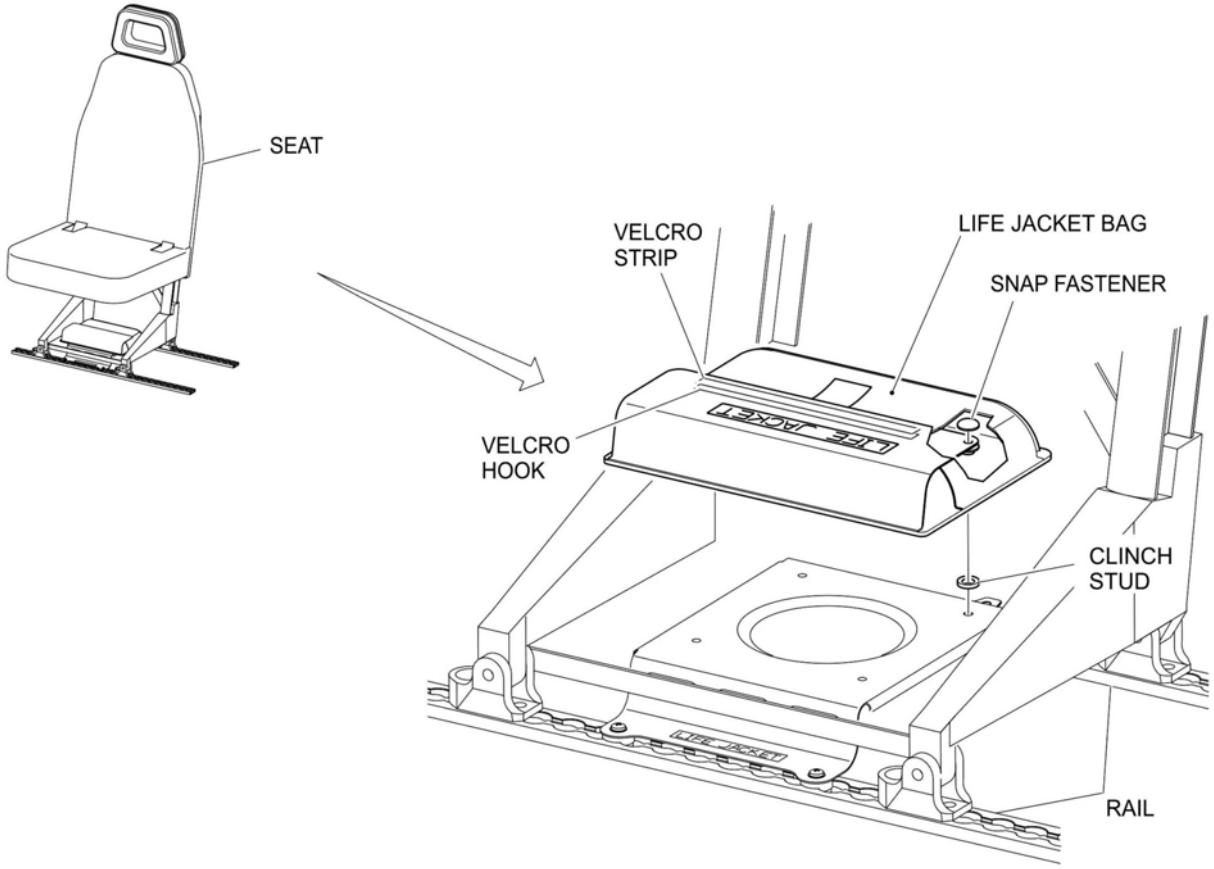
ICN-39-A-250000-G-A0126-00020-A-01-1

DEPLOYABLE EMERGENCY LOCATOR TRANSMITTER (OPTIONAL)



ICN-39-A-250000-G-A0126-00021-A-01-1

FLASHLIGHT INSTALLATION



ICN-39-A-250000-G-A0126-00022-A-01-1

LIFE JACKET BAG INSTALLATION

CABIN FURNISHING

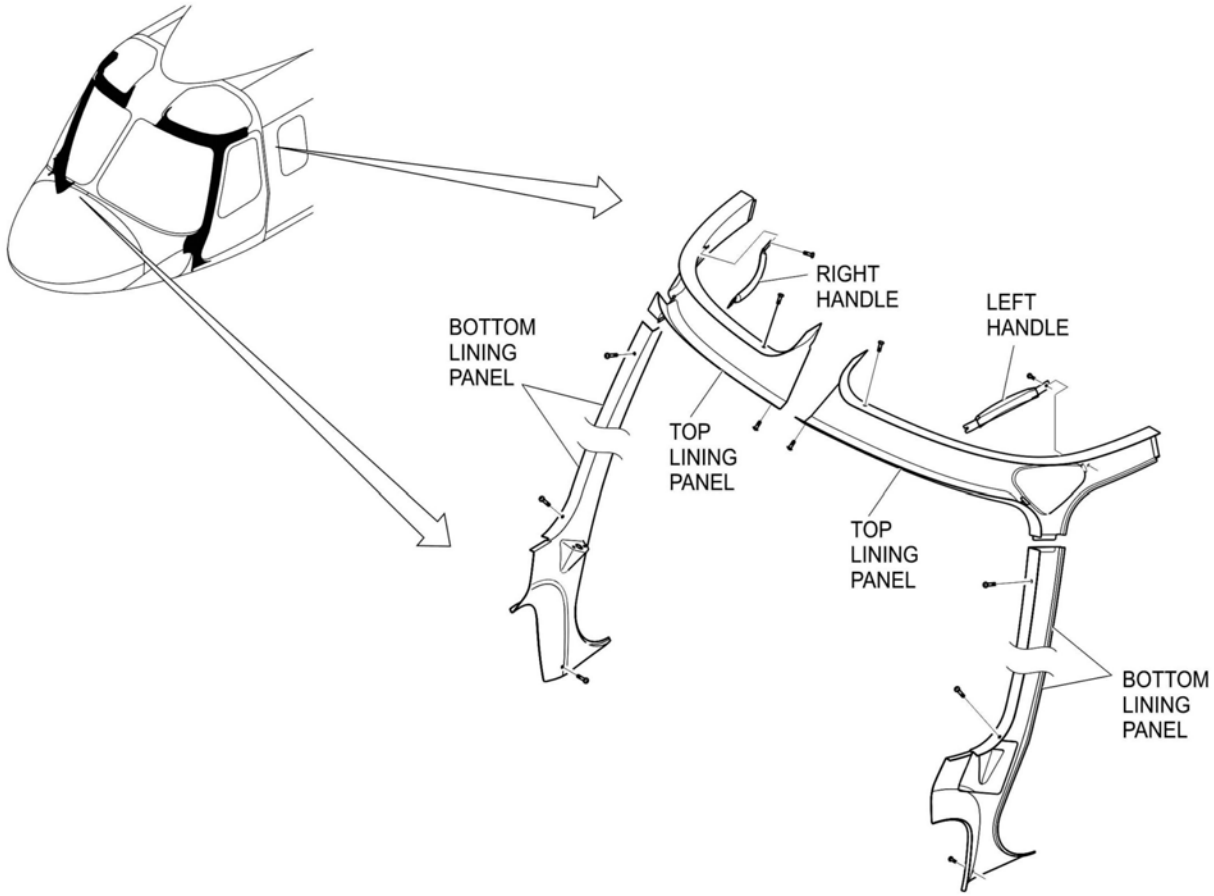
The cabin furnishings can have three configurations:

- The VIP cabin furnishing installation;
- The VVIP cabin furnishing installation.

INSULATION AND LINING

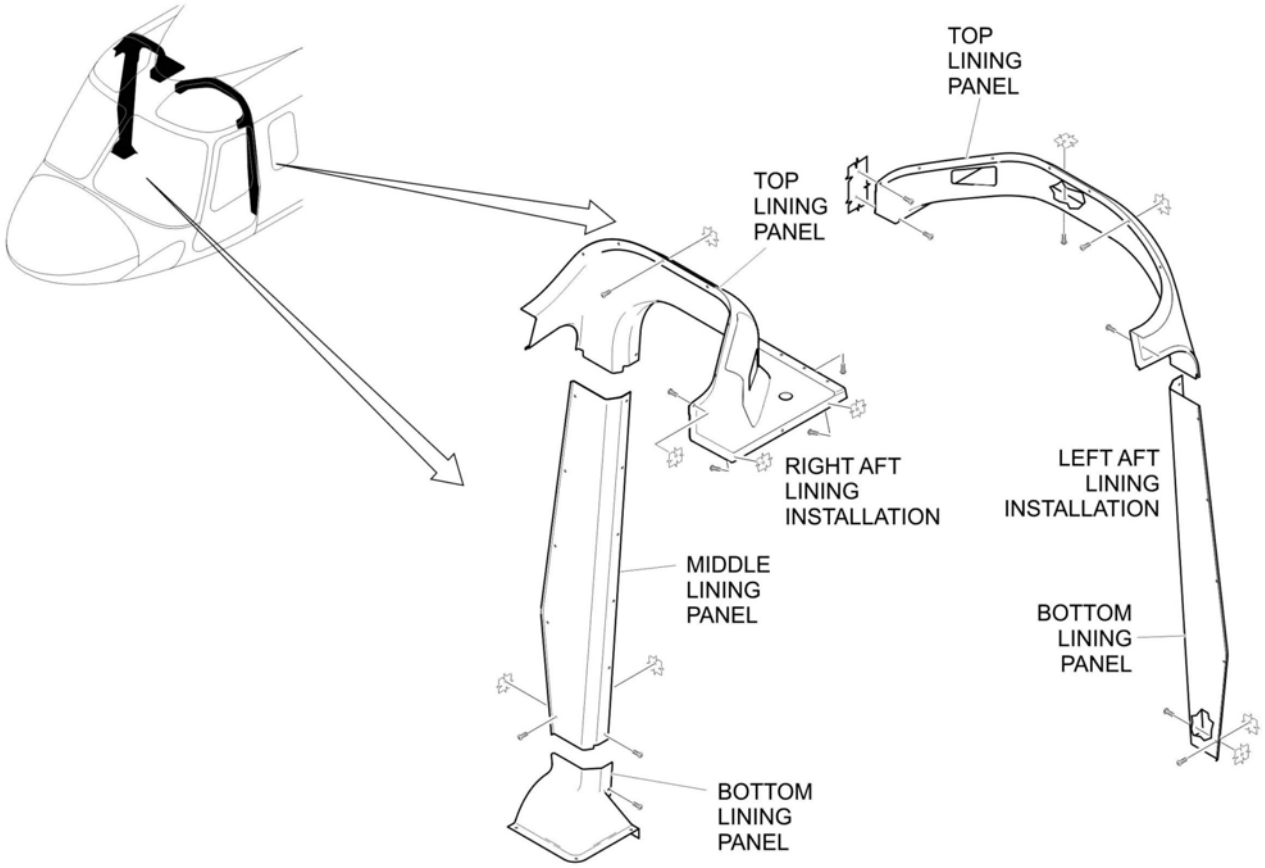
The lining is installed in three parts of the helicopter:

- In the cockpit;
- In the cabin;
- In the aft fuselage.



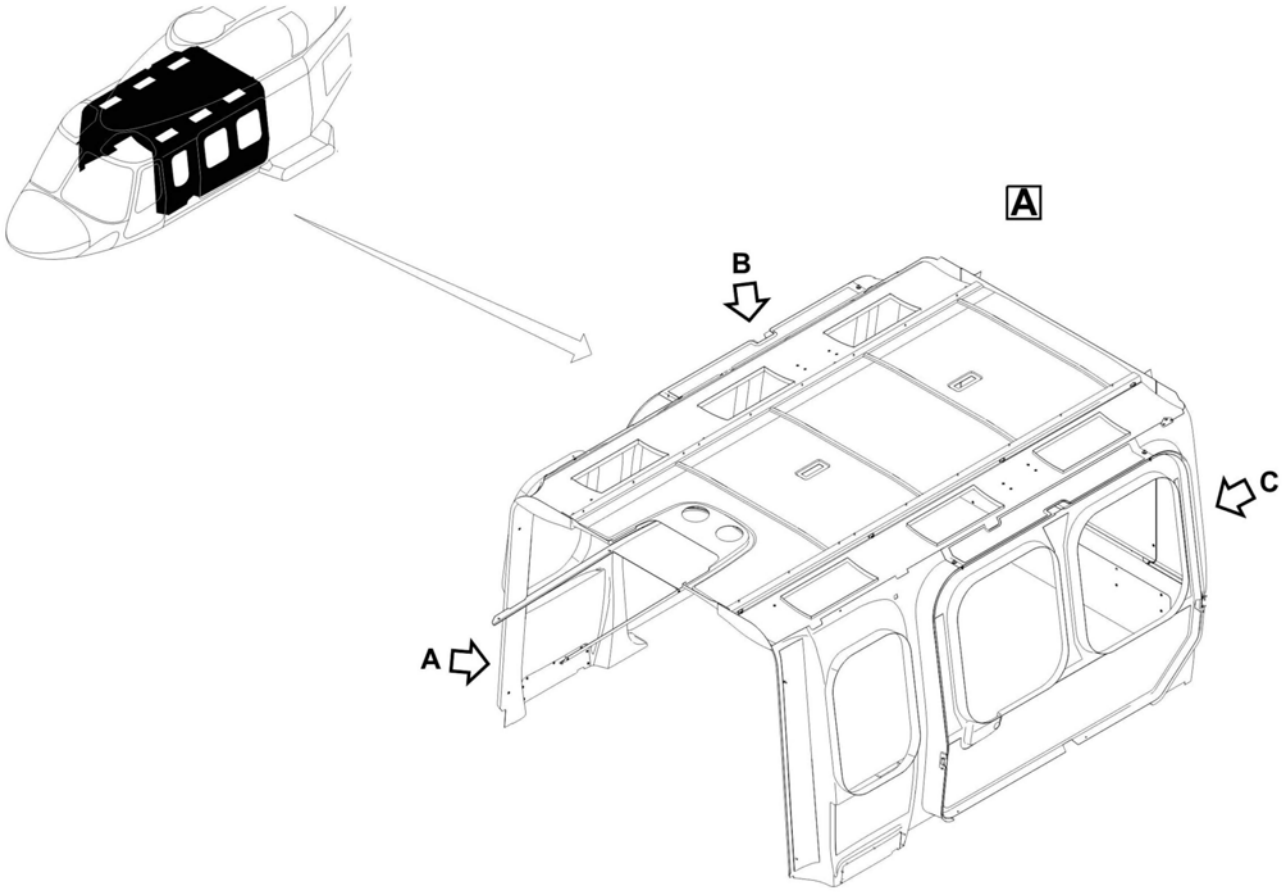
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COCKPIT LINING (SHEET 1 OF 2)



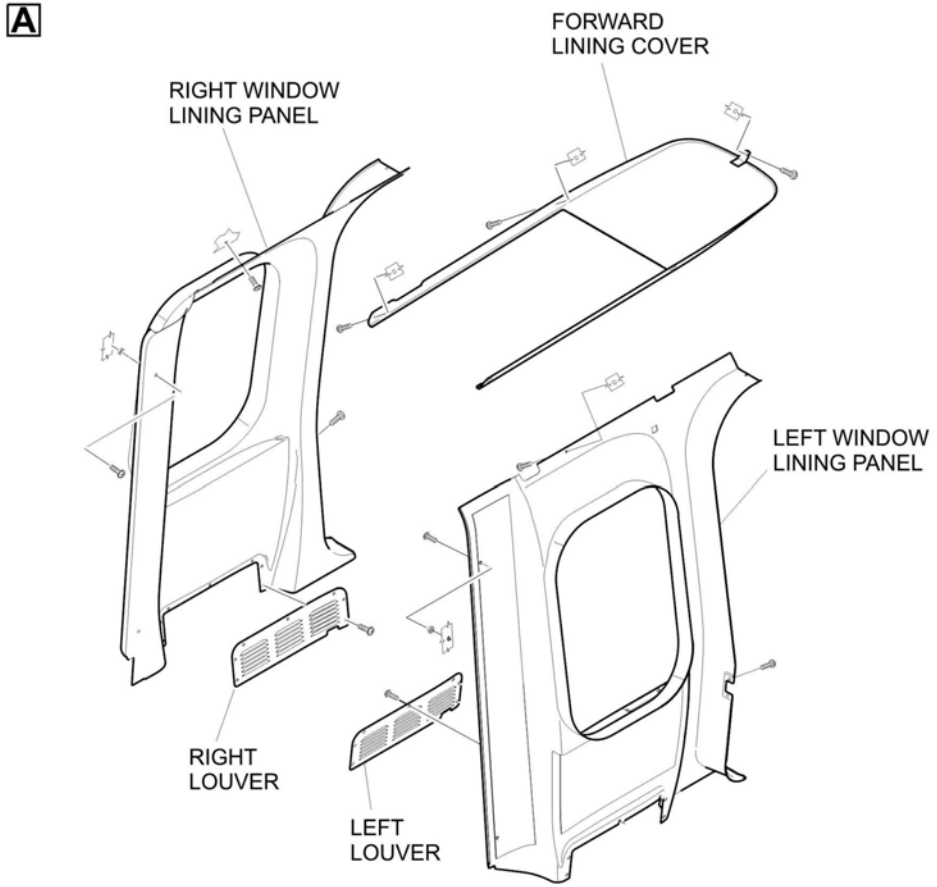
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COCKPIT LINING (SHEET 2 OF 2)



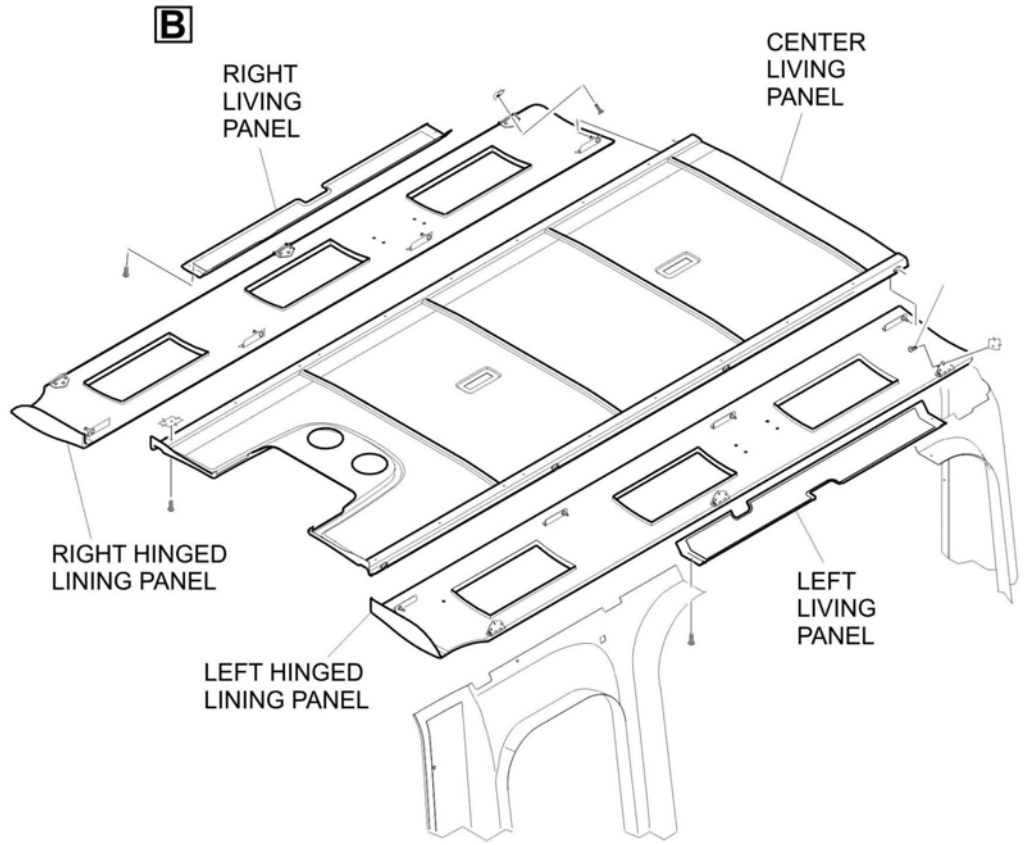
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CABIN LINING (SHEET 1 OF 3)



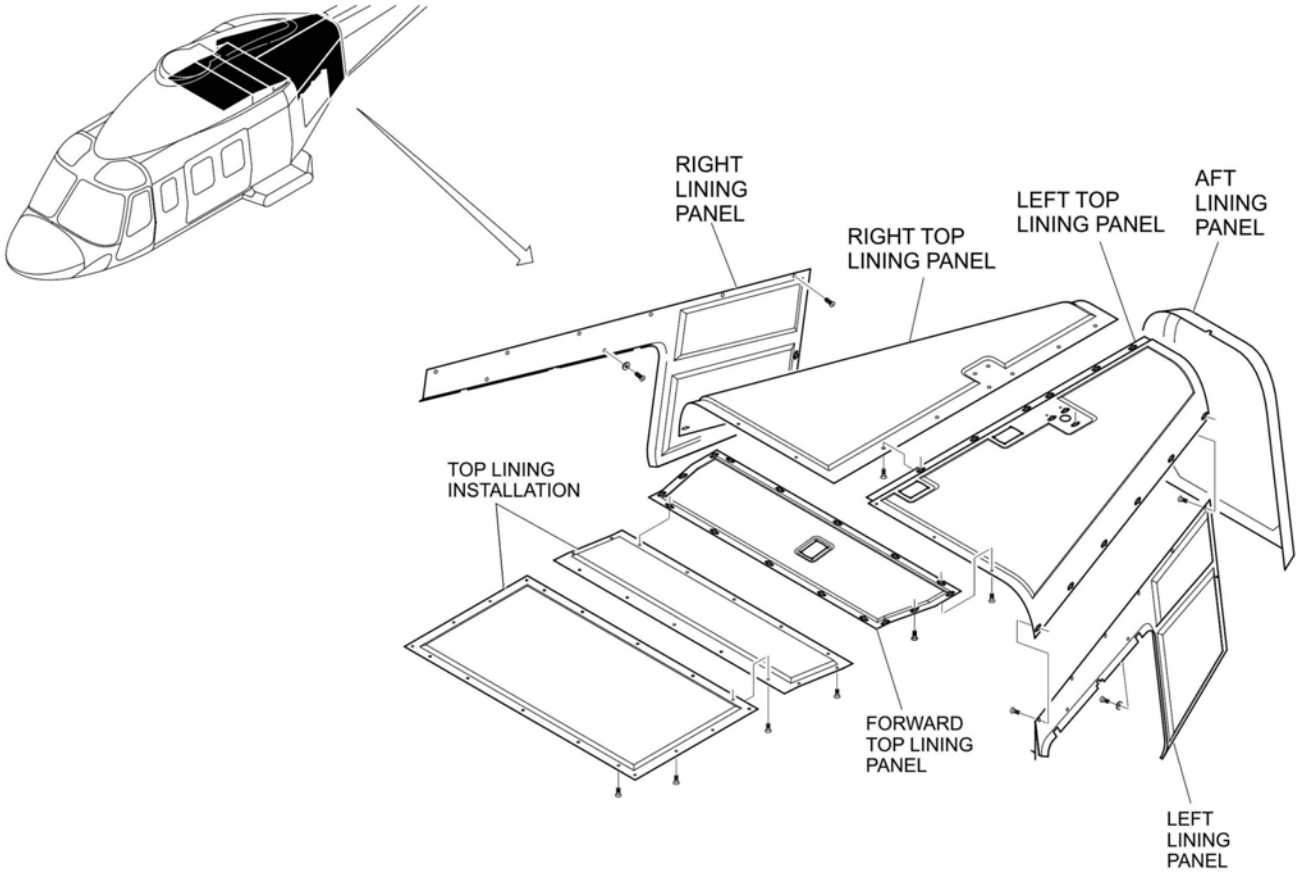
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CABIN LINING (SHEET 2 OF 3)



ICN-39-A-250000-G-A0126-00027-A-01-1

CABIN LINING (SHEET 3 OF 3)



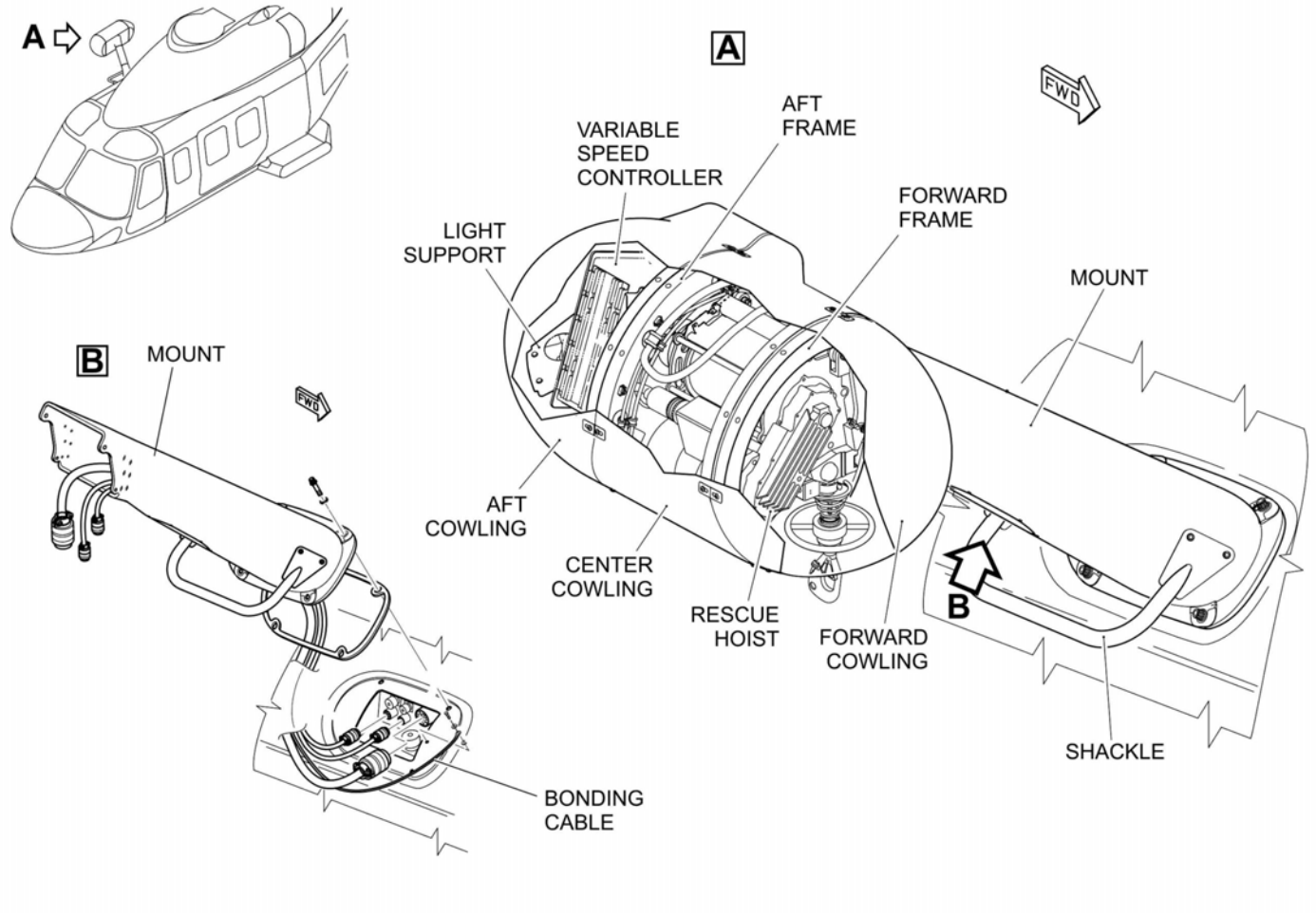
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AFT FUSELAGE LINING INSTALLATION

AERIAL DELIVERY

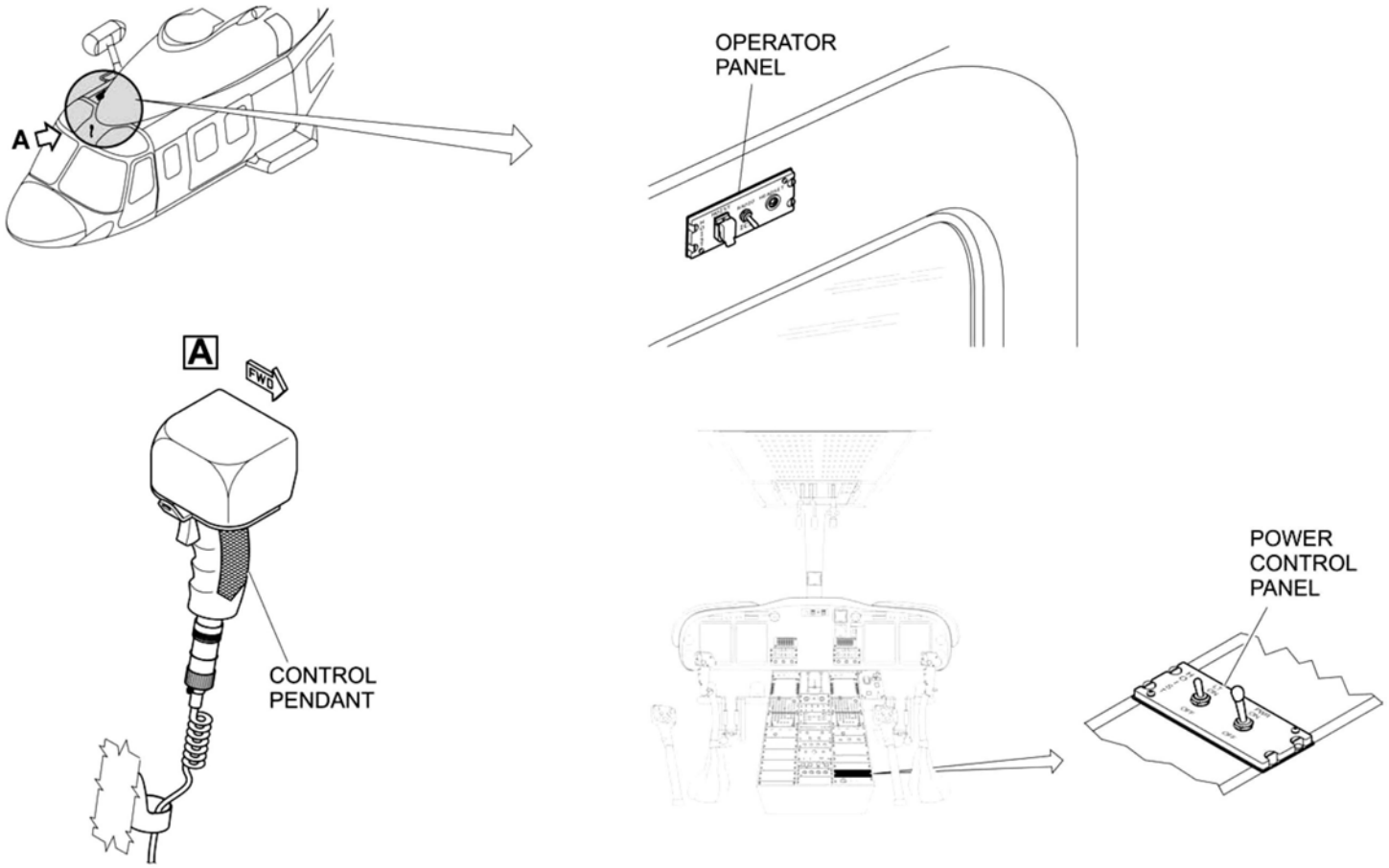
The aerial delivery system can be composed by one or more of this components:

- The rescue hoist system (optional);
- The cargo hook system (optional);
- The cargo hook video camera system (optional).



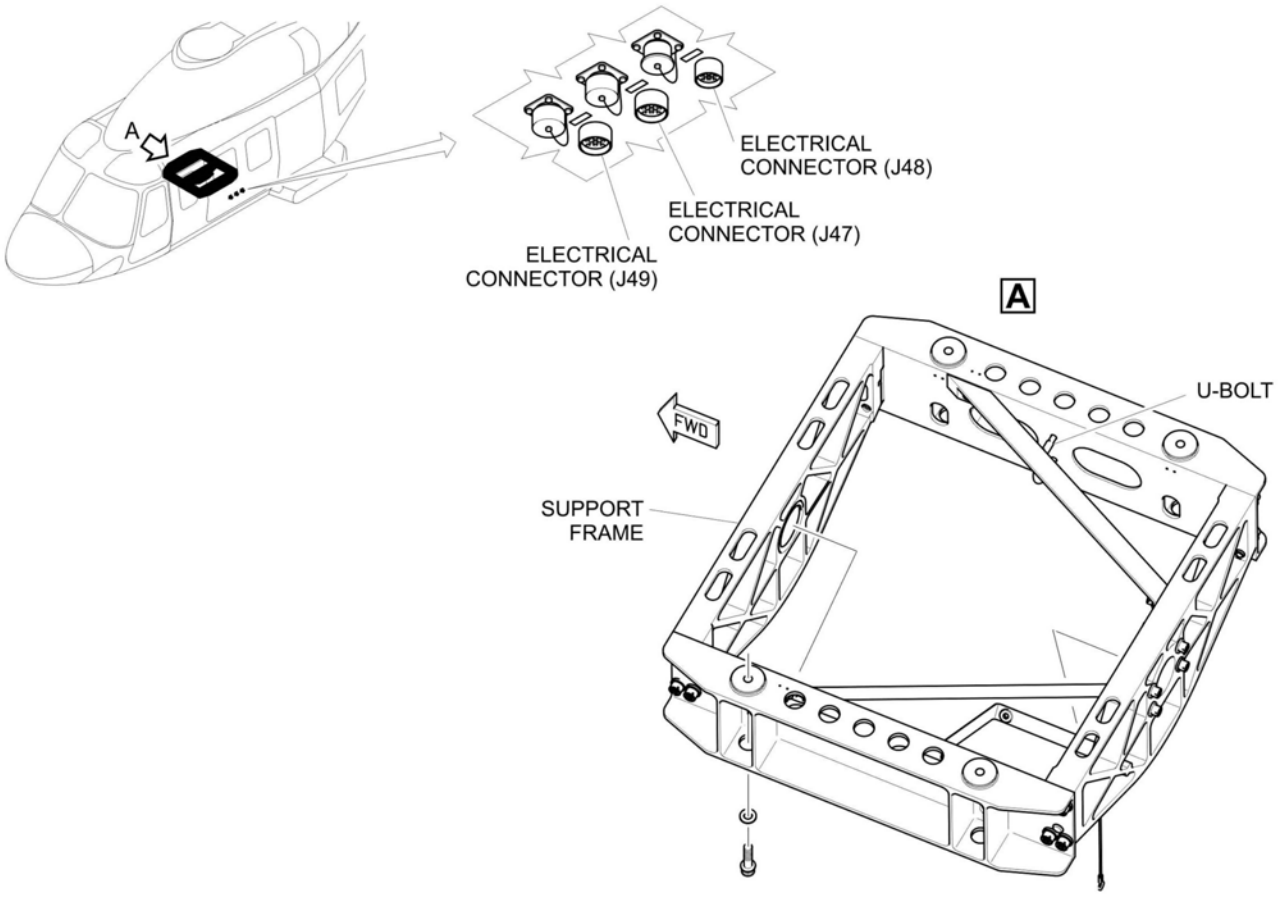
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RESCUE HOIST SYSTEM (SHEET 1 OF 2)



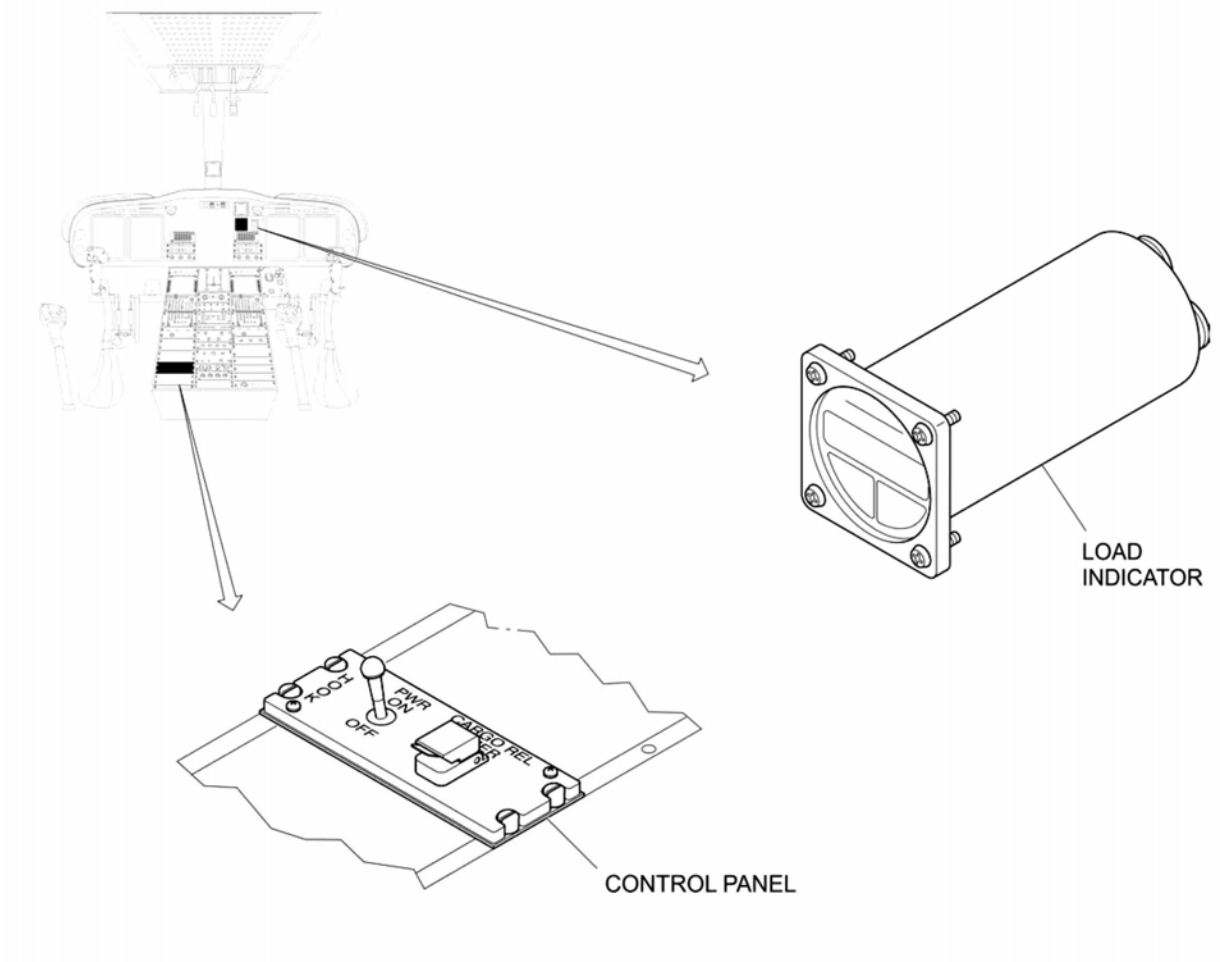
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RESCUE HOIST SYSTEM (SHEET 2 OF 2)



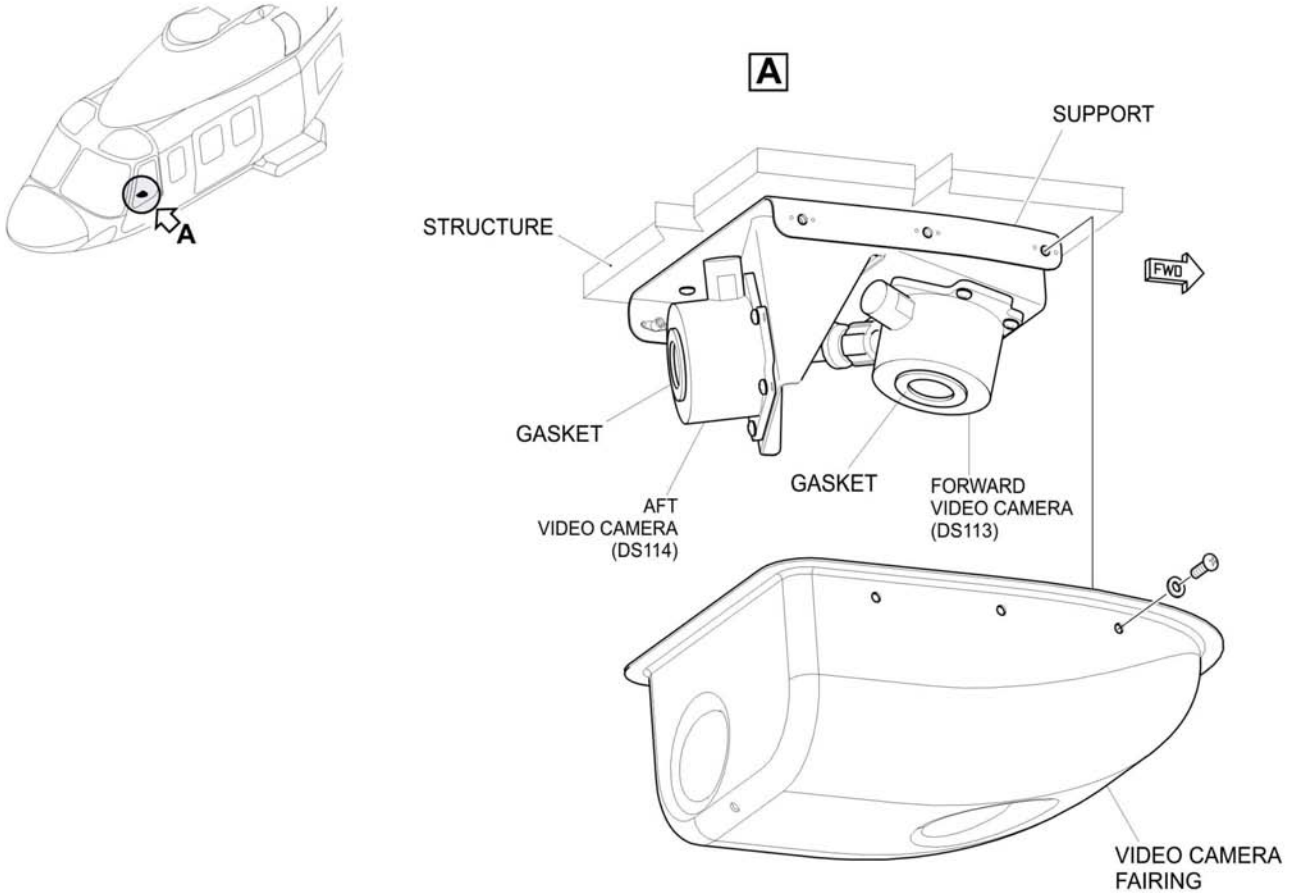
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CARGO HOOK SYSTEM (SHEET 1 OF 2)



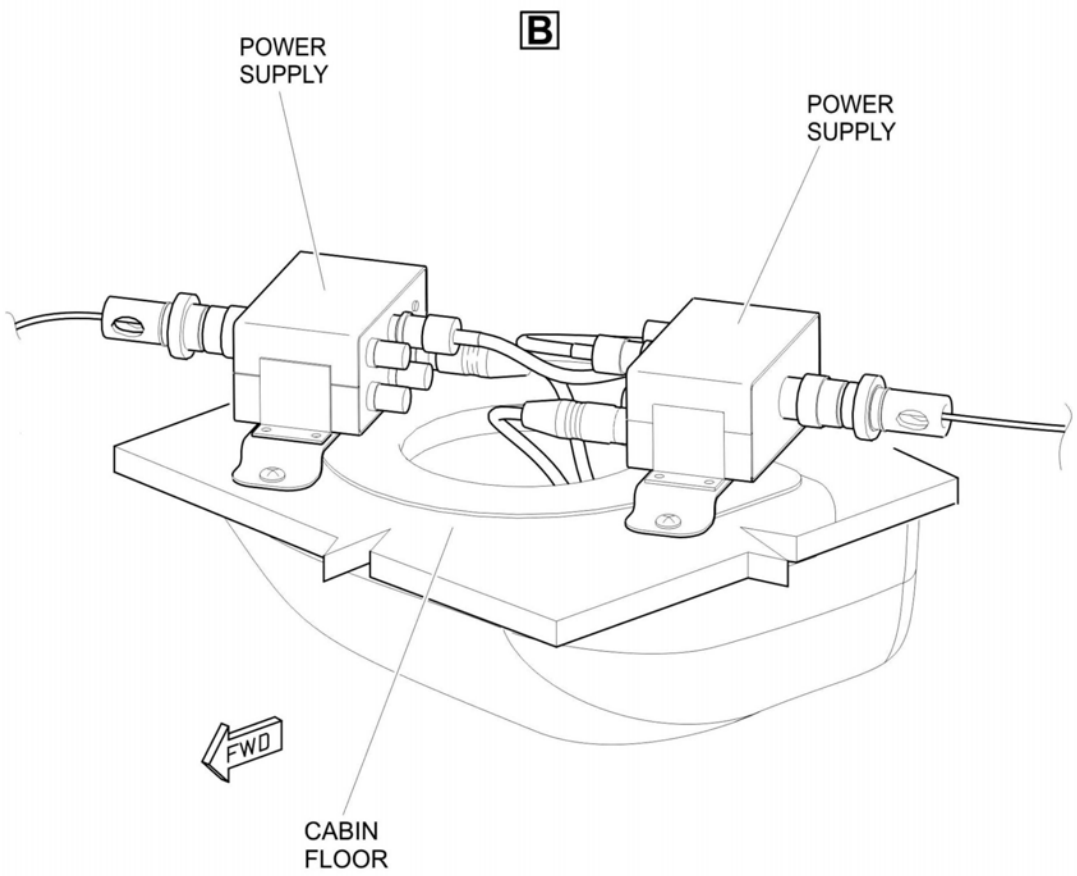
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CARGO HOOK SYSTEM (SHEET 2 OF 2)



ICN-39-A-250000-G-A0126-00033-A-01-1

CARGO HOOK CAMERA SYSTEM (SHEET 1 OF 2)



ICN-39-A-250000-G-A0126-00034-A-01-1

CARGO HOOK CAMERA SYSTEM (SHEET 2 OF 2)

SECTION 20 - PASSENGER/OPERATING CREW COMPARTMENT

DESCRIPTION

The cabin seat installation 12 seat configuration includes these primary components:

- The forward cabin seats
- The aft cabin seats
- The safety belts
- The headrests

FORWARD CABIN SEAT INSTALLATION

The forward-cabin seat-installation includes the front row of seats in the cabin. There are four seats in this row. These seats point to the tail of the helicopter. The structure of the seats is of composite material. Each seat has two cushions (back and pan) of flexible polyurethane, a safety belt with shoulder harness and a headrest.
Four quick-disconnect pins attach the front cabin seats to the rail on the cabin floor through the seat guides.

AFT CABIN SEAT INSTALLATION

The aft-cabin seat-installation includes the two aft row of seats in the cabin. There are four seats in each row. These seats point to the nose of the helicopter. The structure of the seats is of composite material. Each seat has two cushions (back and pan) of flexible polyurethane, and a safety belt with shoulder harness.

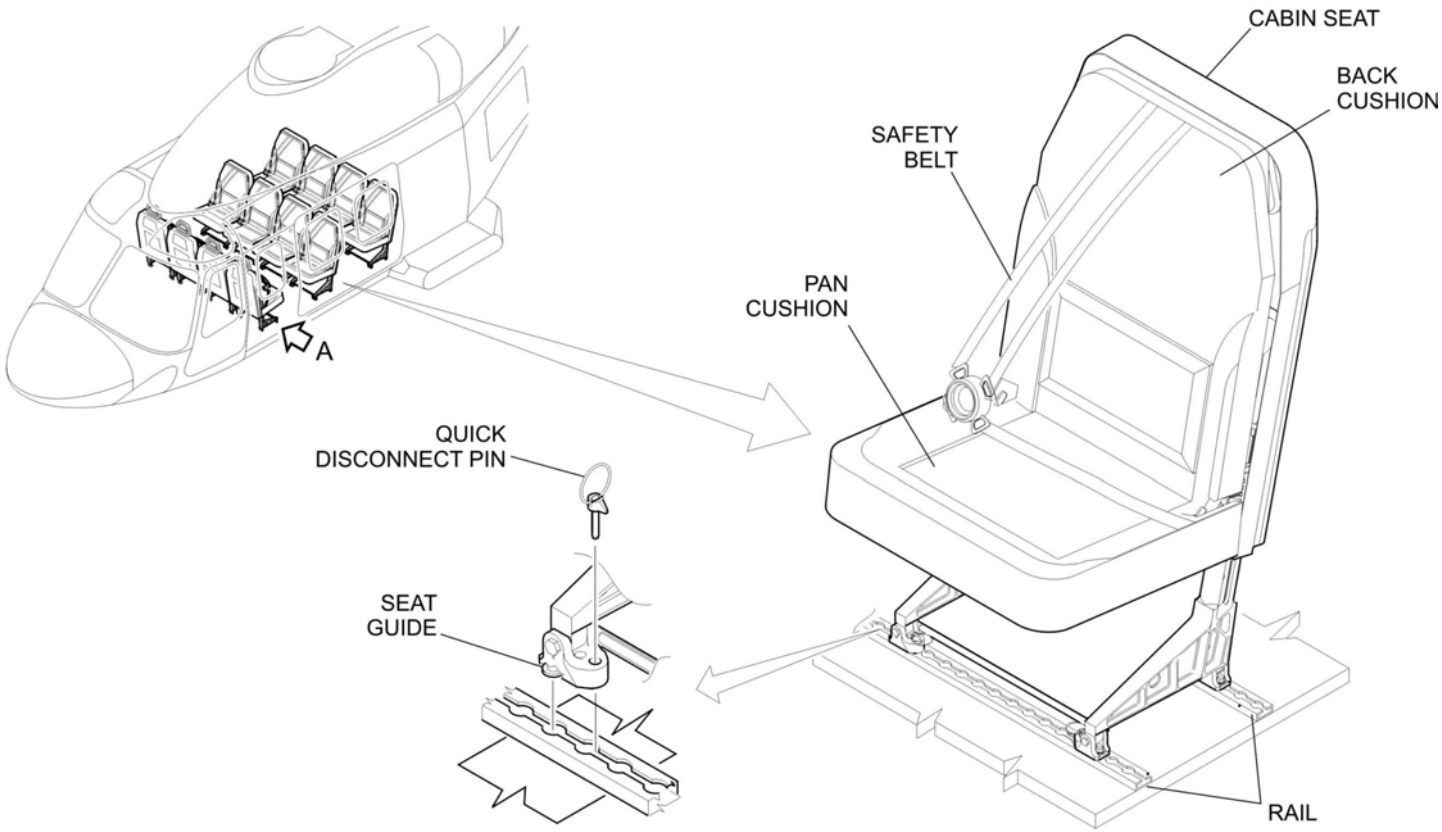
Four quick-disconnect pins attach the aft cabin seats to the rail on the cabin floor through the seat guides.

SAFETY BELTS

Each cabin seat has a safety belt with shoulder harnesses with a quick release fitting to attach and release the belts. The safety belts include an inertial reel that has an automatic locking mechanism that prevents the unwanted forward movement of the passenger in the seat.

HEADREST

Each forward cabin seat has a headrest. The headrest is installed on the top of the seat.



ICN-39-A-252000-G-A0126-00035-A-01-1

PASSENGERS SEATS (12 PAX CONFIGURATION)

DESCRIPTION

The cabin seat installation 15 seat configuration include these primary components:

- The forward cabin seats
- The aft cabin seats
- The safety belts
- The headrest

FORWARD CABIN SEAT INSTALLATION

The forward cabin seat installation is on the forward row seats of the cabin. The forward cabin seats are in composite material. The structure of the seat includes the two cushions of the flexible polyurethane and the headrest.
The four quick-disconnect pins safety the cabin seat to the rail of the cabin floor. The safety belt installed in each cabin seat for safety the passenger to the seats. The forward cabin seat includes one row of the cabin seats. The row of the forward cabin seat installation includes the five cabin seats.

AFT CABIN SEAT INSTALLATION

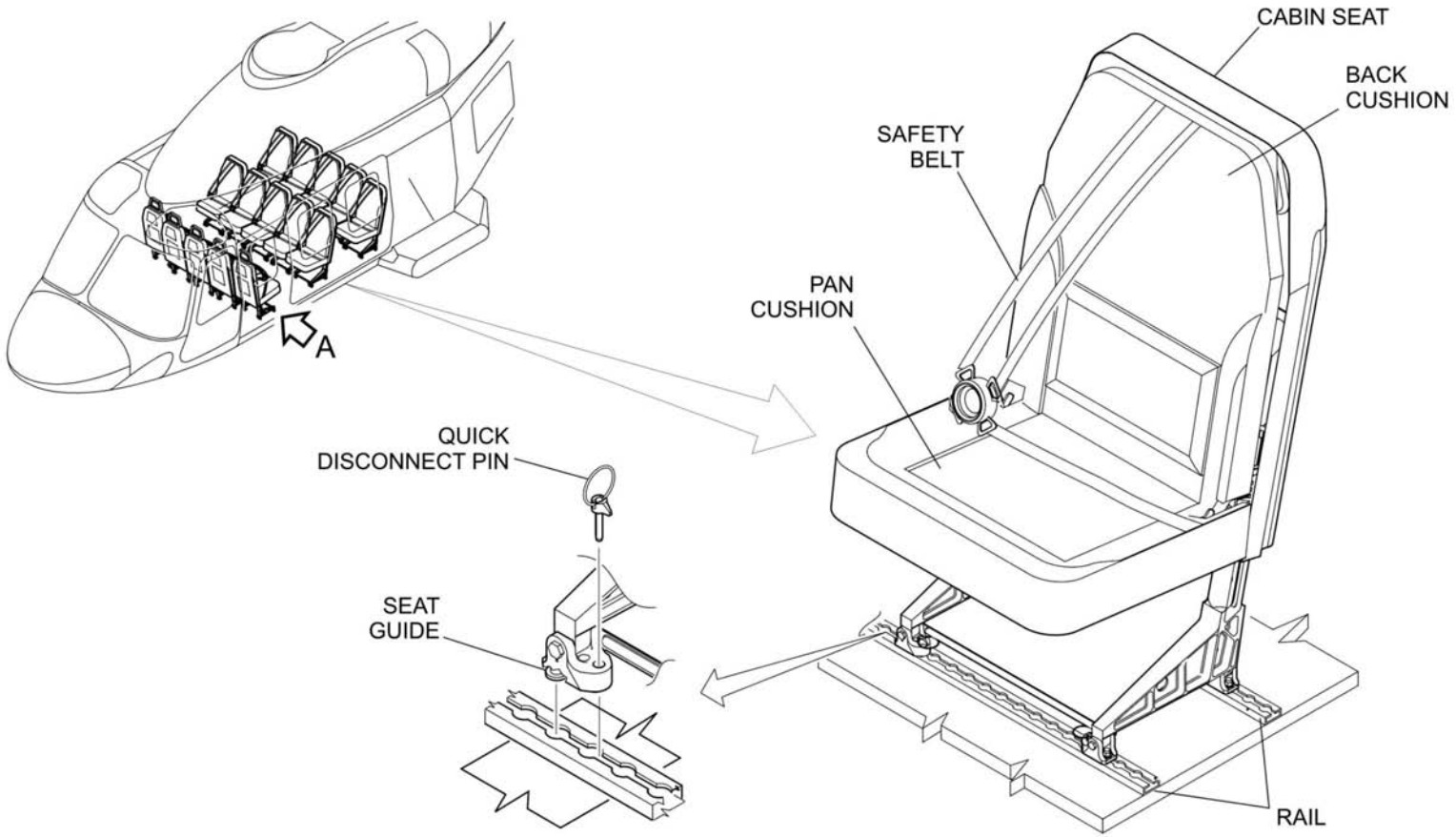
The aft cabin seat installation is on the aft row seats of the cabin, in front of the row of the forward seats. The forward cabin seats are in composite material. The structure of the seat includes the two cushions of the flexible polyurethane. The four quick-disconnect pins safety the cabin seat to the rail of the cabin floor. The safety belt installed in each cabin seat for safety the passenger to the seats. The aft cabin seat installation includes the two rows of the cabin seats. Each row includes the five cabin seats.

SAFETY BELTS

Each cabin seat is equipped with safety belts and inertia reel shoulder harnesses that consist of the inertial reel, the shoulder harness, the quick-release fitting and the safety belts. The inertial reel incorporates an automatic locking mechanism that prevents the seat occupant from pitching forward in the event of rapid deceleration.

HEADREST

The row of the forward cabin seat has the headrest. The headrest is installed in the top side of the each seat.



ICN-39-A-252000-G-A0126-00036-A-01-1

PASSENGERS SEATS (15 PAX CONFIGURATION)

SECTION 60 - EMERGENCY

DESCRIPTION

The Emergency Locator Transmitter (ELT) system is designed for installation in helicopters and fixed wing aircraft. The purpose is to provide the localization of the aircraft after a crash or an emergency landing.
The whole system consists of:

- an Emergency Locator Transmitter
- an ELT/NAV Interface Unit
- a Remote Switch Panel
- a buzzer
- a rod antenna
- a ELT circuit breaker

The system transmits automatically, in the event of crash, the standard swept tone on 121.5 MHz. (Civil) and 243.0 MHz (Military) and every 50 seconds for 520 milliseconds (long message protocol) the aircraft position on the 406.025 MHz (SARSAT).
During that time an encoded digital message is sent to the satellite. The information contained in that message includes:

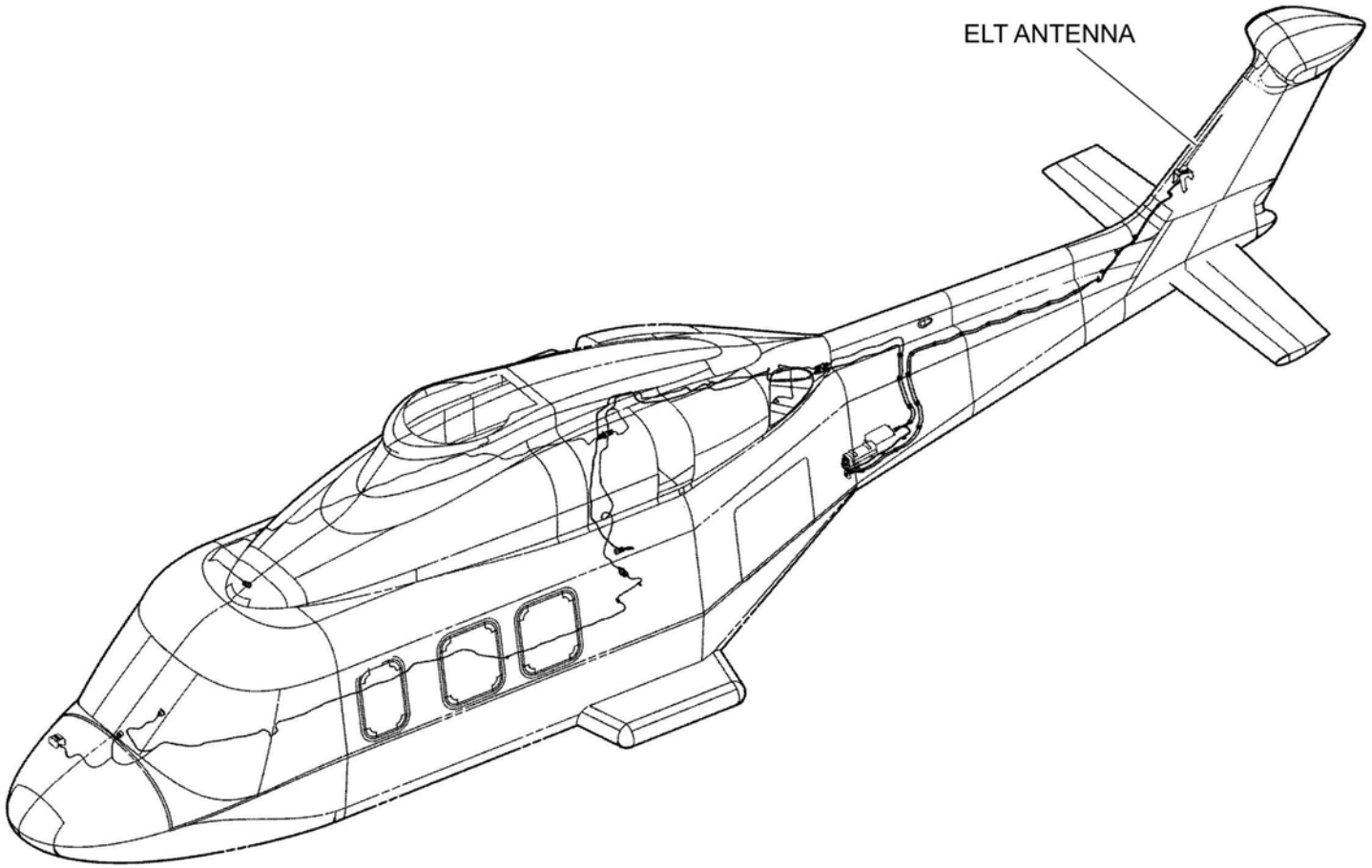
- serial number of the transmitter or aircraft ID
- country code
- ID Code
- position coordinates when coupled to an ELT/NAV Interface Unit. ELT/NAV is programmed by ARTEX. The ELT is automatically reconfigured by ELT/NAV. This allows to move the ELT from one aircraft to another.

The ELT digital message can also contains information which allow the search and rescue authorities to contact the owner/operator of the aircraft through a data base.

Information contained in the database that may be useful in the event of a crash is shown below:

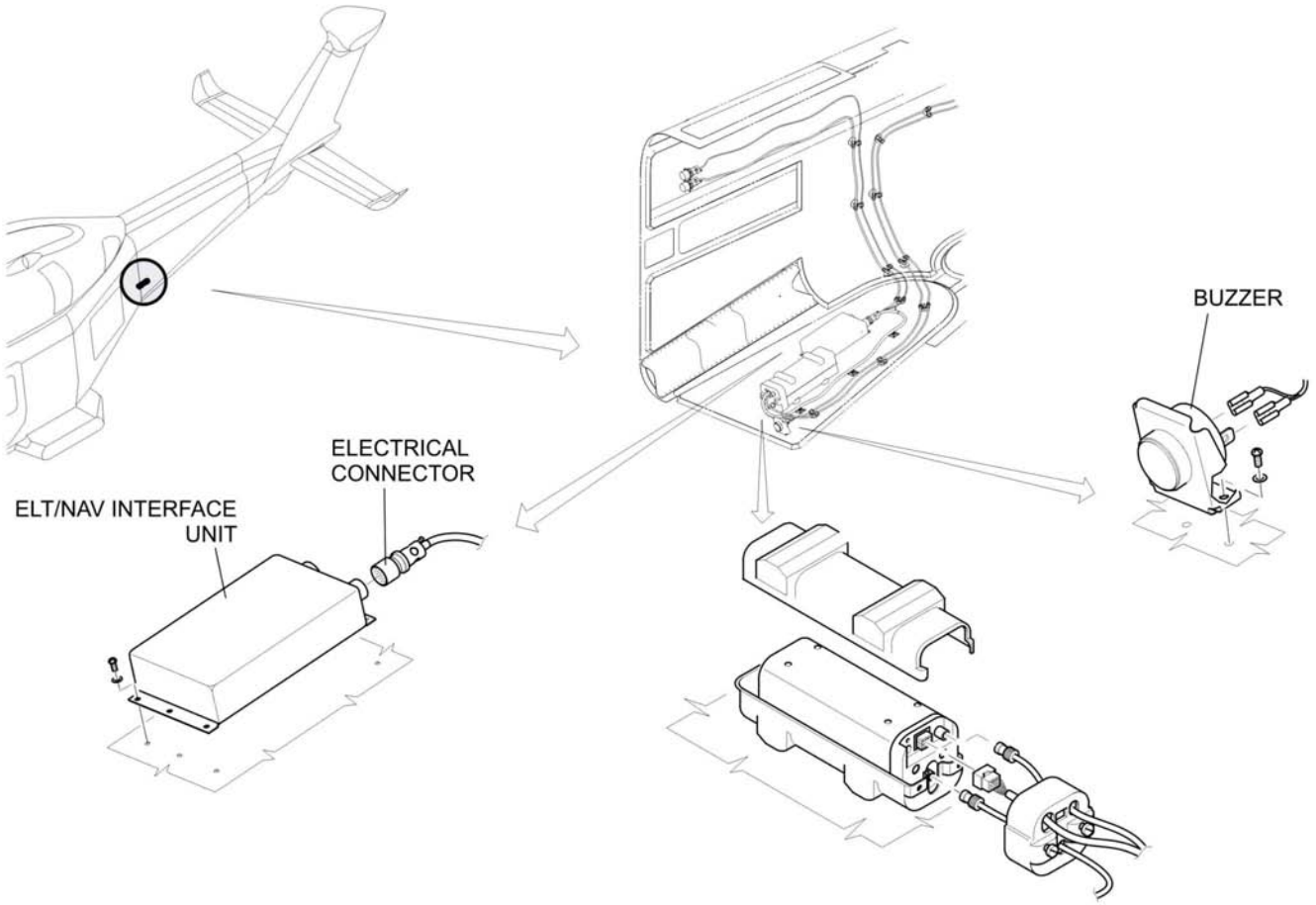
- Type of Aircraft
- Address of Owner
- Telephone Number of Owner
- Aircraft Registration Number
- Alternate Emergency Contact

The 406.025 MHz transmitter operates for 24 hours and then shuts down automatically. The 121.5 / 243.0 MHz transmitter continues to operate until the unit has exhausted the battery power which typically will be at least 72 hours. When the ELT is coupled with the aircraft's navigation system, the position accuracy improves to approximately 100 meters. Programming the ELT with the 24 bit long message protocol allows the ELT to be automatically programmed by the ELT/NAV interface with the aircraft's 24 bit address.
The ELT/NAV interface unit must be strapped (binary "1" bits tied to ground) with the same 24 bit address as the ICAO or Mode S transponder system 24 bit address.
Once the ELT is activated and the 406.025 MHz signal is detected from the satellite and a position is calculated, the 121.5 / 243.0 MHz transmission are used to home in on the crash site.



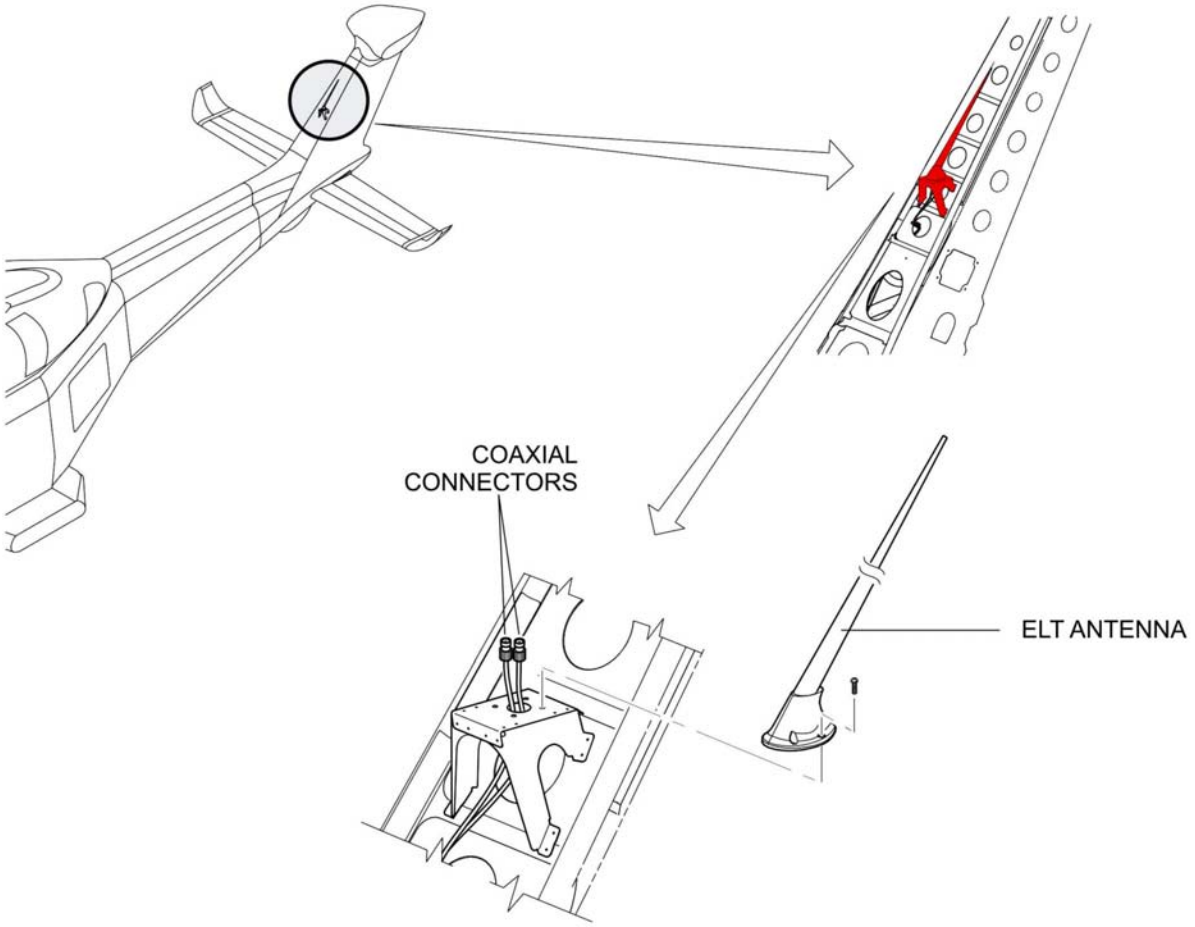
ICN-39-A-256000-G-A0126-00037-A-01-1

ELT SYSTEM COMPONENTS LOCATION (SHEET 1 OF 3)



ICN-39-A-256000-G-A0126-00038-A-01-1

ELT SYSTEM COMPONENTS LOCATION (SHEET 2 OF 3)



ICN-39-A-256000-G-A0126-00039-A-01-1

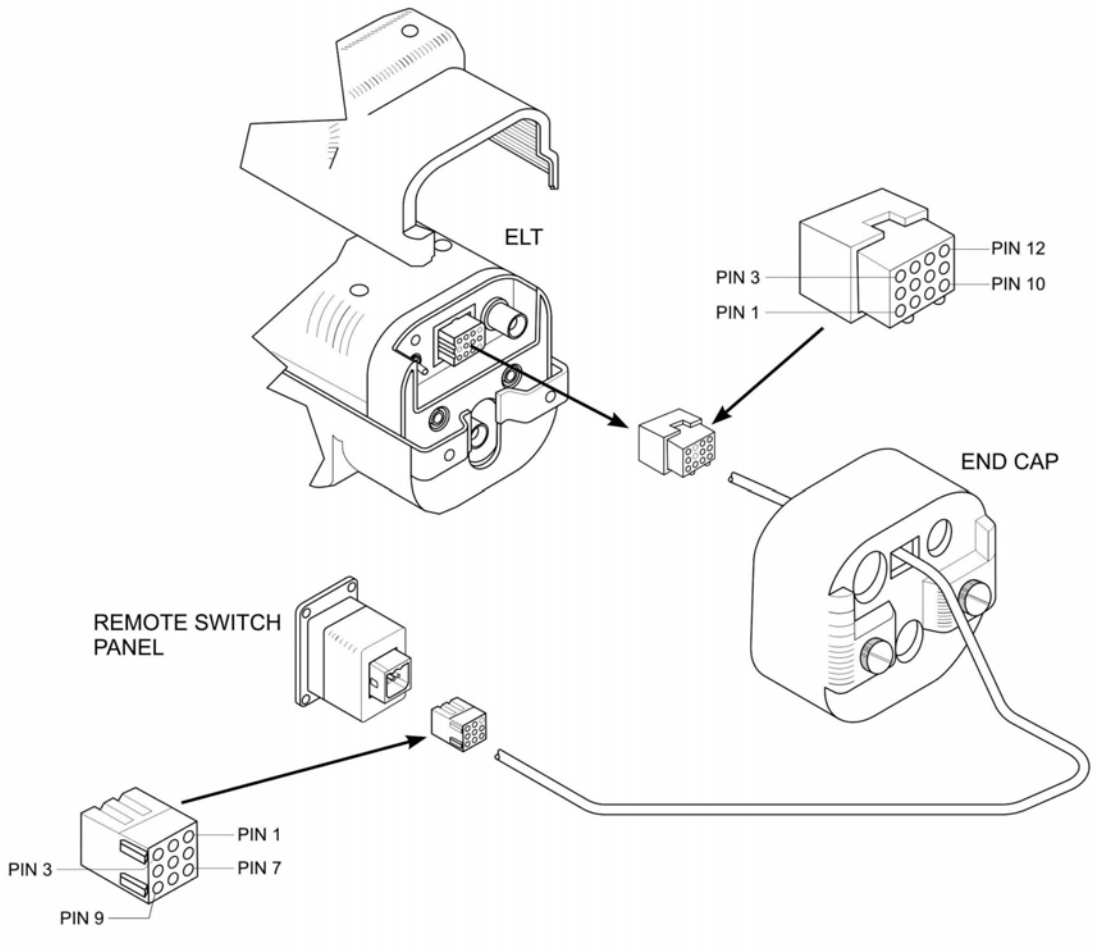
ELT SYSTEM COMPONENTS LOCATION (SHEET 3 OF 3)

OPERATION

The unit activates only when securely mounted in its tray and has pins 5 and 8 jumpered. The ELT can not be accidentally activated by dropping, rough handling or during shipping.
When the ELT is activated, the presence of the emergency swept tone and a flashing front panel light indicates a normally functioning unit.
Under normal operation the switch configuration on the front panel is the down position, reading “ARM” while the switch on the ELT unit is also positioned down to read “OFF”. The periodicity of operational checks is at the operator’s discretion, however, a monthly test is recommended.
An aural and a visual monitor is provided to alert the pilot when the ELT has been activated and is transmitting.
The following requirements apply to the activation monitors:

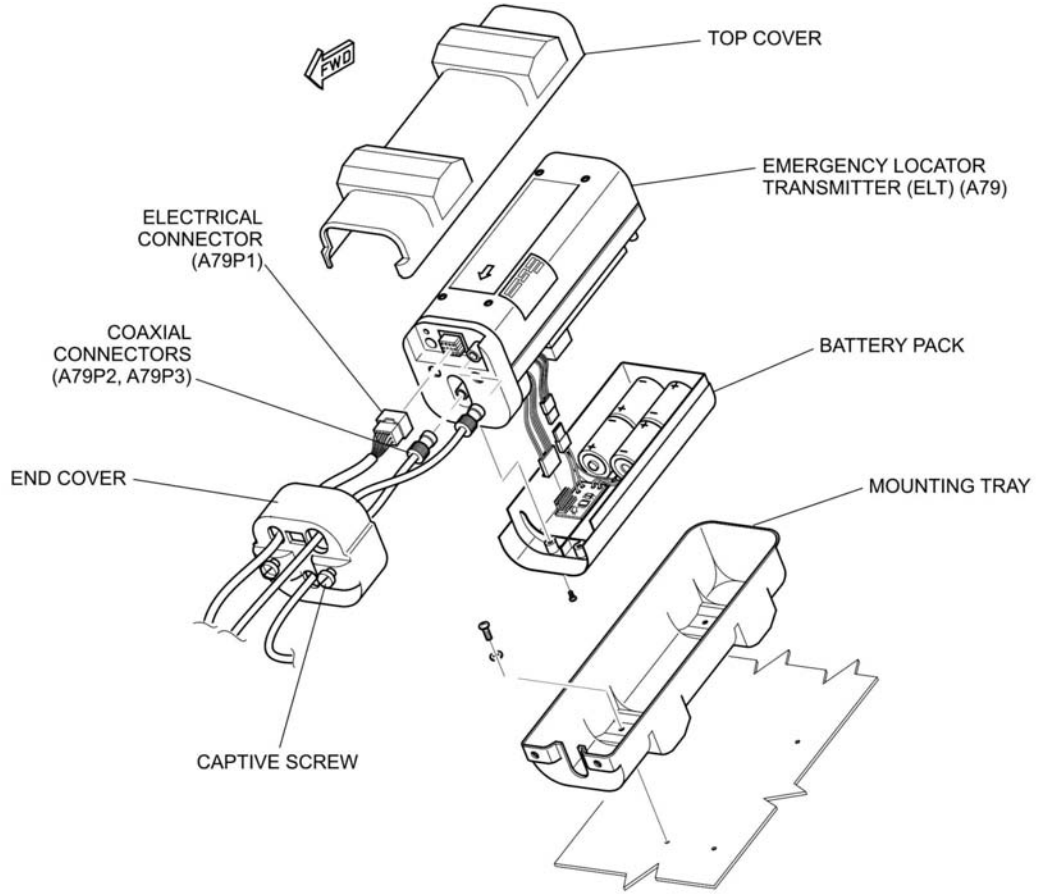
- the aural monitor provides a distinct signal enabling a search and rescue team to locate an aircraft with a transmitting ELT in a confined area with a large number of aircraft (i.e. such as an airport). The search and rescue team would listen for the aural monitor and easily locate and disable the offending ELT.
- the visual monitor is designed to be installed so that it can be viewed from the pilot’s position.
  Its intended function is to inform the pilot that the ELT is transmitting, avoiding a situation where an aircraft is flying with its ELT transmitting.

The crash force activation sensor or G-Switch is designed to activated with a change of velocity of 4.5 fps ± 0.5 fps both under normal conditions and while being subjected to 30 G’s of cross axis forces.
The battery pack of the ELT consists of 4 “D” size Lithium Manganese Dioxide cells connected in series. To prevent the cells from being charged, diodes are connected across each cell and fuses are connected to the output.
The battery pack expires 5 years from the date (month) it is shipped from Artex.
The rod antenna (dual input) has two connectors: one for the 121.5 / 243.0 MHz and the other one for the 406.025 MHz.



ICN-39-A-256000-G-A0126-00040-A-01-1

ELT TO REMOTE SWITCH INTERFACE



ICN-39-A-256000-G-A0126-00041-A-01-1

EMERGENCY LOCATOR TRANSMITTER

CONTROLS AND INDICATORS

The ELT controls are housed on the Remote Switch Panel and on the ELT Unit.
The Remote Swicth Panel provides the following functions:

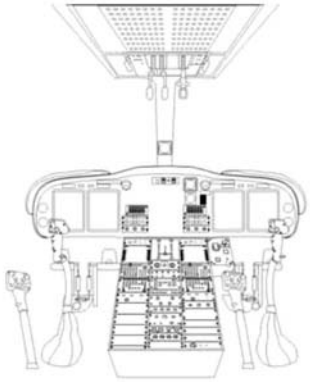
1 The red switch:
- ARM / RESET position: …. the ELT automatically activates on impact.
- ON position: ..... when selected, the pilot activates manually the ELT.

2 The red LED: ..... flashes when the ELT is activated.

NOTE A

If the ELT is activated accidentally, the switch on the panel must be set to ON then immediately back to ARM/RESET. The ELT may also be reset by positioning the unit switch to On then immediately back to OFF.
The ELT can not be RESET if either the panel switch or the unit switch is in ON position.

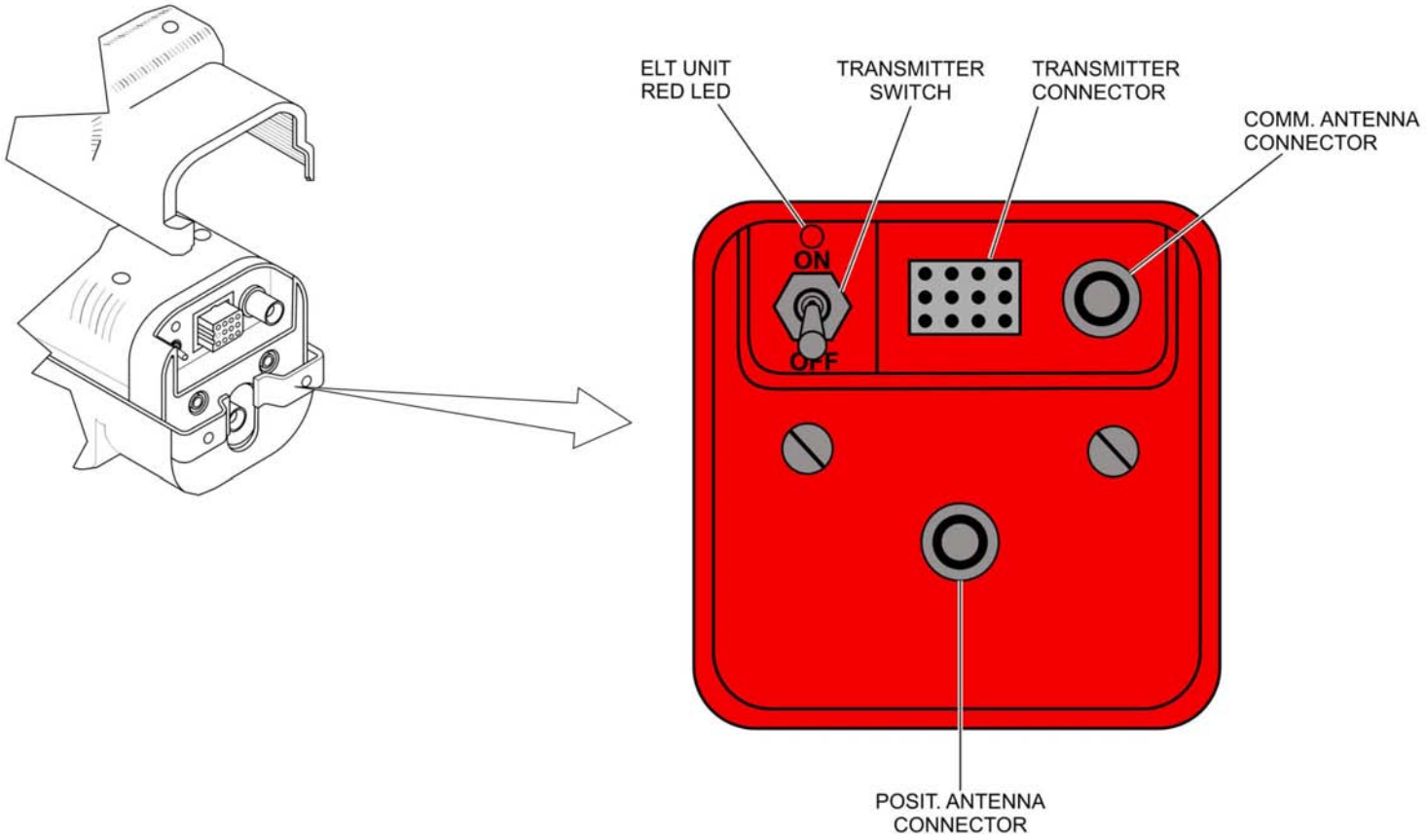
The buzzer provides:
- the aural signal:
  - it is a standard swept tone that is activated in tandem with the red LED.





ICN-39-A-256000-G-A0126-02922-A-01-1

ELT SYSTEM – REMOTE SWITCH PANEL CONTROLS



ICN-39-A-256000-G-A0126-00043-A-01-1

ELT SYSTEM – ELT UNIT CONTROLS

ELT SYSTEM FUNCTIONAL TEST

WARNING

DO NOT ALLOW TEST DURATION TO EXCEED 5 SECONDS. THE ELT WILL TRANSMIT A 406.025 MHZ. SIGNAL AFTER THE ELT IS ACTIVE APPROXIMATELY 47 SEOCNDS, THE SATELLITE SYSTEM CONSIDER THE TRANSMISSION TO BE A VALID DISTRESS SIGNAL.
BE SURE TO NOTIFY ANY NEARBY CONTROL TOWER OF YOUR INTENTIONS.
ALWAYS PERFORM THE TEST WITHIN THE FIRST 5 MINUTES OF THE HOUR UTC (COORDINATED UNIVERSAL TIME) AS REQUIRED.

- Tune the receiver (usually the aircraft radio) to 121.5 MHz.
- Push the switch on the Remote Switch Panel to ON, wait for 3 sweeps on the receiver, which takes about 1 second and then the switch back to ARM while paying special attention of the LED upon entering the ARM position.
- To pass the test, you must hear 3 sweeps and the front panel LED immediately begins to flash continuously. During the ON/ARM transition, the microprocessor in the ELT checks the G-Swicth (automatic activation switch) latching circuit, pins 5 and 8 on the 12 pin connector; the 406.025 MHz transmitter for proper RF output; the position data is present and a battery check. If the ELT is working properly, the sequence following entry to the ARM condition will result in LED stays in illuminated for approximately 1 second, then extinguishing.
- After the single flash and a 1 second pause if the circuit failure will be detected, a flashes sequence would appear as detailed below:
  - 1 flash indicates a G-Switch loop open failure.
  - 3 flashes indicates a 406.025 MHz transmitter problem (i.e. bad or unconnected coax cable, an antenna problem. low power output or a programming error).
  - 5 flashes indicates there is non navigation data present. this is most likely due to improper wiring between the system interface connections, improper programming, invalid navigation data on an ELT/NAV Interface Unit is not being utilized.
  - 7 flashes indicates a battery problem (i.e. battery usage time over an hour).

NOTE B

For installation using the ELT/NAV Interface, there is an error condition where the LED on ELT will flash rapidly. This occurs 2 minutes after power is applied to the ELT/NAV Interface if the ELT is not responding because either:

- the ELT is programmed for a protocol other than 24 bit and the ELT/NAV Interface has been strapped for a 24 bit address; or
- the RS-232 TX line from the ELT (pin 12) to the ELT/NAV Interface (pin 10) is not connected.

CAPTIONS

No warning / cautions are involved with the ELT system.

LEADING PARTICULARS

- Remote Switch Panel
  - Nominal supply voltage: 28 VDC (from CB 28 Battery Bus)

- ELT/NAV interface unit
  - Nominal supply voltage: 28 VDC (ESS BUS 2)
- Battery package
  - Power supply: 12,0V / 10Ah

SECTION 91 - RESCUE HOIST

GENERAL DESCRIPTION

The Rescue Hoist kit consists of:
An electric hoist motor and winch assembly actuated by an electrical motor and controlled by an electronic remote pendant.
An electronic control system allow the pilot to operate the hoist from the cockpit.
The main power requirement for the Rescue Hoist System is 28 VDC, 160 Ampere. The power required for the controls in the Control Pendant and the cable cutter for the Hoist, is 28 VDC, 5 Ampere. The power supplied by the helicopter is converted into the power required for the Rescue Hoist System by the Variable Speed Controller. This Variable Speed Controller provides the necessary power to the electric motor to drive the drum on the Hoist.

DESCRIPTION AND OPERATION OF THE MAJOR COMPONENTS

The primary components of the rescue hoist system are:

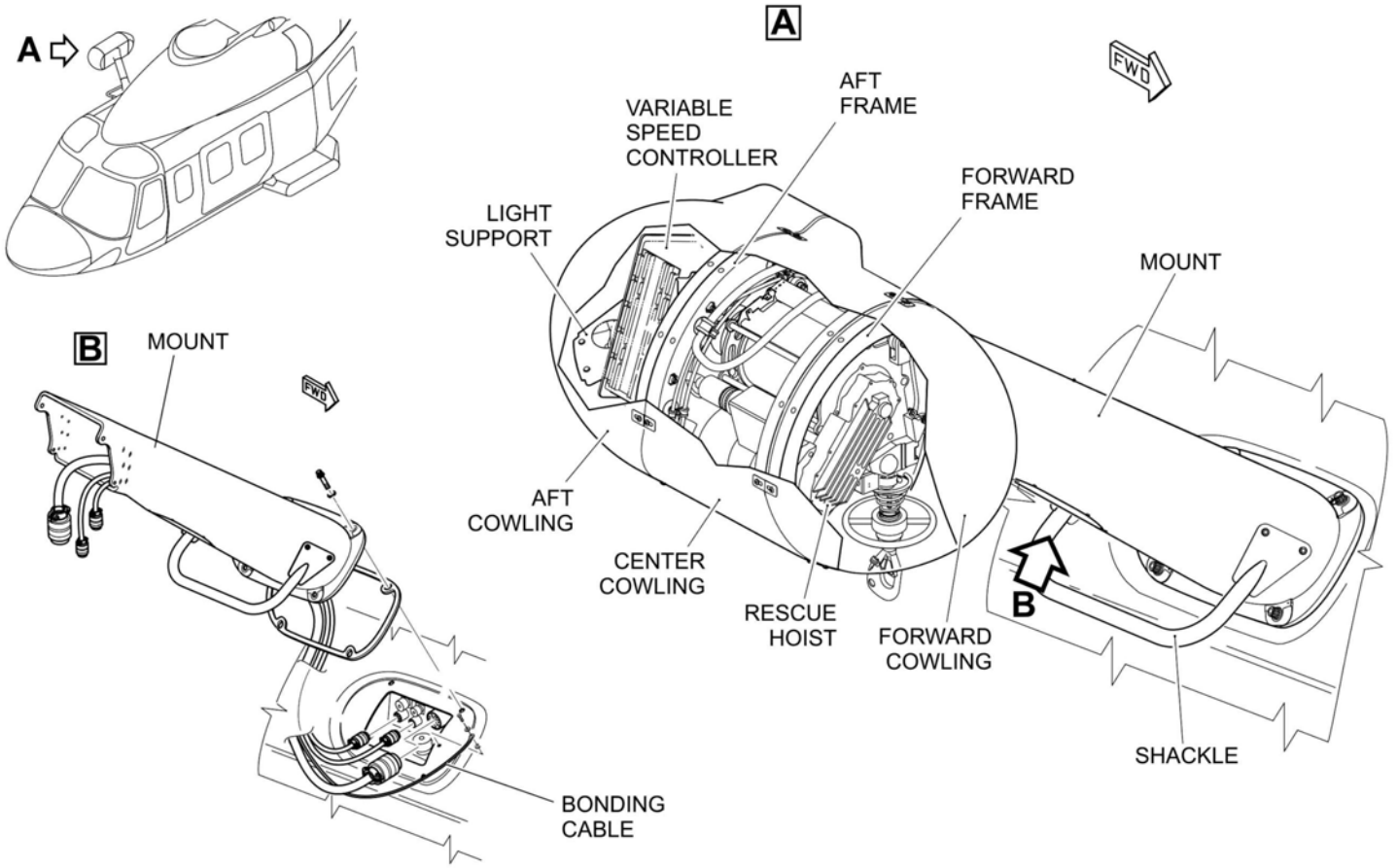
- The rescue hoist
- The variable speed controller
- The control pendant
- The power control panel
- The operator panel
- The hoist mount and cowlings.

RESCUE HOIST

The rescue hoist is installed externally on the right side of the helicopter above the cabin door, at STA X3900. It is installed on a mount and has cowlings for protection.
The rescue hoist has:

- An electric motor
- A rotating drum
- A load cable
- A hook assembly
- A cycle counter
- A cable guide
- A cable foul assembly
- A cable cutter.

The electric motor moves the rotating drum through the gear train and slip clutch assembly. It operates at constant speed when the pilot or the copilot controls it, at variable speed when the hoist operator controls it.
The rotating drum has a slot to attach the load cable to the drum flange.



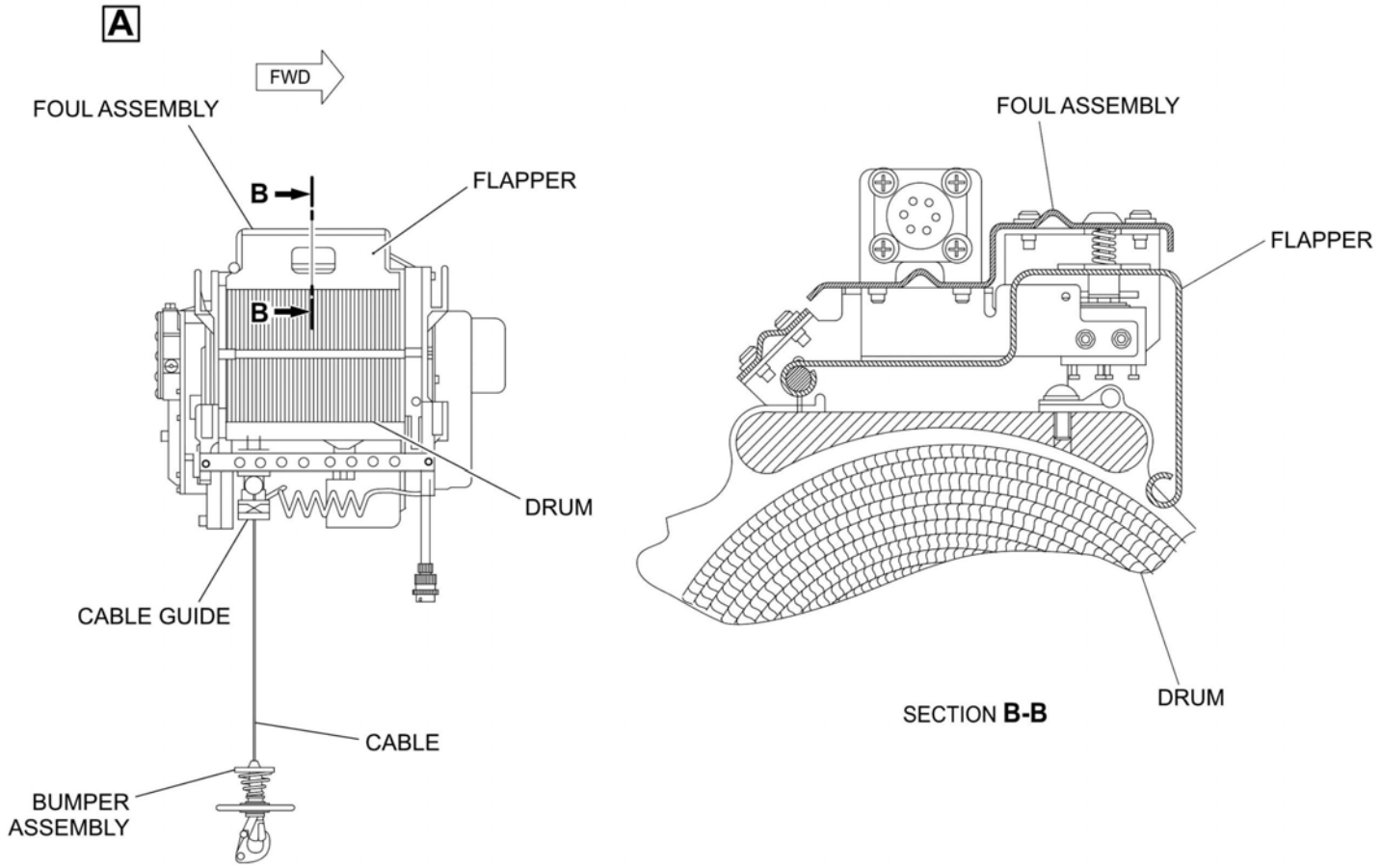
ICN-39-A-259100-G-A0126-00247-A-01-1

MAIN COMPONENTS LOCATION (SHEET 1 OF 6)

The load cable is a wire rope with a swaged stainless steel ball/shank fitting. The usable cable length is 74,7 m (245 ft).
The hook assembly is of the swivel type and can turn 360 degrees. It is attached to the ball end of the load cable with a terminal clamp, a nut and a housing. The hook assembly has a lockable keeper that prevents an accidental release of the load.
A gear guard is the protection of the cable from cuts and damages, if a foul cable condition occurs between the drum gear and the overrunning clutch.
The slip clutch, installed in the drum, operates if the cable load exceeds 544 to 816 kg (1200 to 1800 lb). The slip clutch is a protection device. It absorbs the load energy if the intermediate-speed limit switch and the full-in limit switch have a failure at the same time. It absorbs also the cable shock loads.
The cycle counter records each turn of the drum. The number on the counter, divided by 334, gives the total number of hoist cycles.
The cable foul assembly is a switch-operated safety device. It is installed on the hoist flanges parallel to the drum axis. The cable foul assembly stops the supply of electrical power to the electric motor when a cable foul starts on the hoist drum. Thus it prevents damage to the load cable.
The limit switches control the cable travel at full-in, full-out and up-intermediate and down-intermediate speed.

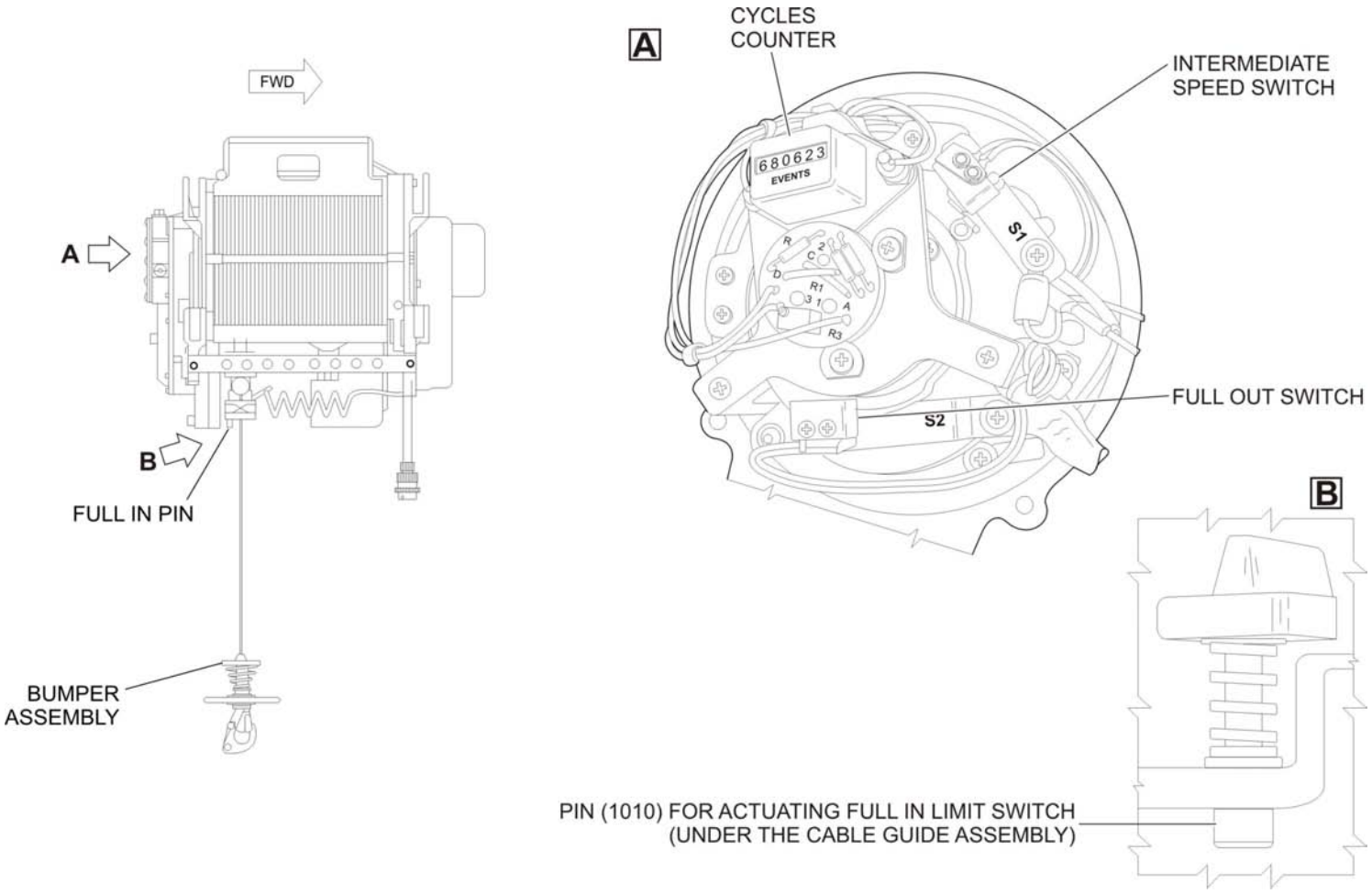
- The full-in limit switch stops the hoist operation when the bumper assembly touches the cable guide.
- The up-intermediate speed-limit switch decreases the hoist speed when the load cable gets to a specified distance from the point at which the full-in limit-switch must operate.
- The down-intermediate speed-limit switch decreases the hoist speed when the load cable gets to a specified distance from the point at which the full-out limit-switch must operate.
- The full-out limit switch stops the hoist operation when there are only 3 1/2 to 4 1/2 wraps on the drum.

The full-out, up-intermediate and down-intermediate speed limit-switches are installed on the drum. The full-in limit-switch is installed in the cable guide.



ICN-39-A-259100-G-A0126-00248-A-01-1

MAIN COMPONENTS LOCATION (SHEET 2 OF 6)



ICN-39-A-259100-G-A0126-00249-A-01-1

MAIN COMPONENTS LOCATION (SHEET 3 OF 6)

The cable cutter is installed in the cable guide. The cable cutter has an electrically-fired pressure cartridge, a guillotine and an anvil to cut the load cable in an emergency.

VARIABLE SPEED CONTROLLER

The variable speed controller (A84) is installed behind the aft frame of the structure that supports the rescue hoist.
The speed command signal is applied to the variable speed controller:

- When the hoist operator operates the thumb wheel on the control pendant (variable speed control mode).
- When the pilot operates the hoist control switch on the collective stick grip (constant speed control mode).

A circuitry installed in the main board of the controller processes the speed command signal.
The variable speed controller has the following safety devices:

- Under voltage lockout. The controller prevents hoist operation when the input dc voltage is <17 V dc.
- Motor overheat indicator. The controller causes the MTR HOT caution light on the control pendant to come on when the motor becomes too hot.

CONTROL PENDANT

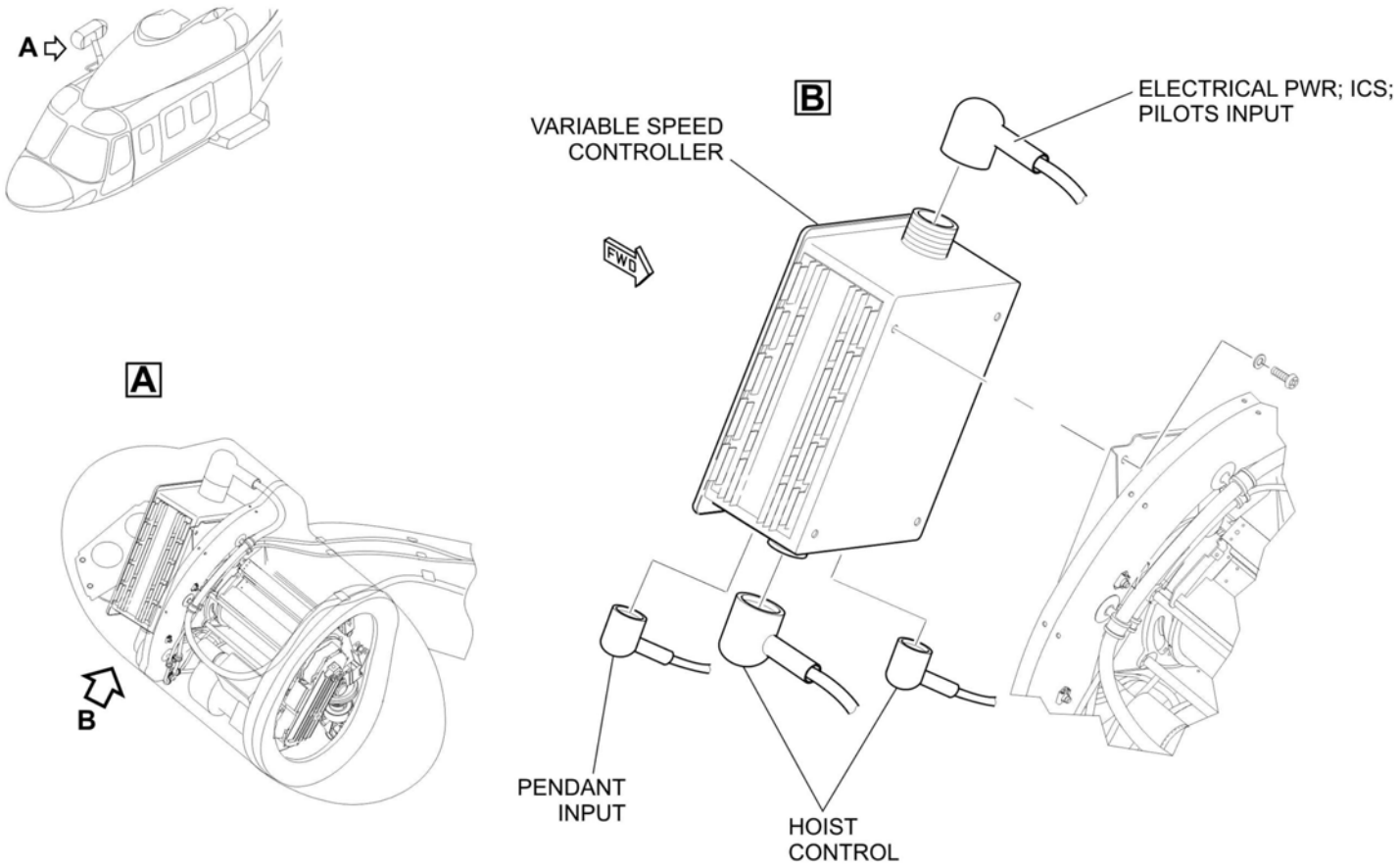
The control pendant is installed on the right side of the cabin. Its rear side has a bracket that goes into the control pendant support. This makes it easier to stow the control pendant. When in use the control pendant can be pulled out from the support. This lets the hoist operator move freely when he/she operates the hoist.
An electrical connector is installed in the bottom part of the control pendant handgrip It lets the control pendant be connected to the variable speed controller.
On the top part of the handgrip there are the controls/indicators that follow:

- The digital cable payout display.
- The direction/speed control thumb wheel that operates two switches and a potentiometer. The two switches (one for each direction) give an on-off signal. The potentiometer gives a speed command signal to the variable speed controller.
- The PWR ON advisory light.
- The SLOW caution light.
- The MTR HOT caution light.

On the rear of the handgrip there is a trigger switch that is used for connection of the hoist operator headset to ICS.

POWER CONTROL PANEL

The power control panel is installed on the interseat console and is attached with four quick-release fasteners.
Two switches are installed on the front panel. An electrical connector is installed on the rear of the control panel.



ICN-39-A-259100-G-A0126-00250-A-01-1

MAIN COMPONENTS LOCATION (SHEET 4 OF 6)

OPERATOR PANEL

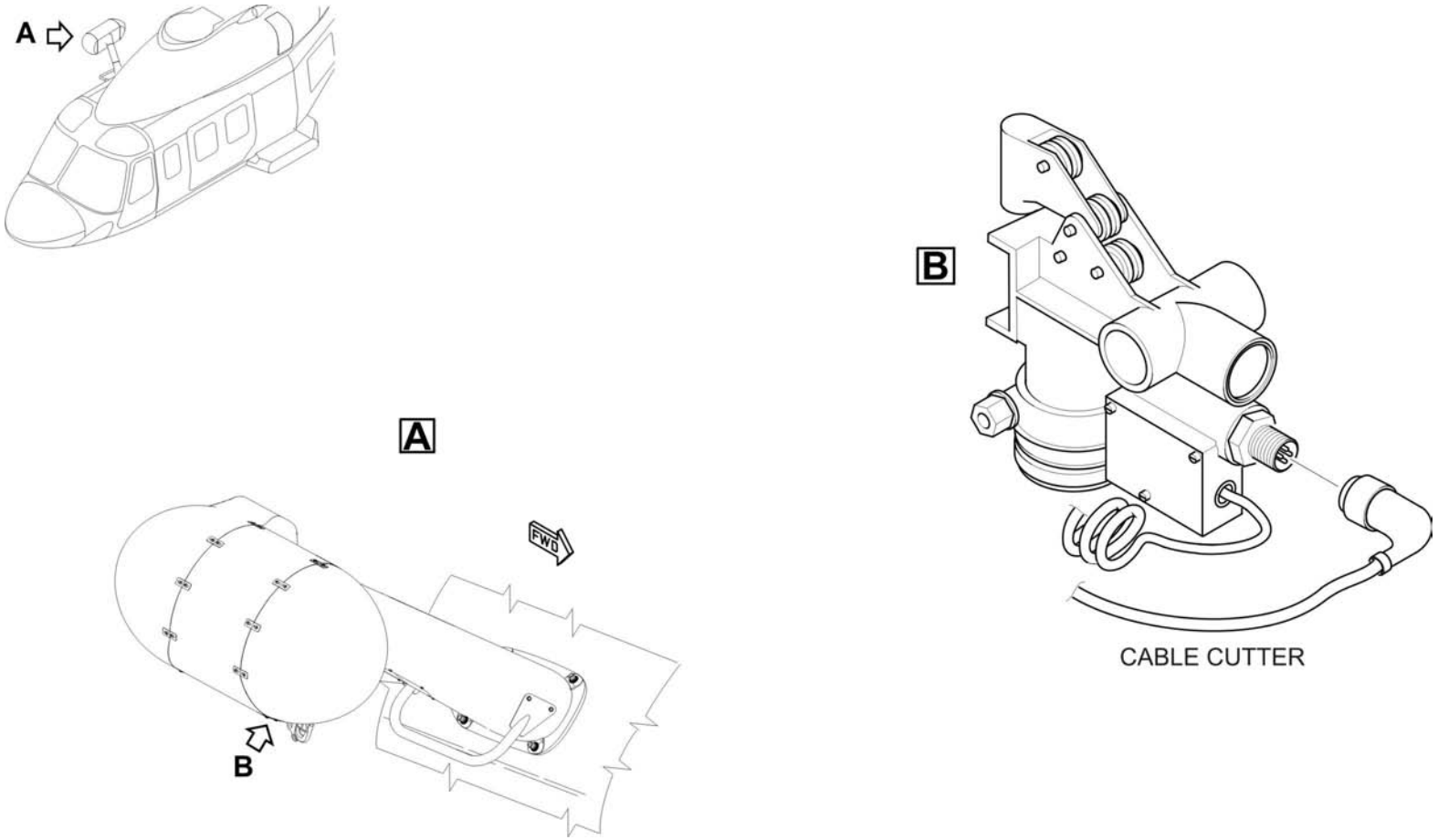
The operator panel is installed on the forward right side of the cabin. It is attached with four quick-release fasteners. Two switches and a receptacle for the headset cord plug are installed on the front panel. An electrical connector is installed on the left side of the operator panel.

HOIST MOUNT AND COWLINGS

The hoist installation structure includes a mount, a forward frame, an aft frame, a forward cowling, a central cowling and an aft cowling.
The mount is a tubular structure that has:

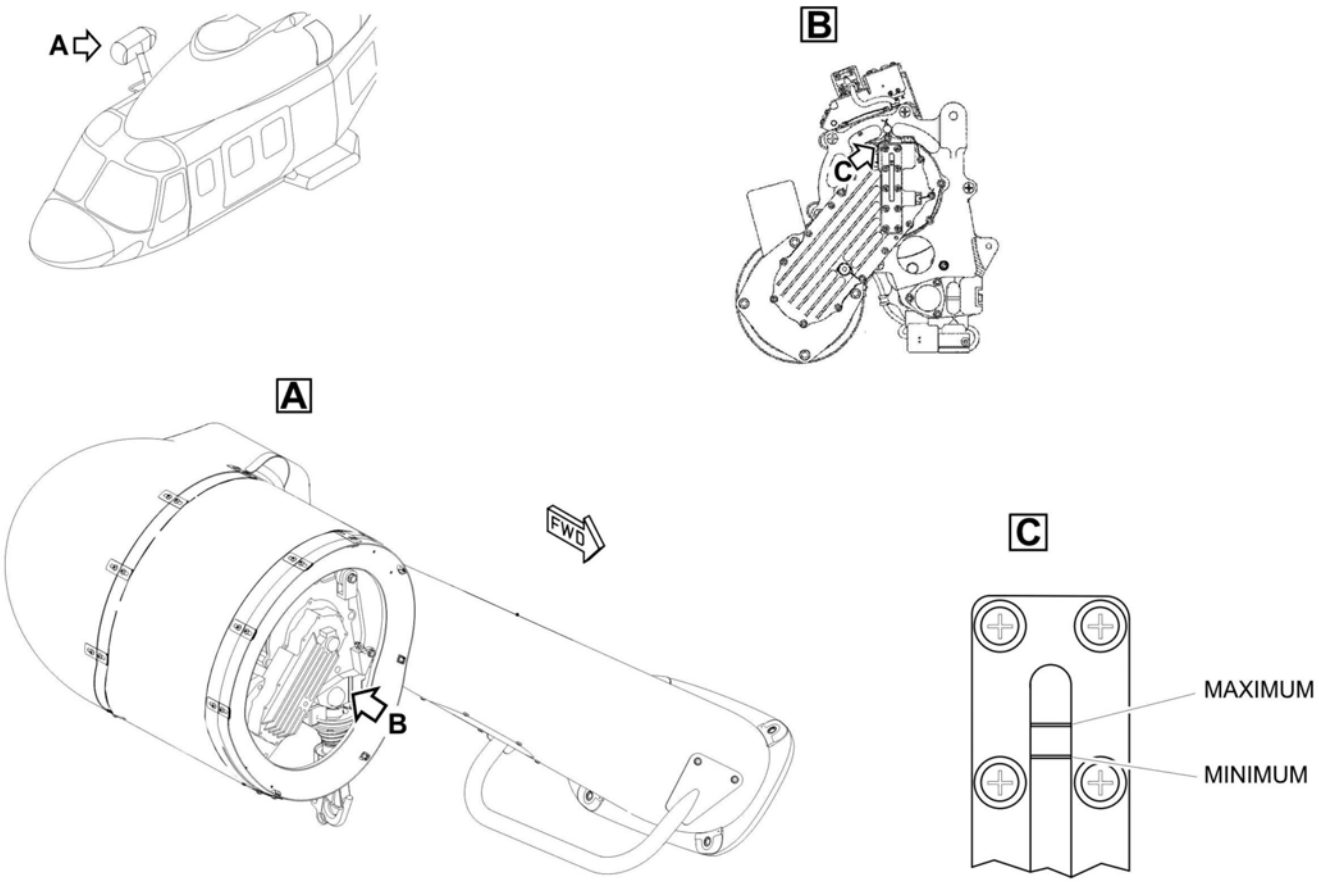
- A flange with four holes to attach the mount to the helicopter structure on one side.
- The holes to attach the hoist and the forward and aft frames on the other side.
- A shackle and an access cover for the connectors, for connection to the helicopter electrical system in the bottom.

The three hoist cowlings are attached to the forward and aft frames. The aft frame has a support with a bracket for the installation of the connectors of hoist light system.
The aft cowling contains the hoist light and its support.



ICN-39-A-259100-G-A0126-00251-A-01-1

MAIN COMPONENTS LOCATION (SHEET 5 OF 6)



ICN-39-A-259100-G-A0126-00252-A-01-1

MAIN COMPONENTS LOCATION (SHEET 6 OF 6)

PILOTS HOIST CONTROL PANEL

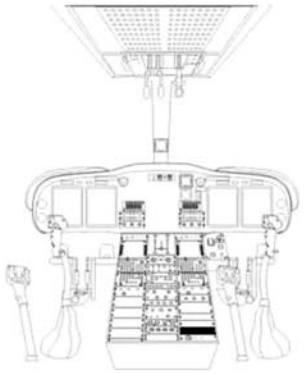
Controls And Indicators

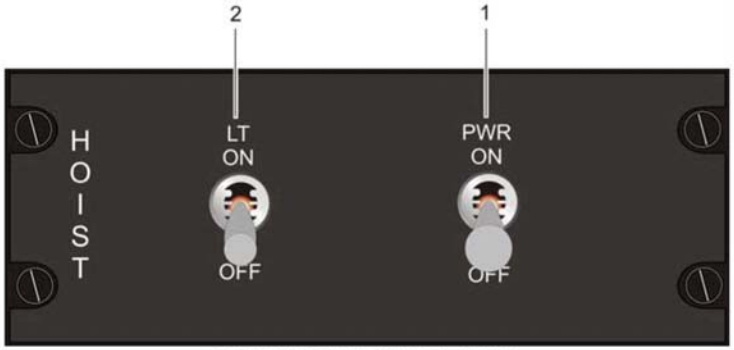
1 PWR Switch:

- ON ..... The system is energized.
- OFF ..... The system is de-energized.

2 LT Switch:

- ON ..... The hoist light system is energized.
- OFF ..... The hoist light system is de-energized.





ICN-39-A-259100-G-A0126-02923-A-01-1

CONTROLS AND INDICATORS (SHEET 1 OF 5)

HOIST OPERATOR CONTROL PANEL

Controls And Indicators

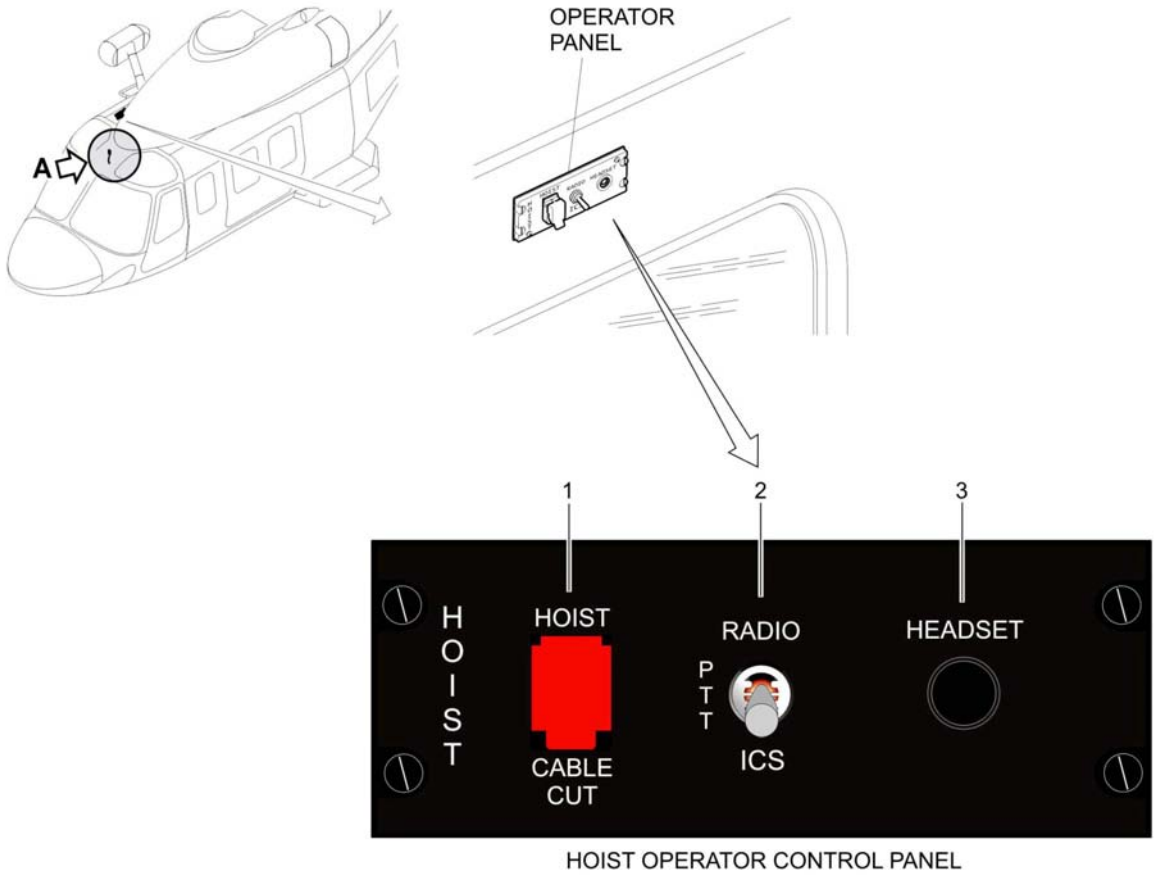
1 HOIST CABLE CUT:

- Guard down ..... The cable cutter is safe.
- Guard up ..... The cable cutter is armed for operation.
- Pushbutton ..... When pushed, the cable cutter is operated to cut the load cable in an emergency.

2 PTT Switch:

- Radio ..... The hoist operator can speak and hear with an operator on the ground through the radio.
- ICS ..... The hoist operator can speak and hear with the pilot and copilot.

3 HEADSET: ..... Receptacle connects the headset cord plug of the hoist operator.



HOIST OPERATOR CONTROL PANEL

ICN-39-A-259100-G-A0126-00258-A-01-1

CONTROLS AND INDICATORS (SHEET 2 OF 5)

PENDANT

Controls And Indicators

1 Cable payout display ..... Shows the length of the load cable unwound from the hoist in meters. When the system is first supplied with electrical power, the display shows 888. Then it shows the length of the unwound load cable (000 is shown if the cable is full-in).

2 SAR mode ..... Provision.

3 Direction/Speed control thumb wheel:

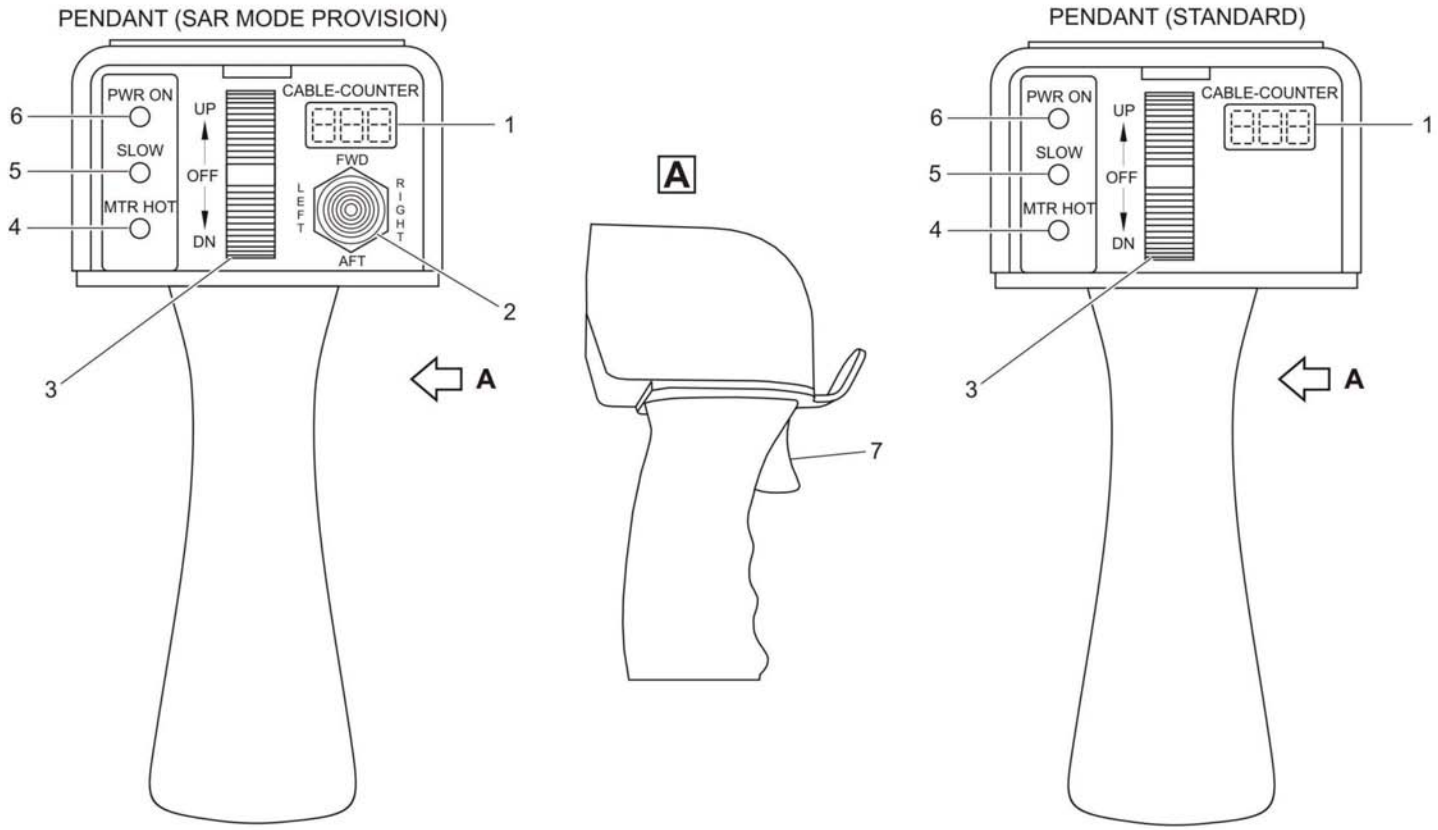
- OFF ..... The hoist is in the rest position.
- UP ..... Operates the switch that controls the up movement of the load cable and sends a speed command signal to the variable speed controller.
- DOWN ..... Operates the switch that controls the down movement of the load cable and sends a speed command signal to the variable speed controller.

4 MTR HOT (caution light) ..... On when a motor overtemperature occurred.

5 SLOW (caution light) ..... On when the system is in the low speed mode.

6 PWR ON (advisory light) ..... On when the electrical power is applied to the system.

7 ICS trigger switch ..... When pushed, it connects the hoist operator headset to the ICS.



ICN-39-A-259100-G-A0126-00261-A-01-1

CONTROLS AND INDICATORS (SHEET 3 OF 5)

PILOT AND COPILOT COLLECTIVE GRIPS

Controls And Indicators

1 Pilot HOIST CUT switch:

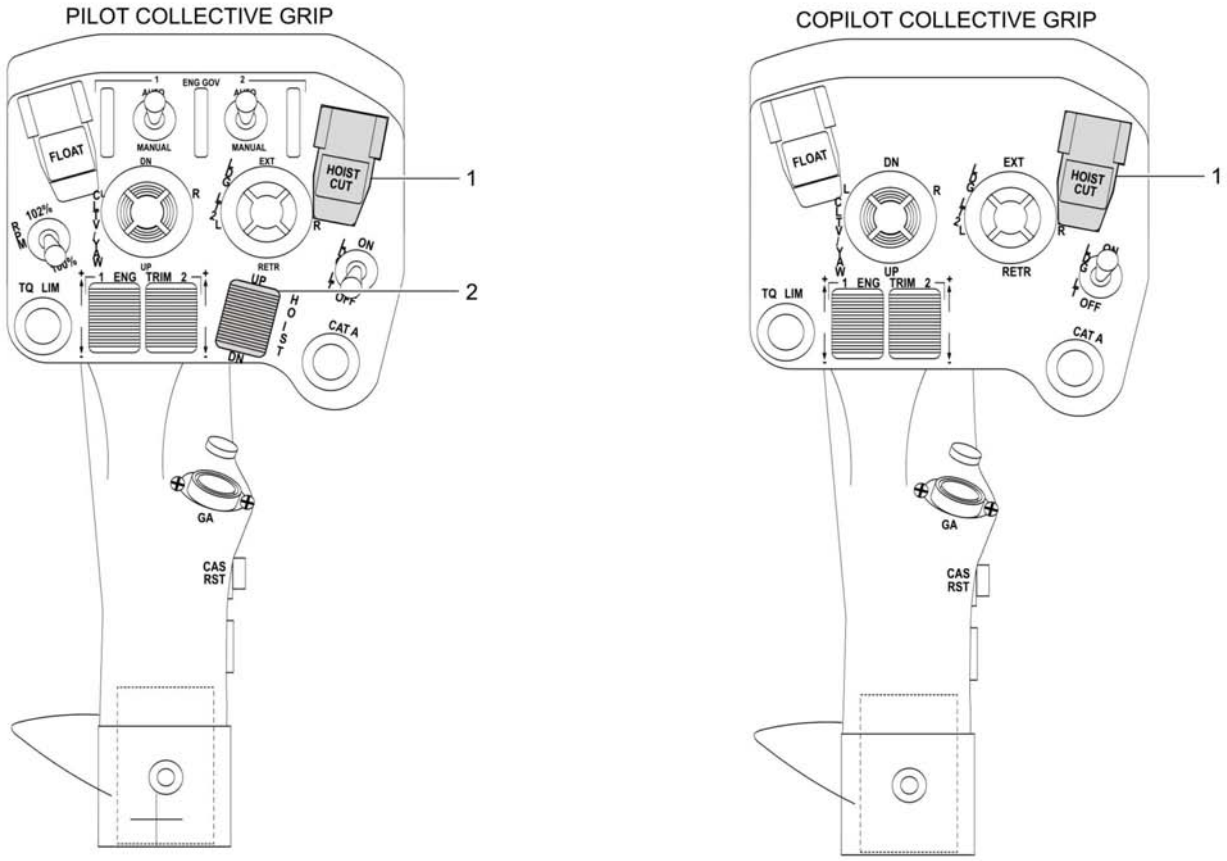
- Guard down ..... The cable cutter is safe.
- Guard up ..... The cable cutter is armed and ready for operation.
- Pushbutton ..... When pushed in an emergency, the cable cutter is operated to cut the load cable.

NOTE A

The switch can be operated also when the overload sensor disconnects the HOIST control switch.

2 HOIST control switch:

- UP ..... Controls the up movement of the load cable at constant speed.
- DN ..... Controls the down movement of the load cable at constant speed.



ICN-39-A-259100-G-A0126-00262-A-01-1

CONTROLS AND INDICATORS (SHEET 4 OF 5)

HOIST MFD INDICATION

The Hoist indication on the MFD are in the following:

1	HOIST CBL FOUL (caution message):	Shown when there is a cable foul condition on the hoist drum.
2	HOIST CUT ARM (caution message):	Shown when the cable cutter is armed and ready for operation.
3	HOIST ON (advisory message):	Shown when the rescue hoist system is ready for operation.

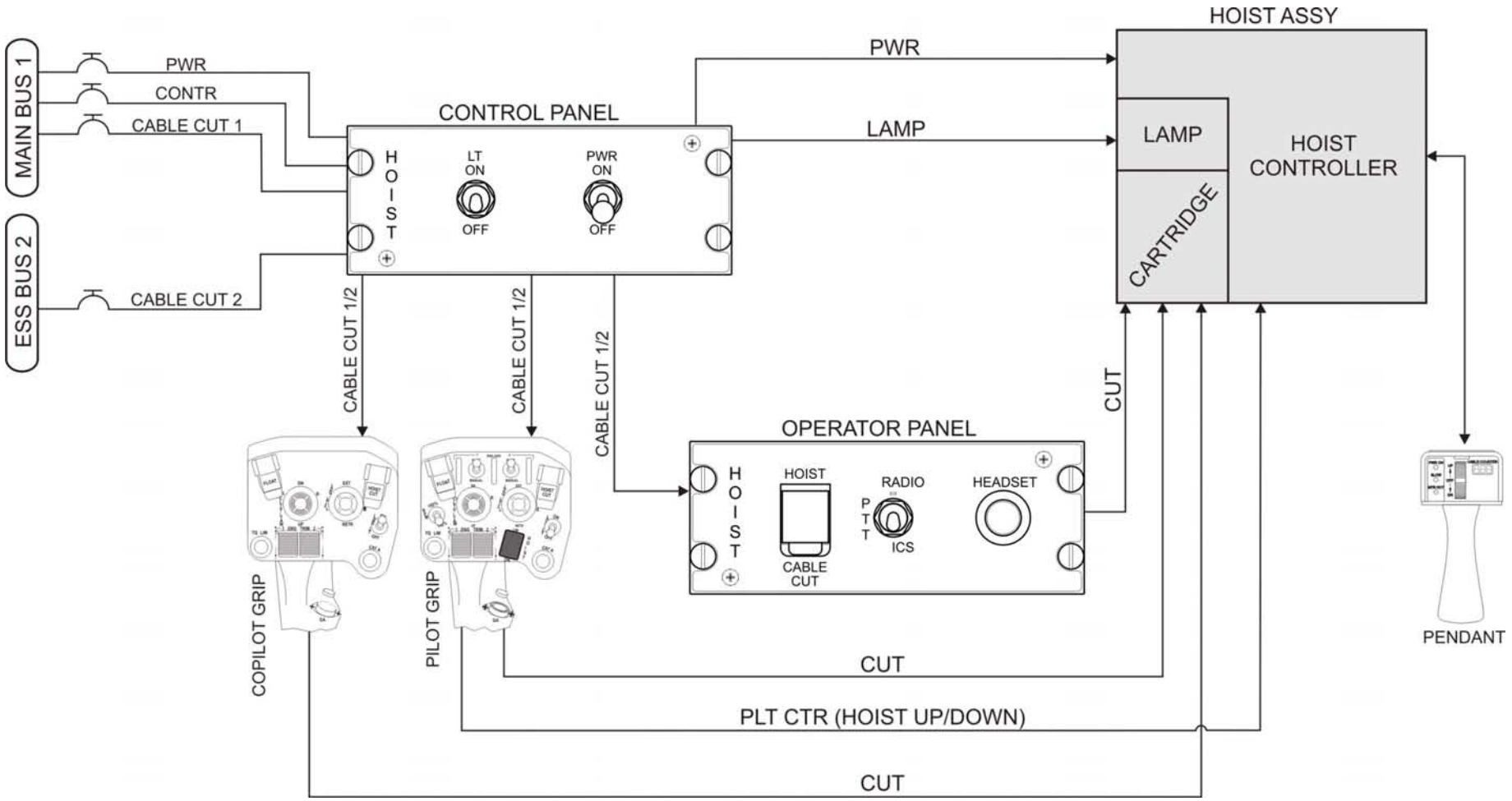


ICN-39-A-259100-G-A0126-00263-A-01-1

CONTROLS AND INDICATORS (SHEET 5 OF 5)

SYSTEM OPERATION

The hoist unit contains 245 ft (75 m) of usable cable and may lower/lift a maximum load of 272 Kg (600 lb).
Cargo hoisting and lowering can be controlled by the HO through the remote pendant thumb wheel which provides variable cable speeds. The maximum cable speed is 45 m/min (150 ft/min) with maximum load. The pilot can also control the operation at a fixed cable speed of 30 m/min (100 ft/min) through the hoist control switch on the collective grip. Both controls automatically slow down and stop the electric motor at the cable extremes. The Pilot control overrides the HO control.
The hoist is provided with a cable foul protection system which automatically stops the motor if the cable is not correctly wound onto the drum and displays the HOIST CBL FOUL caution message on the MFD.
An electrical cable cutter system is available to the Pilot and Copilot, via a guarded switch on the collective grips and to the HO, via a guarded switch on the operator HOIST control panel mounted in the cabin.
The hoist kit also comprises a manual cable cutter installed under seat and a right side cabin door step protection pad to prevent damage to the hoist cable if it contacts the step.
In the event of a failure of the electrical cable cutter system, the cable may be cut with the manual shears accessible to the HO.
The HO Control Panel, is used to control the HO communication with other crew members. With HO-ICS switched to ICS the HO can use either VOX or the PTT trigger on the Pendant to speak on the aircraft intercom.
With the switch selected to RADIO the HO can transmit on COM3 or COM4 only, using the Pendant HO PTT trigger.



ICN-39-A-259100-G-A0126-00264-A-01-1

RESCUE HOIST SYSTEM OPERATION

FUNCTIONAL DESCRIPTION

The rescue hoist system is used for rescue operations and to lift and lower cargo loads in areas where the helicopter cannot land.
The system is supplied with electrical power from:

- The 28 V dc main 1 bus bar through the HOIST PWR, HOIST CONTR and HOIST CABLE CUT 1 circuit breakers
- The 28 V dc ess 2 bus bar through the HOIST CABLE CUT 2 circuit breaker.

It is usually the hoist operator that operates the hoist rescue system through the control-pendant. When the system is first supplied with electrical power, the cable payout display on the control pendant shows 888 momentarily. Then the cable payout display shows the length of the unwound load cable (000 if the load cable is full-in).
When the variable speed controller receives the reel-in, or reel-out command signal, it sends a signal to the motor to lift or lower the load cable. The motor torque moves the gear train, which moves the level wind mechanism, the tension roller and the drum. This causes the load cable to wind or unwind.
When the variable speed controller receives the down-signal, the load cable unwinds. The speed increases when the hoist operator moves the thumb wheel farther from the center. When the load cable is almost full-out, the speed must be decreased. When the down-intermediate speed-limit switch operates, the hoist speed decreases to 11,4 thru 21 m/min (38 thru 68 fpm) until the full-out limit switch is touched. At this time 3 1/2 to 4 1/2 wraps of load cable are wound on the drum.
When the cable is unwound, and the orange paint comes into view on the load cable, there are approximately 4,3 m (14 ft) of cable travel before the full-out limit is touched.
When the variable speed controller receives the up-signal, the load cable winds. The speed increase when the hoist operator moves the thumb wheel farther from the center. The maximum permitted speed of the load cable is 65,4 m/min (215 fpm) at 113 kg (250 lb) load. When the load cable is almost full-in, the speed must be decreased. When the up-intermediate speed-limit switch operates, the hoist speed decreases to 11,4 thru 21 m/min (38 thru 68 fpm) until the full-in limit switch is touched. At this time the hoist operation is stopped.
When the cable is wound, and the orange paint starts to go out of view, there are approximately 7,6 m (25 ft) of cable travel before the full-in limit is touched.
The load cable winds on the drum. It must wind on the drum equally. If there is separation between the cable wraps, or the wraps are one on the top of the other, a cable foul condition occurs. When the load cable winds on the last permitted wrap, the cable-foul indicator-switch operates mechanically and stops the hoist. The cable foul must be removed on the ground, and the cause of the foul must be found before the hoist can be used again.
While the load cable winds slowly to the full-in position, the hook must be moved manually. This makes sure that the spring in the bumper is compressed sufficiently and the hook does not move when the helicopter is in flight.
The MTR HOT caution light comes on the control pendant when the motor is too hot and damage to the hoist can occur. When the light is on, the operation of the hoist must be stopped. Then the hoist motor must stay off until its temperature is the same as the ambient temperature. Usually 30 minutes are sufficient to have a cool motor. But the ambient conditions and the time during which the light stay on before the hoist operation was stopped can change this period.
When it is necessary to release the load in an emergency, the pilot, the copilot or the hoist operator can cut the load cable:

- The pilot or the copilot do this through the HOIST CUT switch on the collective stick grip.
- The hoist operator does this through the HOIST CABLE CUT switch on the operator panel.

The status of the rescue hoist system is shown on the MFD through the messages that follow:

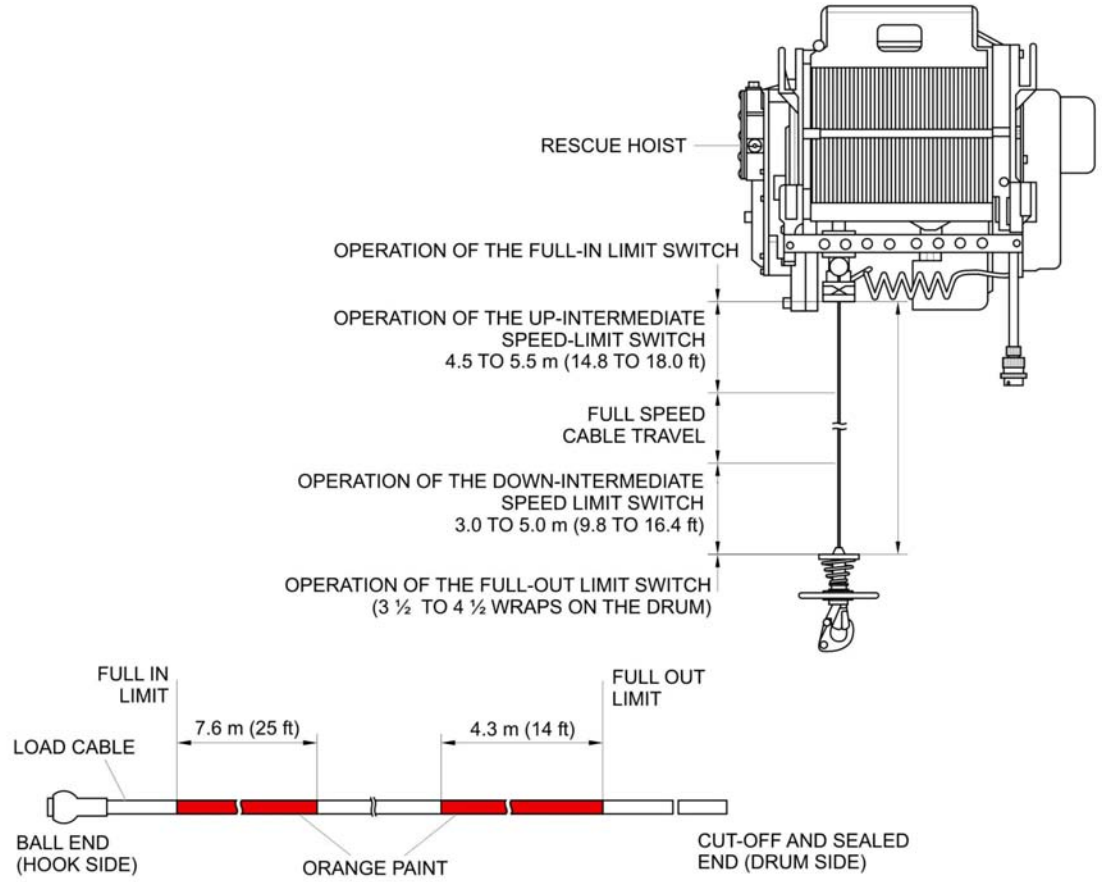
- HOIST ON advisory message.
- HOIST CUT ARM caution message.
- HOIST CBL FOUL caution message.

NOTE B

- During hoist operations the Pilot must fly manually.
- Take Off and Landing with a suspended load on hoist is prohibited.

WARNING

- HOISTING OR LOWERING AN EMPTY LITTER IN OPEN POSITION IS PROHIBITED. AN EMPTY LITTER SUSPENDED FROM HOIST IN OPEN POSITION CAN OSCILLATE UNCONTROLLABLY IN ROTOR DOWNWASH AND CAN FLY UPWARD, STRIKING FUSELAGE OR TAIL ROTOR.
- PRIMUS HF 1050 RADIO WITH THE CABIN DOOR OPEN, HF 1050 RADIO TRANSMISSIONS ARE PROHIBITED.
- WEATHER RADAR MUST BE IN SBY OR OFF DURING HOIST OPERATIONS.
- RADIO EQUIPMENT V/UHF FLEXCOMM II WITH THE CABIN DOOR OPEN, FLEXCOMM II RADIO TRANSMISSIONS ARE PROHIBITED.
- DURING HOIST OPERATIONS THE PILOT MUST FLY MANUALLY OR, IF 4 AXIS FLIGHT DIRECTOR IS INSTALLED (SUPPLEMENT 34) AND HOV, RHT, OR ALT MODES ARE ENGAGED, THE PILOT MUST FLY ATTENTIVE.



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RESCUE HOIST SCHEMATIC DIAGRAM

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SECTION 92 - CARGO HOOK

DESCRIPTION

The primary components of the cargo hook system are:
The cargo hook
The pressure cartridge assembly
The support frame
The cargo hook control panel
The load indicator.

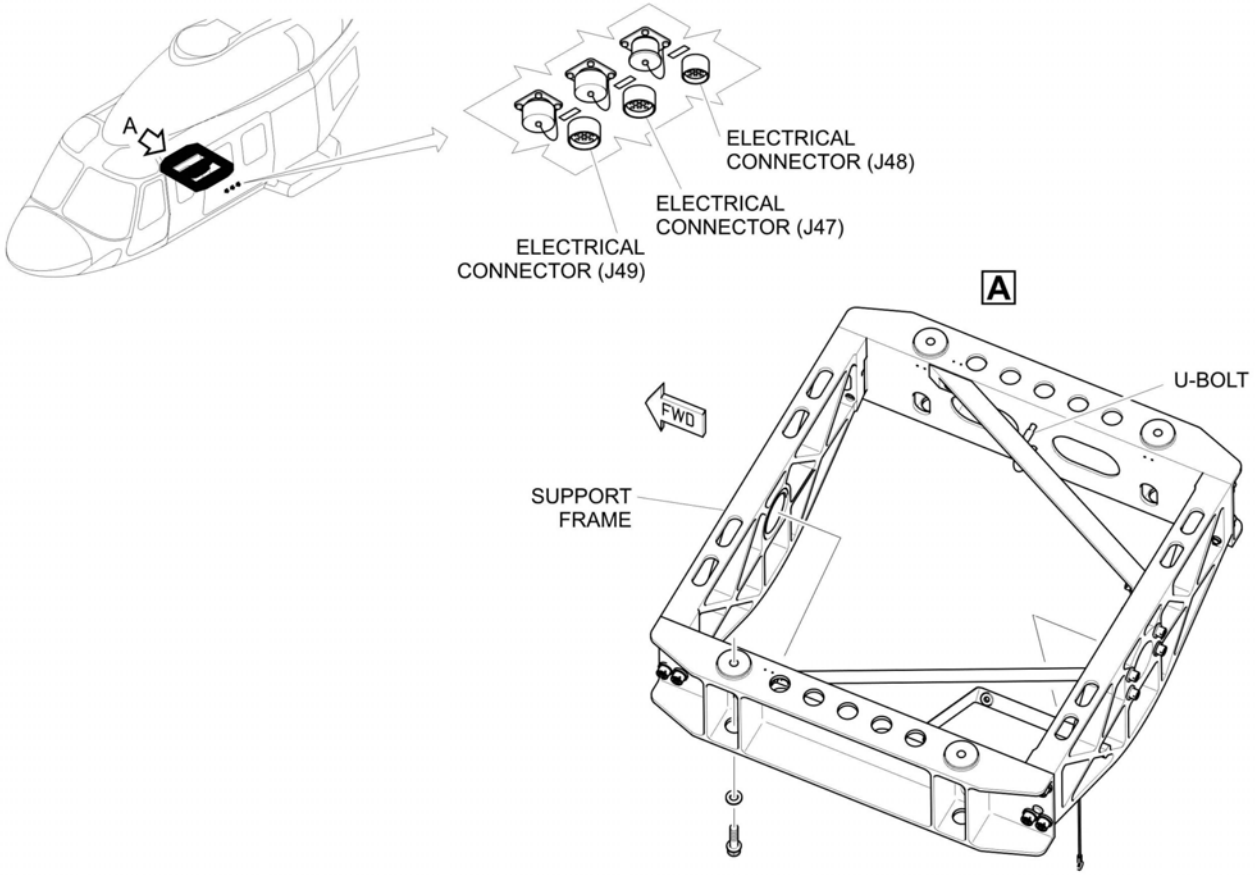
CARGO HOOK

The cargo hook is installed on the support frame attached to the bottom of the fuselage, near the center of gravity of the helicopter.
The cargo hook (MP15) has:

- A load arm
- A keeper
- A latch arm
- A latch assembly
- A lock
- Two side plates
- Two mounts
- A load cell
- An electrical box.

The load arm holds the loads that are hung below the helicopter. The keeper prevents an accidental release of the loads. Two springs safety the keeper in the closed position.

The load arm is attached to the two side plates with a nut and a bolt around which it turns when the load is released. The latch arm keeps the load arm in the closed position.
The latch arm is attached to the side plates with a bolt around which it turns. The latch assembly keeps the latch arm closed.
The latch assembly is attached to the side plates with a bolt around which it turns. The lock keeps the latch assembly closed.
The lock is attached to one of the side plates with a bolt around which it turns. The lock is attached also to the shaft of the solenoid.
The two mounts are installed at the forward and aft ends of the cargo hook. These mounts are attached to the bushings in the support frame. The forward mount has a connector for the connecting cable. This cable connects the pressure cartridge to the electrical connector (J49), installed on the left side of the bottom fuselage. The load cell is installed on the aft mount. An electrical cable connects the load cell to the electrical connector (J48) on the left side of the bottom fuselage.
The electrical box is installed on the left side of the cargo hook. The electrical box includes a solenoid, the safety switch, the “hook open” switch and a diode. The electrical box is connected to the electrical connector (J47) installed on the left side of the fuselage bottom through an electrical cable.



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COMPONENTS LOCATION (SHEET 1 OF 4)

PRESSURE CARTRIDGE ASSEMBLY

The pressure cartridge assembly is installed on the top of the cargo hook. The components of the pressure cartridge assembly are an electrically-fired pressure cartridge, a cartridge housing and a piston. The pilot or the copilot can operate the pressure cartridge in an emergency through the CARGO REL EMERG switch installed on cargo hook control panel.

SUPPORT FRAME

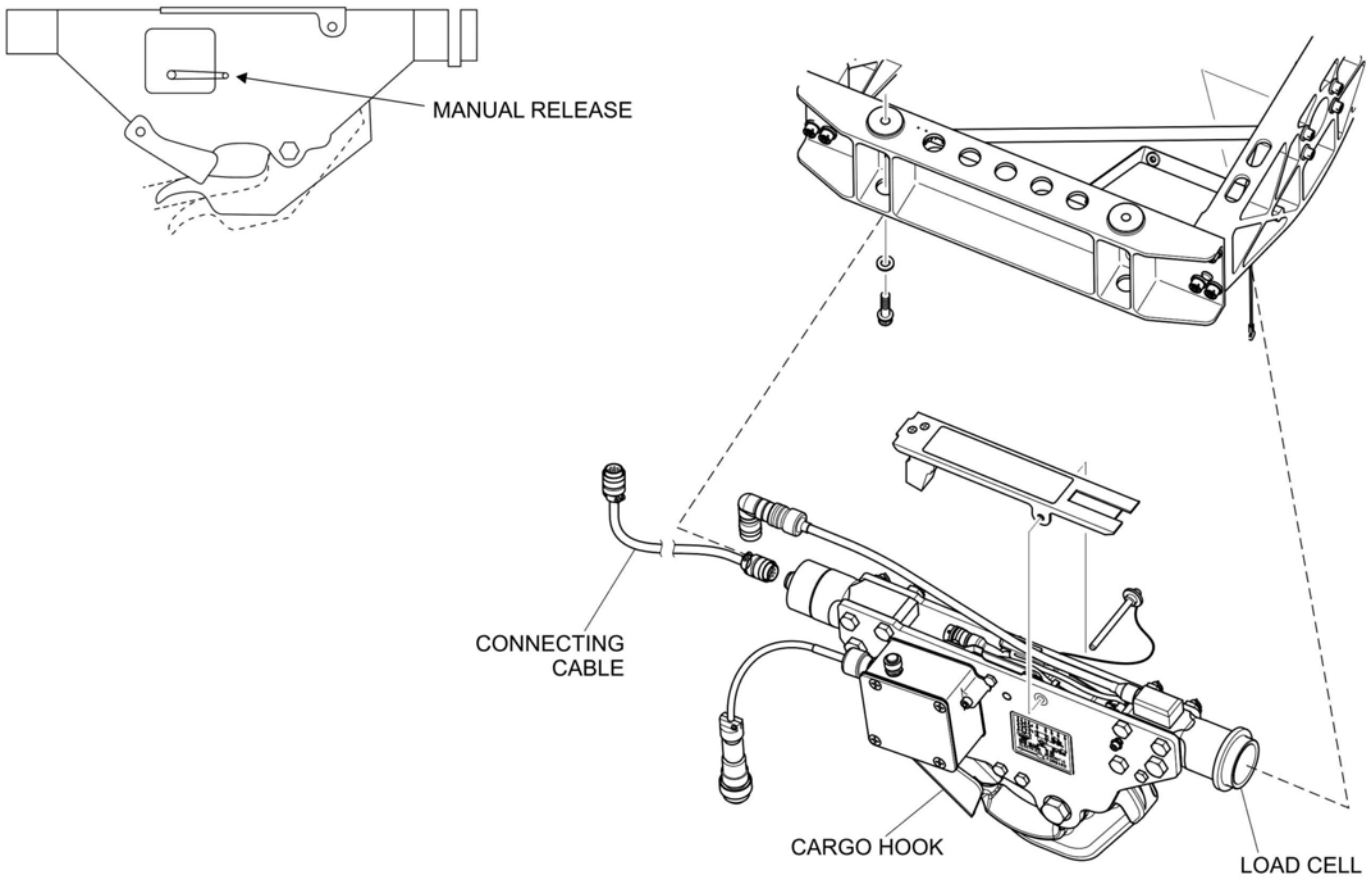
The support frame is a rectangular aluminium frame, which has the forward and aft flanges and the left and right channels.
The forward and aft flanges are the support of the cargo hook. The forward flange has an internal bushing that holds the forward mount of the cargo hook. The bushing can be moved out of the frame for easy mount installation. The aft flange has a split bushing for easy installation of the aft mount of the cargo hook.
The left and right channels are attached to the helicopter with four bolts. A U-bolt is installed on the right channel to safety the cargo hook in the stowed position when the hook is not used.
The left and the right struts are installed between the left and right channels and the aft flange to increase the strength of the frame.

CARGO HOOK CONTROL PANEL

The cargo hook control panel is installed on the interseat console and is attached with four quick-release fasteners. Two switches are installed on the front panel. An electrical connector is installed on the rear of the control panel.

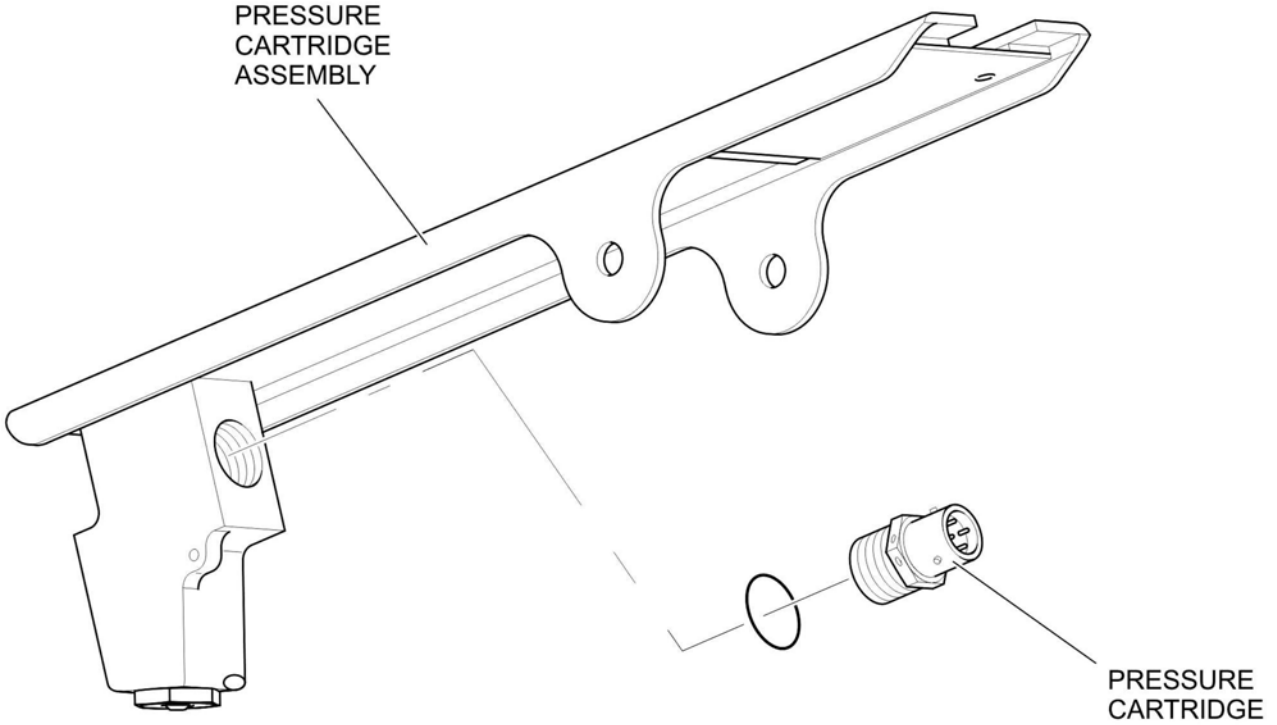
LOAD INDICATOR

The load indicator is installed on the instrument panel. The load indicator shows the load on the cargo hook in increments of 10 lb or 10 kg. An electrical connector is installed on the rear of the load indicator.



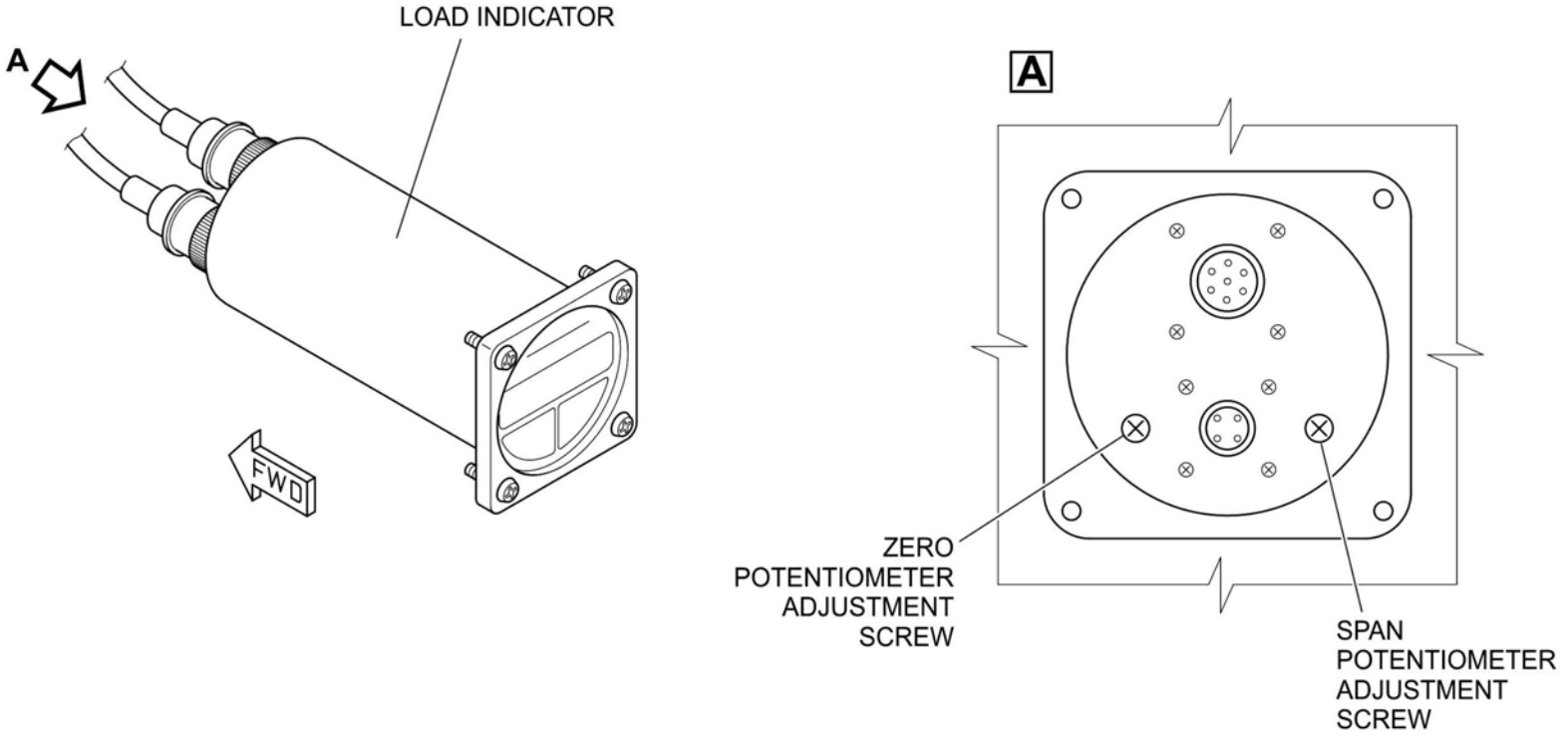
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COMPONENTS LOCATION (SHEET 2 OF 4)



ICN-39-A-259200-G-A0126-00285-A-01-1

COMPONENTS LOCATION (SHEET 3 OF 4)



ICN-39-A-259200-G-A0126-00286-A-01-1

COMPONENTS LOCATION (SHEET 4 OF 4)

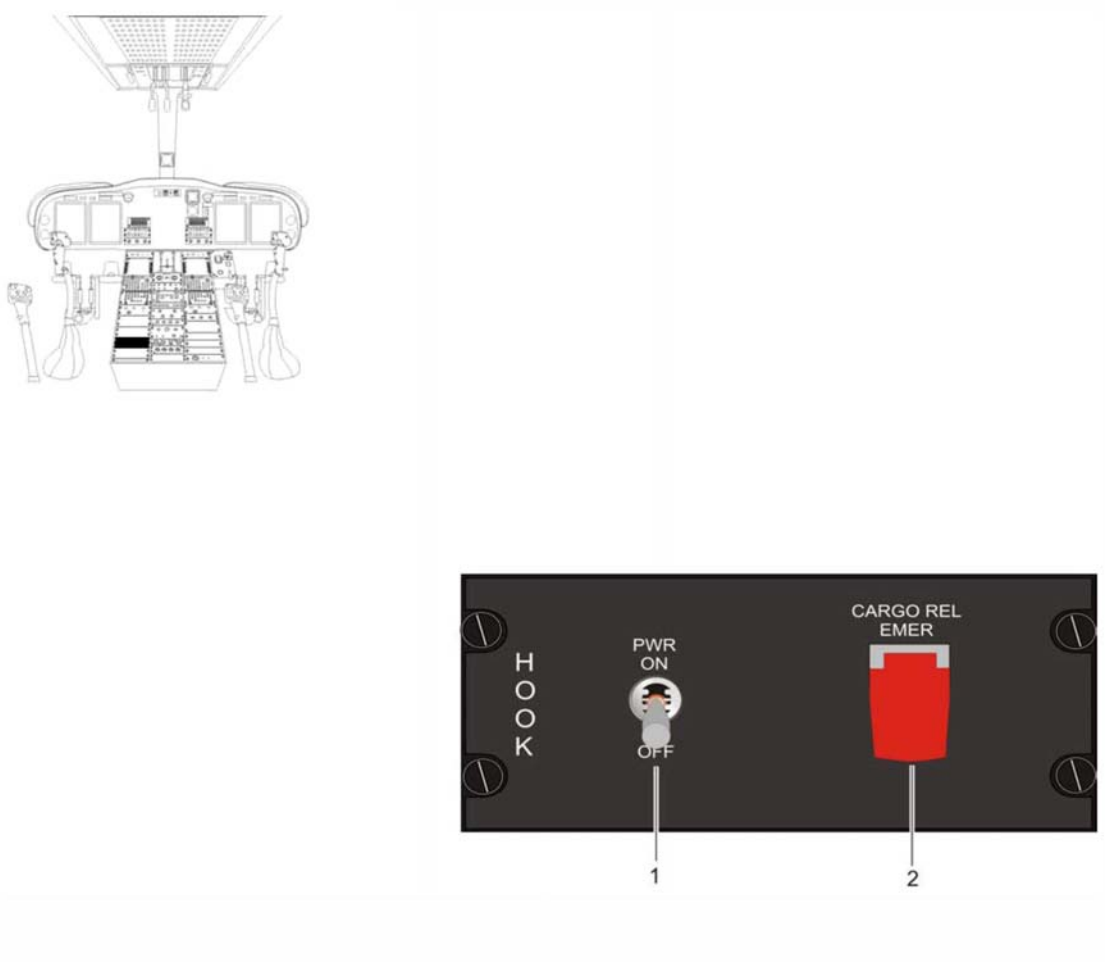
CARGO HOOK CONTROL PANEL CONTROLS AND INDICATORS

1 PWR switch:

- ON ..... The system is energized.
- OFF ..... The system is de-energized.

2 CARGO REL EMERG switch:

- Guard down ..... the fire circuit of the pressure cartridge is safe.
- Guard up ..... the fire circuit of the pressure cartridge is armed.
  When pushed in an emergency the pressure cartridge is fired.



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CARGO HOOK CONTROL PANEL CONTROLS AND INDICATORS

LOAD INDICATOR CONTROLS AND INDICATORS

1 Load Indicator: ..... shows the weight of the load hung below the cargo hook.

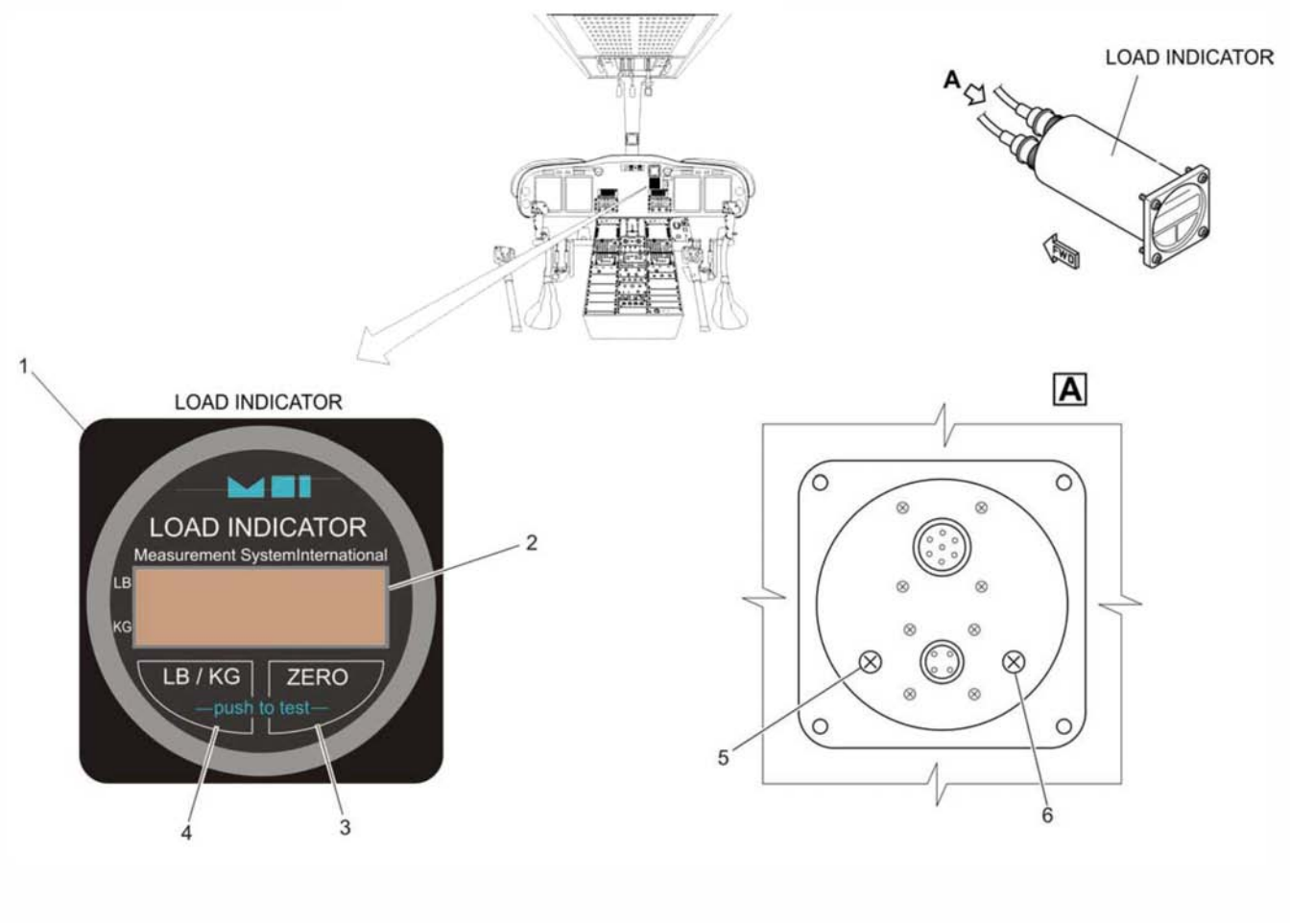
2 Display: ..... shows the weight sensed by the load cell in the unit of measure set with the LB/KG button.

3 ZERO button: ..... when pushed, resets the display to zero.

4 LB/KG button: ..... sets the display to show the weights in pound (lb) or in kilograms (Kg).

5 Zero potentiometer adjustment screw: .....
Rotating ..... when rotating adjust the zero indication of Load Indicator.

6 Span potentiometer adjustment screw: .....
Rotating ..... when rotating set the max value of 9000 lb (4082 Kg).



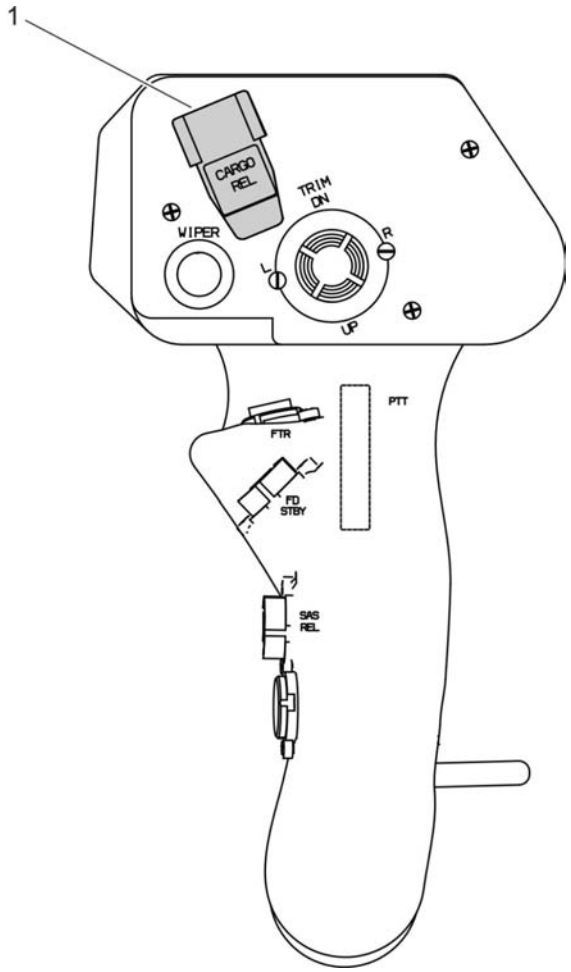
ICN-39-A-259200-G-A0126-02925-A-01-1

LOAD INDICATORS CONTROLS AND INDICATORS

PILOTS STICKS CONTROLS AND INDICATORS

1 CARGO REL switch:

- Guard Down ..... the solenoid circuit of the cargo hook is safe.
- Guard up ..... the solenoid circuit of the cargo hook is ready for operation. When pushed, the solenoid circuit is energized and the load arm of the cargo hook opens to release the load.



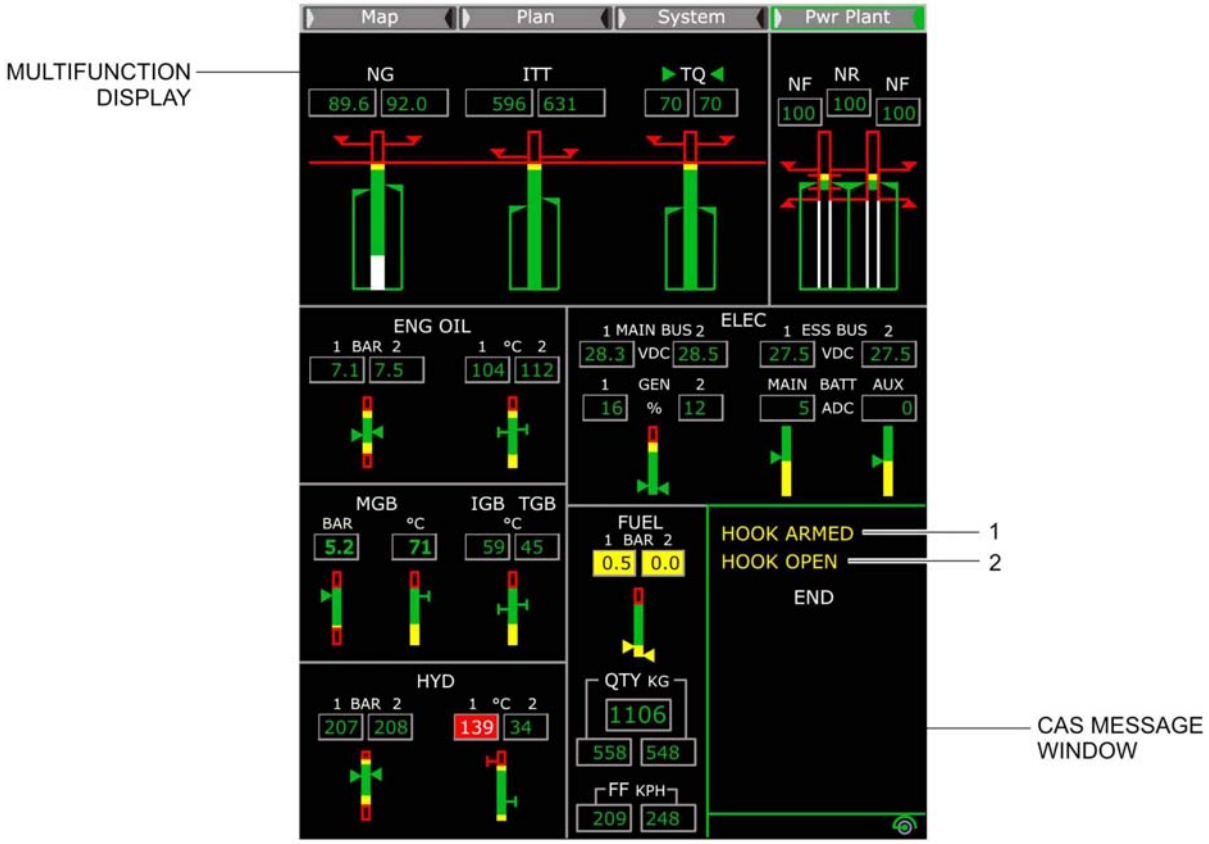
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PILOTS STICKS CONTROLS AND INDICATORS

CARGO HOOK CAPTIONS

In order to monitoring the unload operations some indicators are provided. The release operation are monitored to Pilot and Copilot via the MFD as following:

1	HOOK ARMED (caution message):	shows when the CARGO REL EMERG switch is ready to fire the pressure cartridge.
2	HOOK OPEN (caution message):	shows when the “hook open” switch closes because the solenoid is energized and starts its travel.



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CARGO HOOK CAPTIONS

SYSTEM OPERATIONS

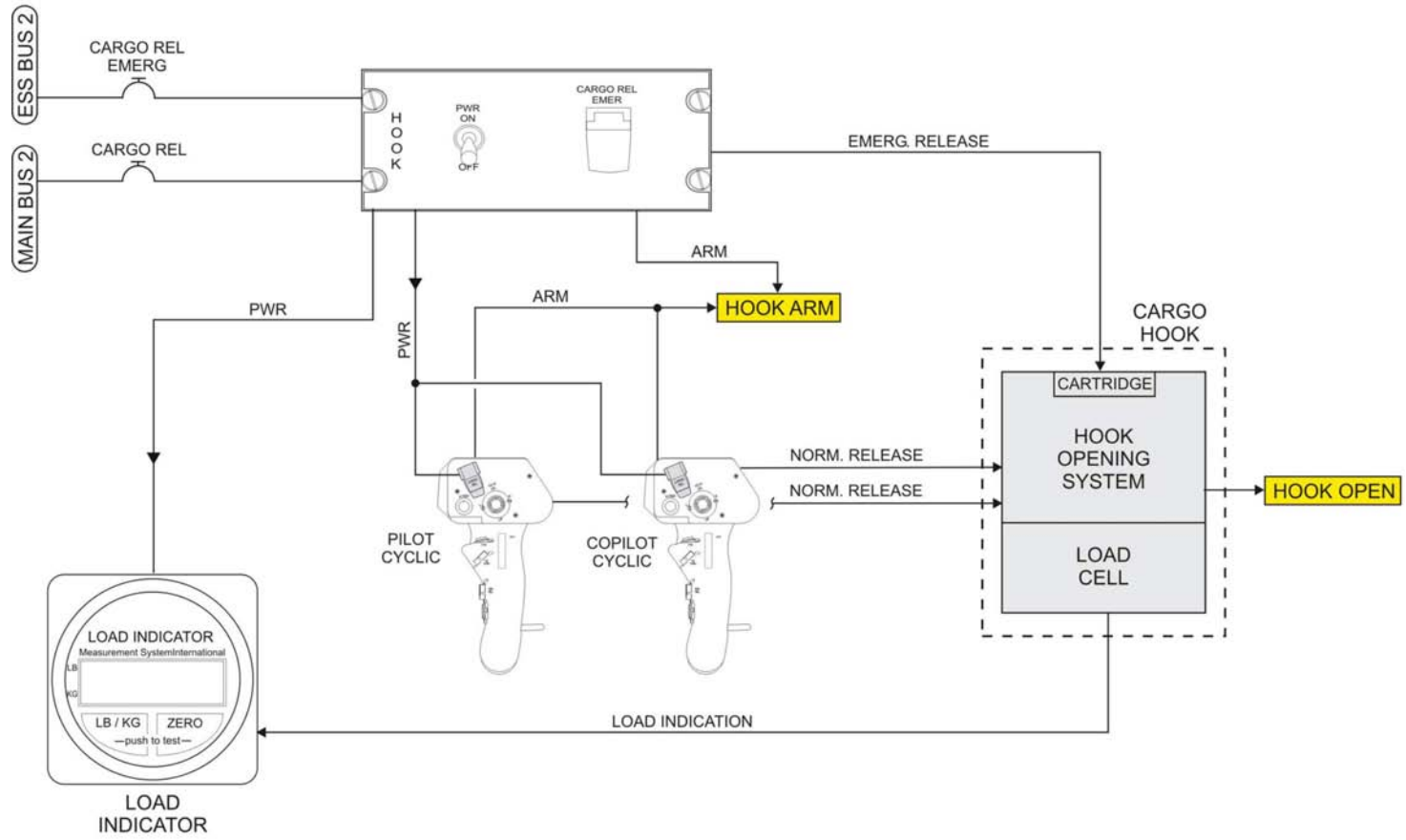
The cargo hook system is used to move loads of a maximum weight of 2200 kg (4850 lb) that are hung below the helicopter.
The system is supplied with electrical power from:

- The 28 VDC MAIN BUS 1 bus bar through the CARGO REL circuit breaker
- The 28 VDC ESS 2 BUS bar through the CARGO REL EMERG circuit breaker.

The electrical power is applied to the cargo hook system when the PWR switch is moved to ON.
When the power is first applied to the system, the display of the load indicator shows 18880 momentarily. Then the display shows 0 ±10 kg (±20 lb), if no load is on the cargo hook. If not, the pilot or the copilot must push the ZERO button on the face of the load indicator below the display. When the load cell senses the load hung below the cargo hook, the display of the load indicator shows the weight of the load.
There are three different procedures to release the load:

- The normal release
- The emergency release
- The manual release.

To do the normal release of a load, the pilot or the copilot can start the procedure in flight. The pilot or the copilot must first set the PWR switch, which supplies the electrical power to the solenoid in the cargo hook, to OFF (Fig 1). The pilot or the copilot must move the guard of the CARGO REL switch to up. When the switch guard is up, the MFD shows the HOOK ARMED caution message. Then the pilot or the copilot can push momentarily the CARGO REL switch on the cyclic stick handgrip to release the load.
When the CARGO REL switch is pushed, the solenoid becomes energized. When the solenoid is energized, the solenoid shaft, on which the lock is installed, turns and disengages the lock. With the lock disengaged, the latch assembly releases the latch arm. With the latch arm released, the weight of the load causes the load arm to open and the load moves-away from the load arm. A spring in the cargo hook sets the load arm to its closed and latched position after the release of the load.
When the lock turns up, a cam on the solenoid shaft operates the safety switch which stops the electrical power to the solenoid circuit (Fig 1). This prevents damage to the solenoid from too much current flow. After the load release the load arm goes back to its closed position. If, at this time, the pilot holds the CARGO REL switch pushed, the solenoid is energized again. The safety switch prevents continuous power-on and power-off of the solenoid if the pilot holds the switch pushed. The diode, included in the solenoid circuit, is also a protection, as it increases the time necessary to energize again the solenoid.
When the lock is disengaged from the latch assembly, the cam on the solenoid shaft operates the “hook open” switch (Fig 1). This switch sends a signal that causes the MFD to show the HOOK OPEN caution message.
If the normal release could not be completed satisfactorily, the pilot can do the emergency release. The pilot can operate the CARGO REL EMERG switch on the cargo hook control panel. When the switch is pushed, the pressure cartridge is fired.
The energy from the cartridge pushes a piston down to disengage the lock. With the lock disengaged, the latch assembly releases the latch arm. With the latch arm released, the weight of the load causes the load arm to open and the load moves away from the load arm.
The personnel on the ground can do the manual release of the load. A hand-operated release handle, external to the cargo hook, is connected to the lock. When the handle is turned-up, it disengages the lock. With the lock disengaged, the latch assembly releases the latch arm. With the latch arm released, the weight of the load causes the load arm to open and the load moves away from the load arm.



ICN-39-A-259200-G-A0126-00291-A-01-1

SCHEMATIC DIAGRAM

REFERENCE LITTERATURE

In section 5° of the AW139 RFM, the supplement No. 13 provides additional information to the system.
In the AMPI Retirement Lives, there are important information about the Penalty Factor that must be apply when the Cargo Hook is used.

The information contained in this document supplements the information of the Basic Flight Manual.
For limitations, procedures and performance data not contained in this Supplement, consult the basic Rotorcraft Flight Manual

SUPPLEMENT 13

CARGO HOOK OPERATIONS

P/N 3G2592F00111

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ICN-39-A-259200-G-A0126-00292-A-01-1

CHAPTER 26

FIRE PROTECTION

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SECTION 00 - FIRE PROTECTION

FIRE PROTECTION – GENERAL

The purpose of the fire protection system is to detect for overheating or fire in the engine compartments and presence of smoke in the baggage compartment.
The fire protection system comprises

- the engine fire detection system
- the engine fire extinguishing system
- the baggage compartment smoke detection

FIRE DETECTION SYSTEM – GENERAL

The purpose of the engine fire detection system is to detect high temperature, fire or hot gas leakage into the engine compartments.
The system comprises a heat fire detector installed in each engine compartment.
When fire or hot gas leak are detected, the indicating system permits to identify the location of the fire.

FIRE DETECTION SYSTEM – MAIN COMPONENTS

HEAT FIRE DETECTOR

The heat detector is made of a continuous firewire element and a responder.

FIREWIRE

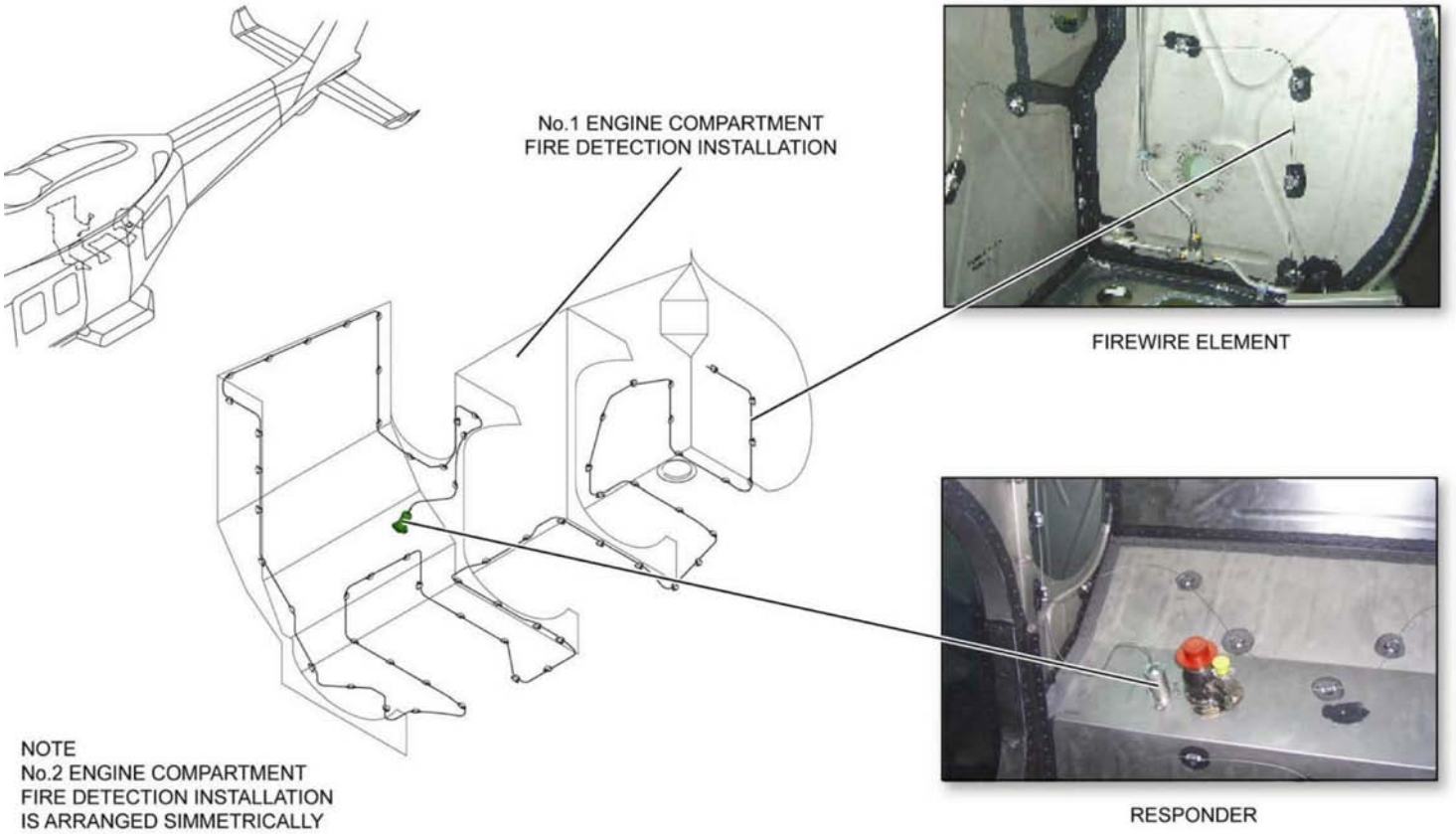
The firewire is the sensing element and consists of:

- a stainless steel tube cointaining the gas (helium) under pressure;
- a core material (wire) located inside the tube.

The core material is a gas absorption material impregnated with hydrogen.
One end of the firewire is connected to the responder and the other end is sealed.

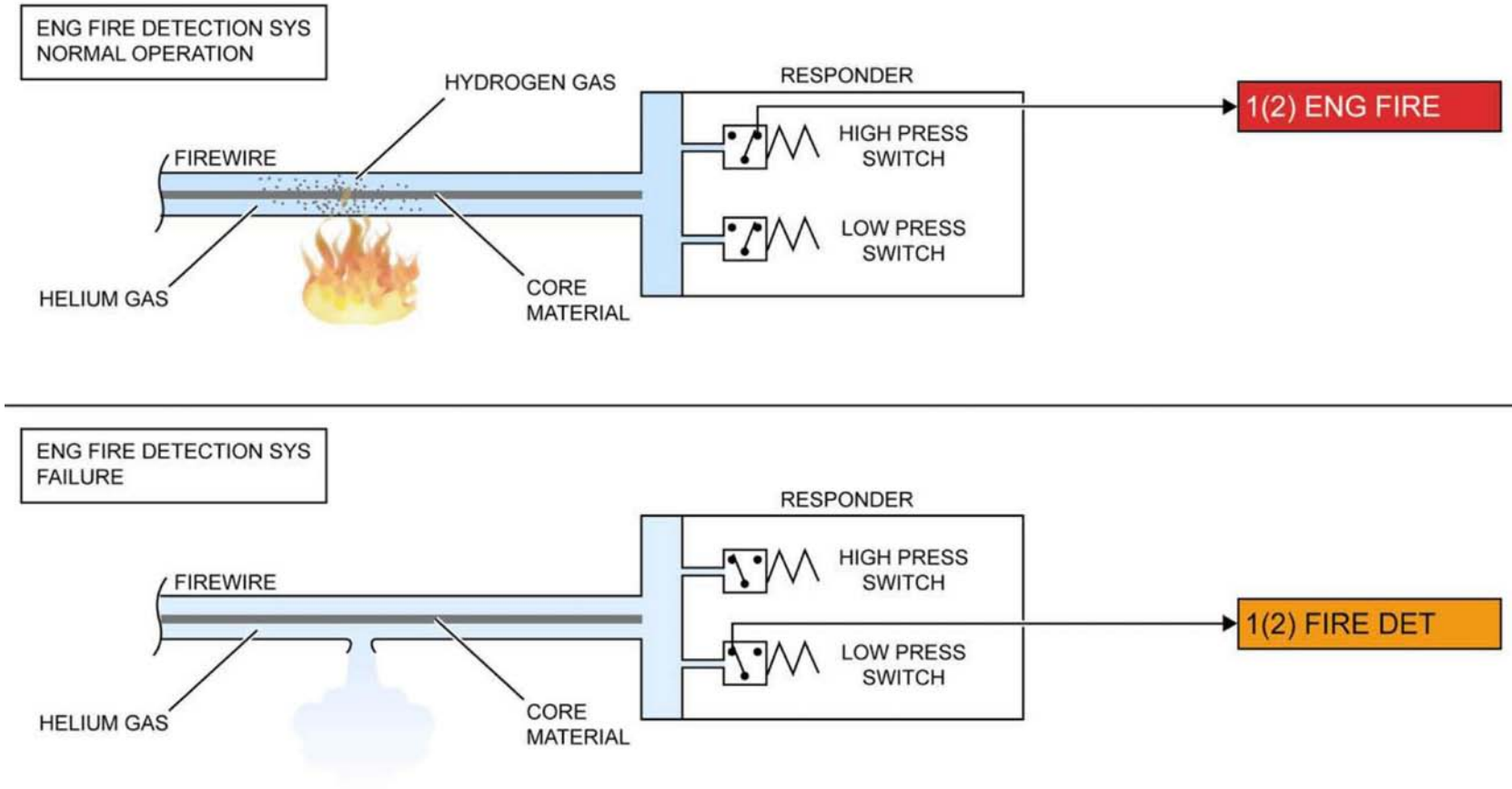
RESPONDER

The responder contains two pressure switches: the high pressure switch and the low pressure switch.
The high pressure switch, in case of fire or hot gas leak, generates the ENG FIRE message.
The low pressure switch, in case of failure of the engine fire detection system (like for example, the breaking of a tube) generates the FIRE DET message.



ICN-39-A-260000-G-A0126-01762-A-01-1

FIRE DETECTION SYSTEM – MAIN COMPONENTS



ICN-39-A-260000-G-A0126-01763-A-01-1

RESPONDER – SCHEMATIC DIAGRAM AND OPERATIONS

FIRE EXTINGUISHING - GENERAL

The purpose of the fire extinguishing system is to protect the engine compartment from fire. The fire extinguisher system is composed by two identical and interconnected sub-systems one for each engine compartment.
Each system comprises a fire extinguisher bottle and double-check T-valves.
The extinguisher bottles are operated manually by the pilot on the FIRE EXTING control panel. When the pilot operates the system, the double-check T-valves make sure that only one of the two extinguisher bottles is operated. If the pilot operates the system a second time, the second bottle will be operated.
In addition two hand-held fire extinguishers are located in the cockpit and in the passenger cabin.

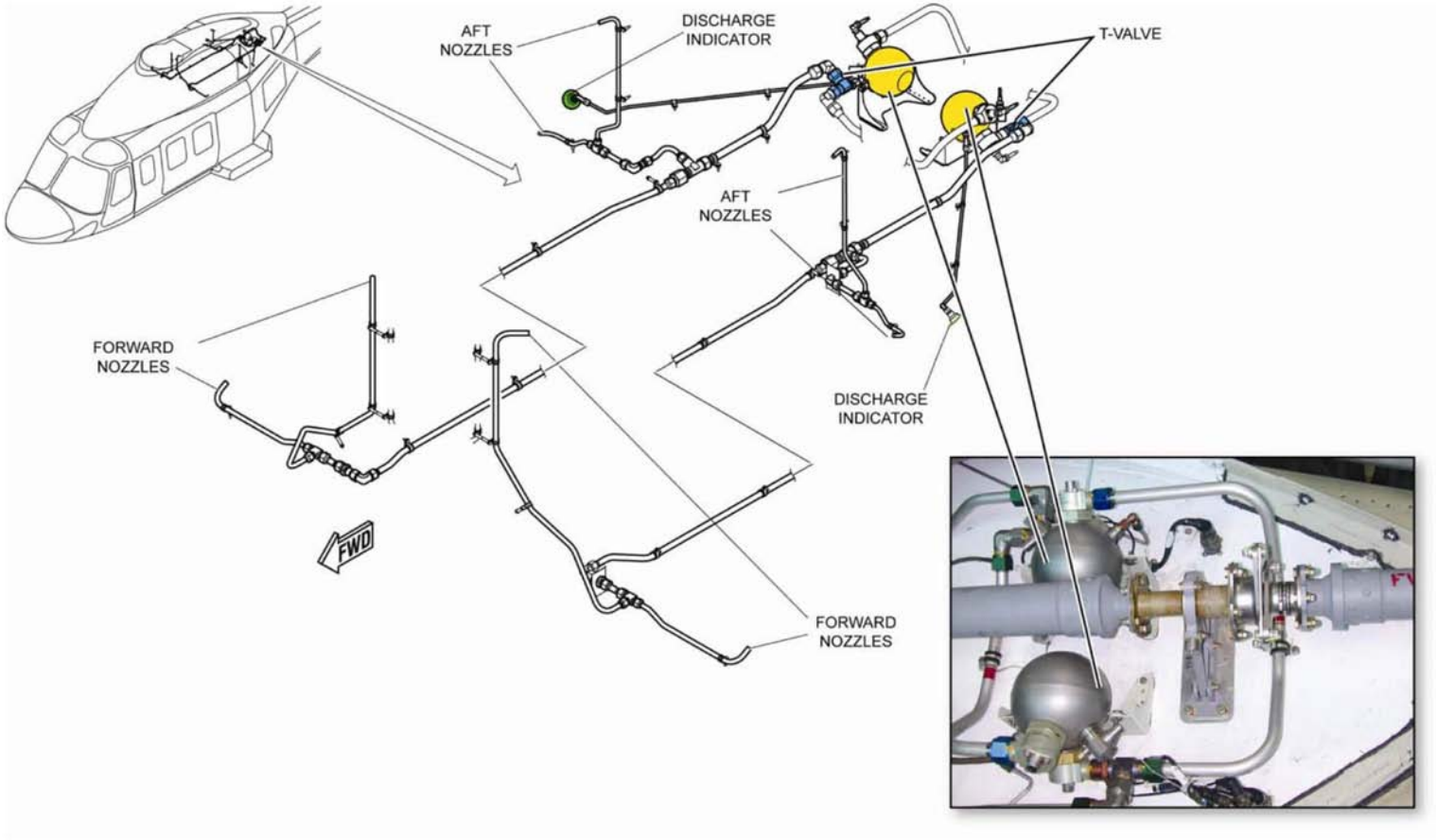
FIRE EXTINGUISHING SYSTEM – MAIN COMPONENTS

EXTINGUISHING BOTTLE

There are two 72 cubic inches extinguishing bottles filled with halon and pressurized with nitrogen gas. The bottles are installed on either side of the engine compartment and are cross-connected so that the content of any one of the two bottle can be discharged into any one engine bay and/or both bottles can be discharged into any one engine bay.
Each bottle is provided with a device which acts as a primary safety relief device. In case of overpressure, the halon agent is fully discharged outside the helicopter through the discharge indicator.
Upon discharge, the outer green disc is fired and a red circular band is displayed providing a visual indication during on ground inspection.

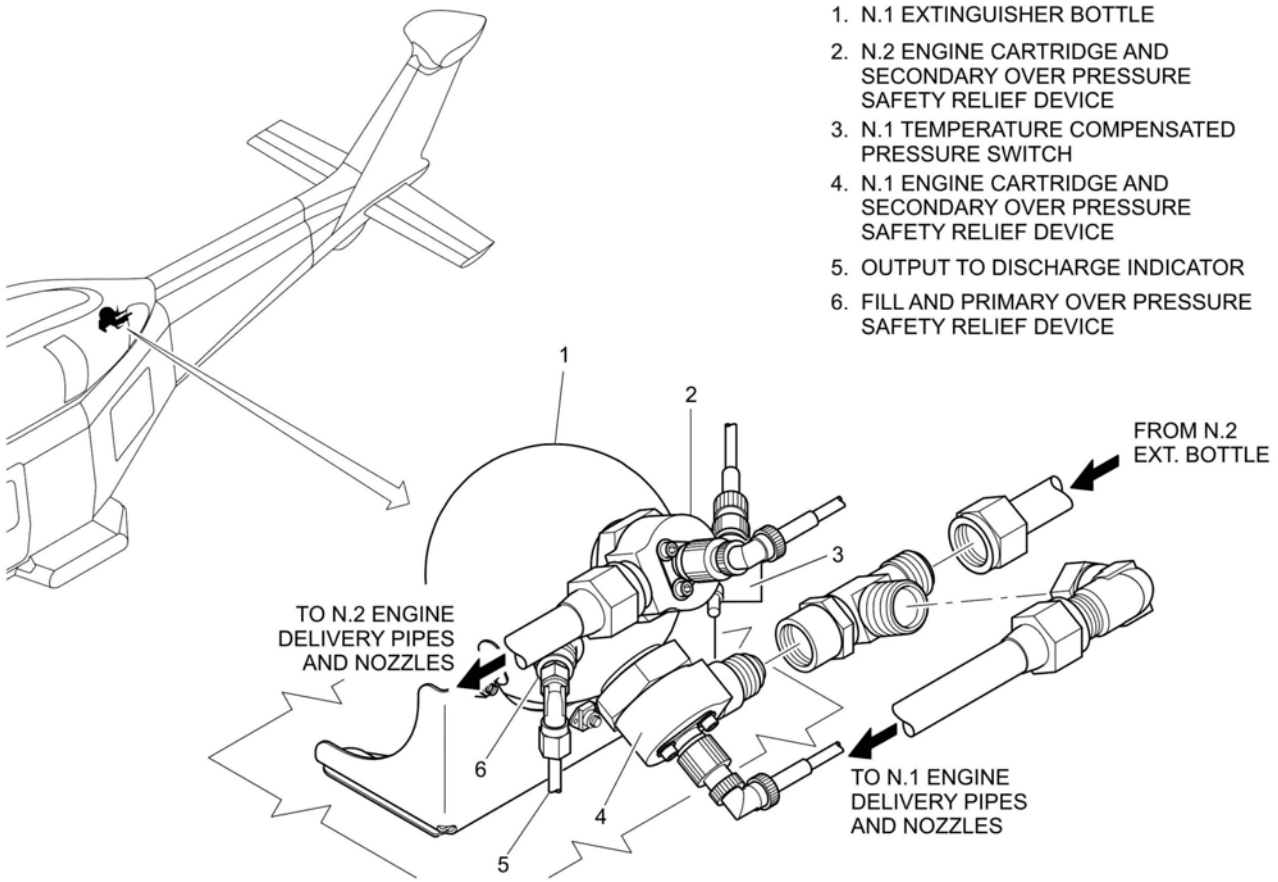
DOUBLE CHECK T-VALVE

Double-check T-valves interconnect the two sub-systems allowing to discharge halon into anyone of the two engine compartments.



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FIRE EXTINGUISHING – MAIN COMPONENTS



ICN-39-A-262000-G-A0126-00213-A-01-1

N.1 EXTINGUISHER BOTTLE LOCATION

FIRE PROTECTION - CONTROLS AND INDICATORS

1 BAG indicator

- FIRE (red) illuminated …… smoke is detected into the baggage compartment

2 ENG 1 guarded indicator / push-button switch

- FIRE (red) illuminated …… fire is detected in the engine no. 1 compartment
- push-button pressed ..... arms (ARM light illuminated) or de-arms the fire extinguisher system outlets towards the no.1 engine bay
- ARM (amber) illuminated ….
  - The engine no.1 outlet cartridges on bottles no.1 and no.2 are armed
  - The no.1 fuel SOV (Shut-Off-Valve) is closed
  - The no.1 engine bleed air SOV is closed
  - the no.1 engine particle separator SOV is closed (if installed/optional)

3 FIRE EXTING bottle 3 position selection switch

- central ..... None extinguisher bottle is selected
- BTL 1 ..... the no.1 extinguisher bottle is selected and the shot to extinguish fire take place
- BTL 2 ..... the no.2 extinguisher bottle is selected and the shot to extinguish fire take place

NOTE A

If fire persists after the first shot, a second shot is available moving the switch to the other BTL position.

4 ENG 2 indicator guarded indicator / push-button switch

- FIRE (red) illuminated …… fire is detected in the engine no. 2 compartment
- push-button pressed ..... arms (ARM light illuminated) or de-arms the fire extinguisher system outlets towards the no.2 engine bay
- ARM (amber) illuminated …
  - the engine no.2 outlet cartridges on bottles no.1 and no.2 are armed
  - the no.2 fuel SOV is closed
  - the no.2 engine bleed air SOV is closed
  - the no.2 engine particle separator SOV is closed (if installed/optional)

5 FIRE indicator

- FIRE (red) illuminated …… fire is detected in the engine no. 1 compartment

6 FIRE indicator

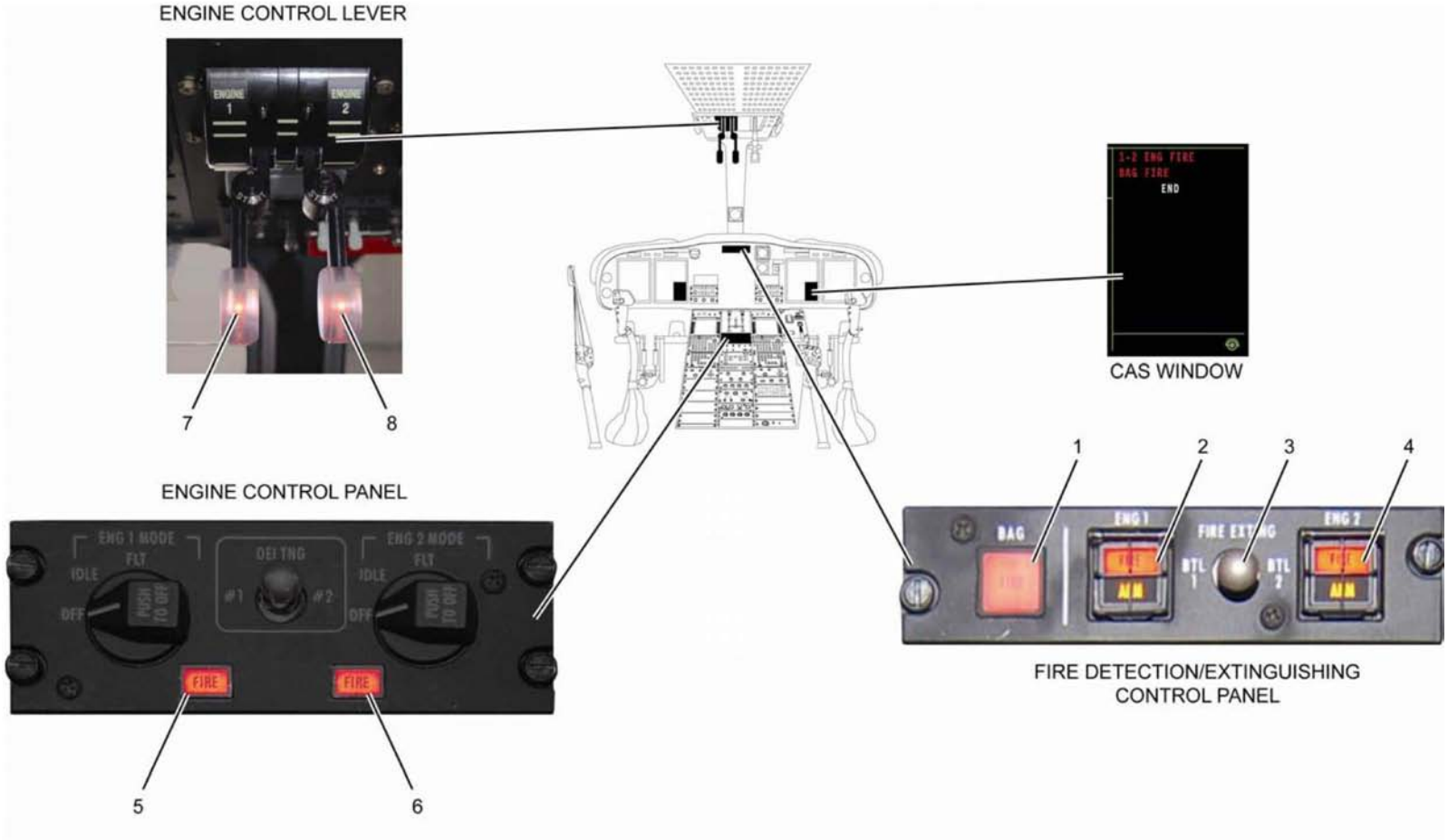
- FIRE (red) illuminated …… fire is detected in the engine no. 2 compartment

7 FIRE indicator

- (red) illuminated ..... fire is detected in the engine no. 1 compartment

8 FIRE indicator

- (red) illuminated ..... fire is detected in the engine no. 2compartment



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FIRE PROTECTION – CONTROLS AND INDICATORS

FIRE PROTECTION SYSTEM – TEST

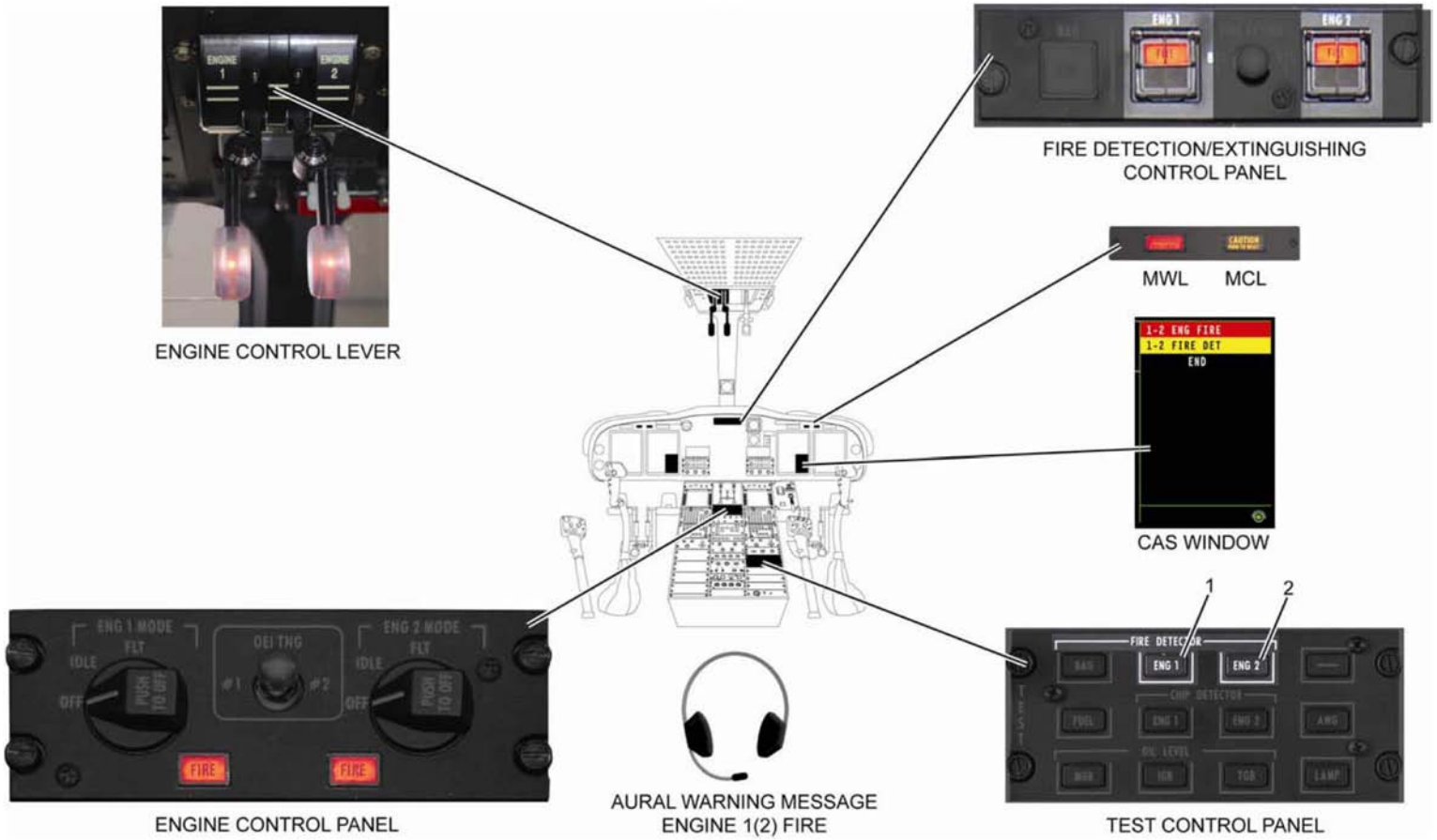
The test of the fire detection system is provided through the TEST panel.

1 FIRE DETECTOR ENG 1 push-button

- pressed .....
  - ENG 1 ECL fire illuminates red
  - ENG 1 FIRE illuminates red on the FIRE EXTING control panel
  - the MWL and MCL illuminate
  - FIRE illuminates red (left indicator) on the ENG control panel
  - 1 ENG FIRE (warning) and 1 FIRE DET (caution) are displayed in the CAS window
  - audio tone and ENGINE 1 FIRE voice warning message

2 FIRE DETECTOR ENG 2 push-button

- pressed .....
  - ENG 2 ECL fire illuminates red
  - ENG 2 FIRE illuminates red on the FIRE EXTING control panel
  - the MWL and MCL illuminate
  - FIRE illuminates red (right indicator) on the ENG control panel
  - 2 ENG FIRE (warning) and 2 FIRE DET (caution) are displayed in the CAS window
  - audio tone and ENGINE 2 FIRE voice warning message



ICN-39-A-260000-G-A0126-01766-A-01-1

FIRE PROTECTION SYSTEM – TEST

FIRE DETECTION – PRINCIPLE OF OPERATION

The firewire element takes into account two types of high temperature conditions:

- the first type is a general overheat condition. In this case if the temperature around the firewire element increases over a preset limit. As a consequence the sensor acts in accordance with the law of gases: if the volume of the firewire element is held constant, its pressure will increase as temperature increases. The helium gas in the tube exerts a pressure which closes the high pressure switch that generates the ENG FIRE message.
- the second type is a fire condition at a short section of the firewire sensor element. In this case the core material releases a large volume of hydrogen gas so that the internal pressure of the firewire element increases very quickly and the gas in the tube exerts a pressure which closes the high pressure switch that generates the ENG FIRE message. After the situation is corrected, the material reabsorbs the hydrogen and the system returns to a stand-by mode.

In case of leakage of the helium gas from the firewire element, the pressure will decrease and operates the low pressure switch which generates the FIRE DET message.

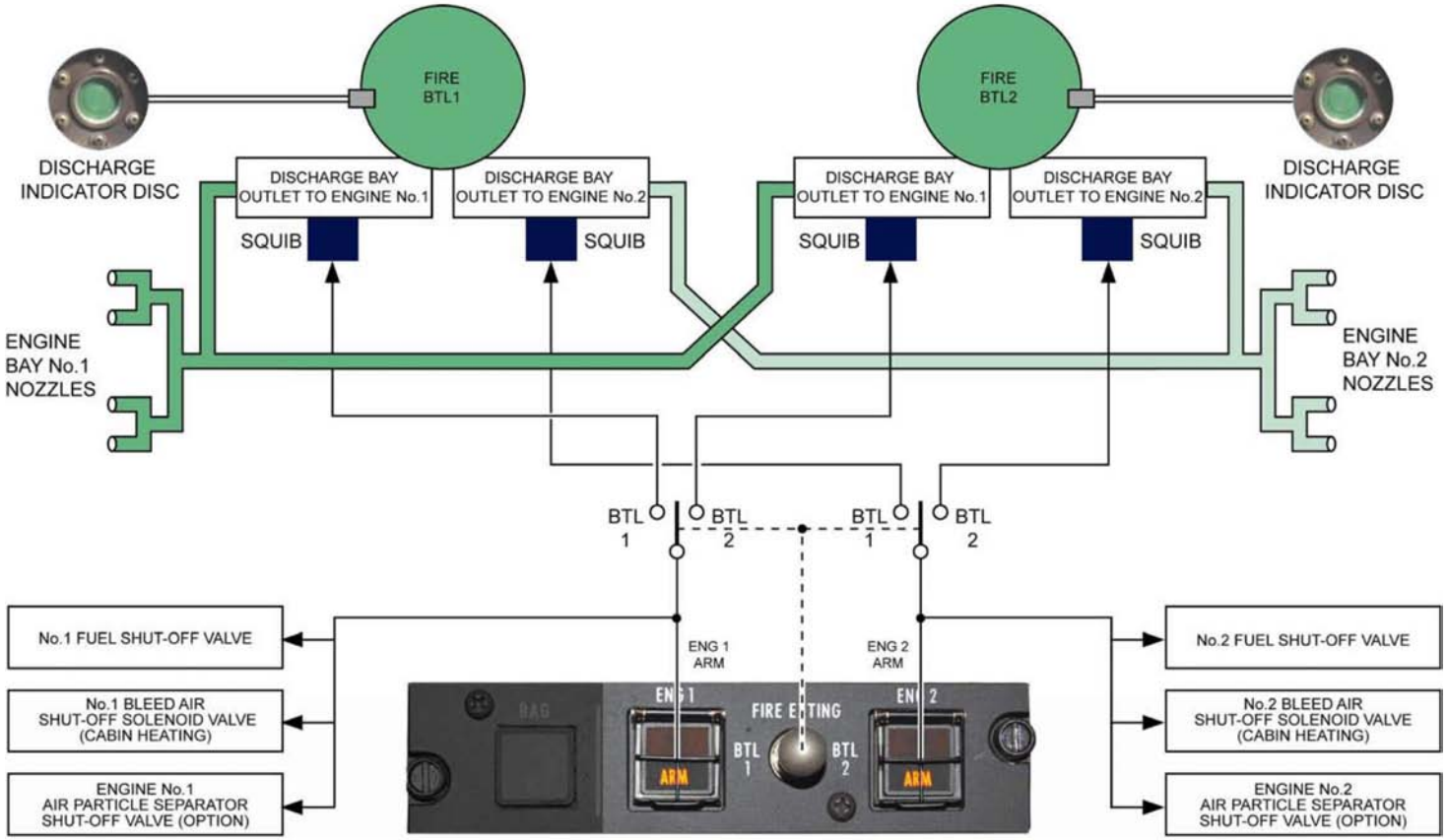
FIRE EXTINGUISHING – PRINCIPLE OF OPERATION

NOTE B

The command to extinguish fire is done via the FIRE switch on the FIRE EXTG panel that, after been pressed, illuminates the amber segment ARM.

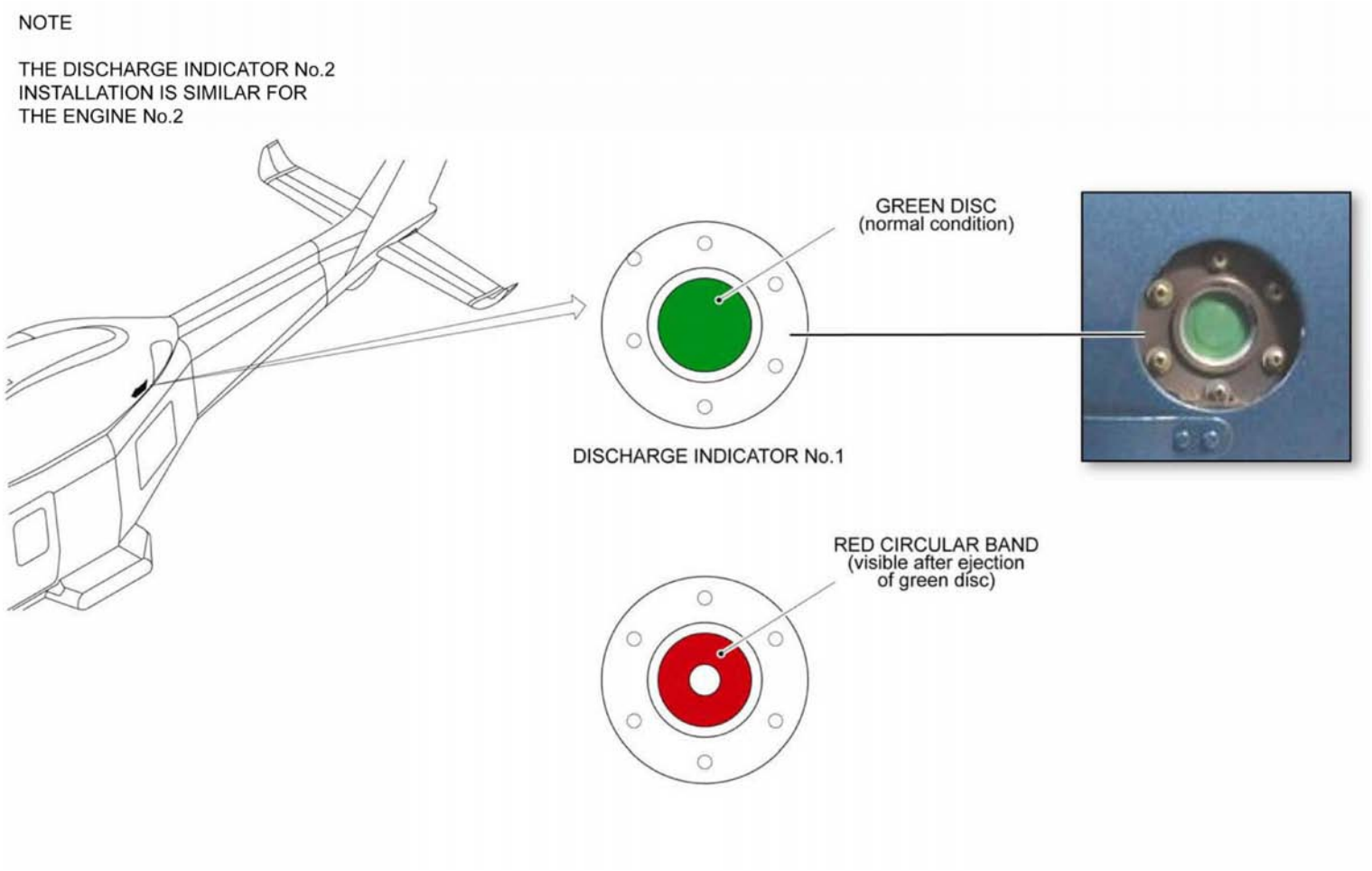
The extinction of the fire does not implies the automatic re-opening of the fuel shut-off valve automatically closed when the switch FIRE EXTING has been moved from the central position.

In case the first shot to extinguish fire fails, a second shot is available moving the switch to the second bottle indication.



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FIRE EXTINGUISHING – SCHEMATIC



ICN-39-A-260000-G-A0126-01768-A-01-1

DISCHARGE INDICATOR

BAGGAGE FIRE DETECTION SYSTEM - GENERAL

The purpose of the baggage fire detection system is to detect smoke in the baggage compartment.
The system comprises a smoke detector installed in the baggage compartment.

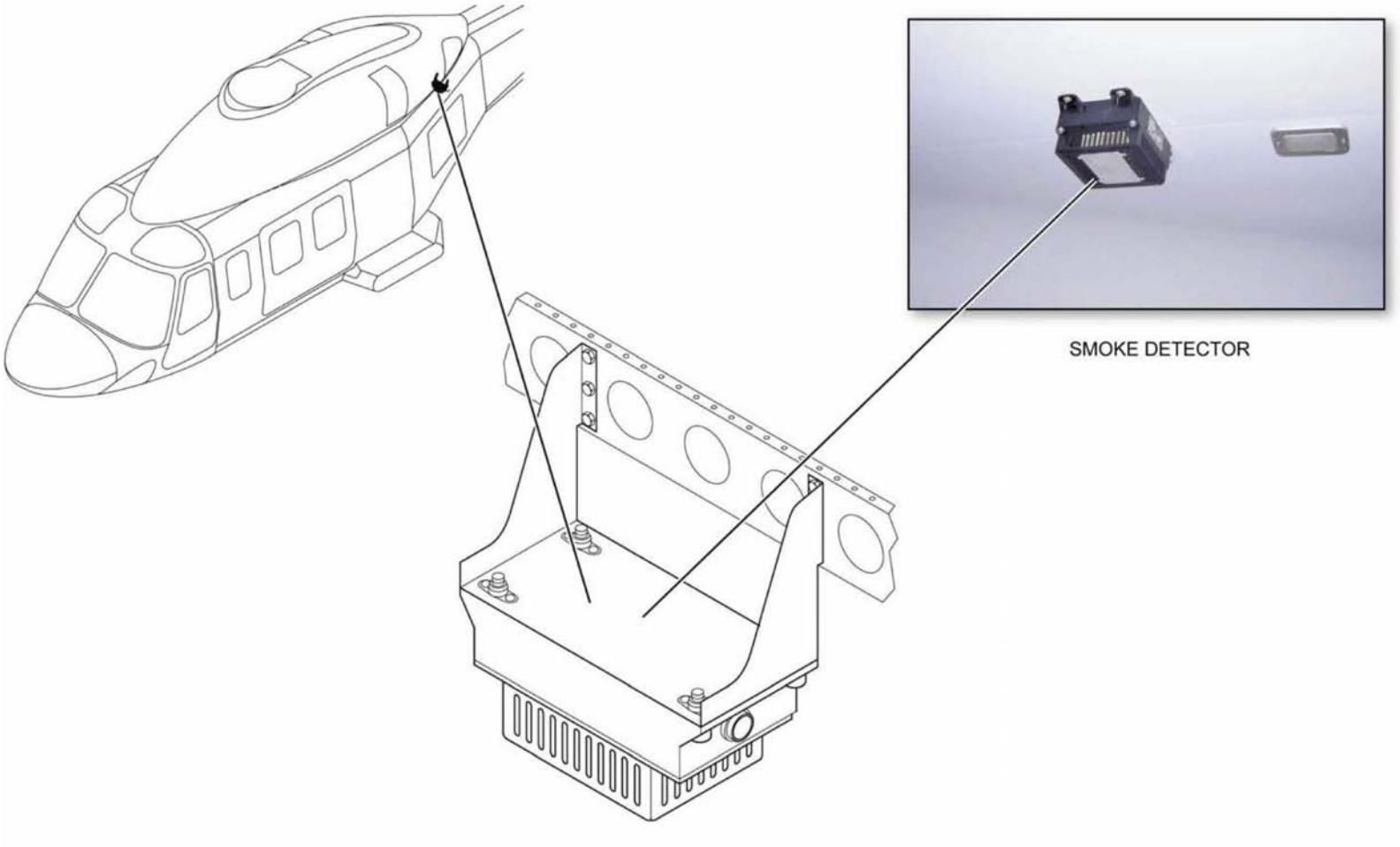
BAGGAGE FIRE DETECTION SYSTEM – MAIN COMPONENTS

SMOKE DETECTOR

The smoke sensor is made by a photoelectric device that operates on the light-scattering principle.
The detector alarms when the smoke concentration level exceeds a predetermined level.
The device employs two light sensing and amplifying channels installed in the baggage bay.

SMOKE DETECTOR – OPERATION

The reference channel senses the amount of light emitted by an internal LED (Light Emitting Diode) source.
The smoke channel senses the amount of emitted light scattered by smoke particles in the baggage.
The smoke detector alarms when the output from the smoke channel exceeds a predetermined smoke concentration level.



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FIRE DETECTION SYSTEM – BAGGAGE COMPARTMENT

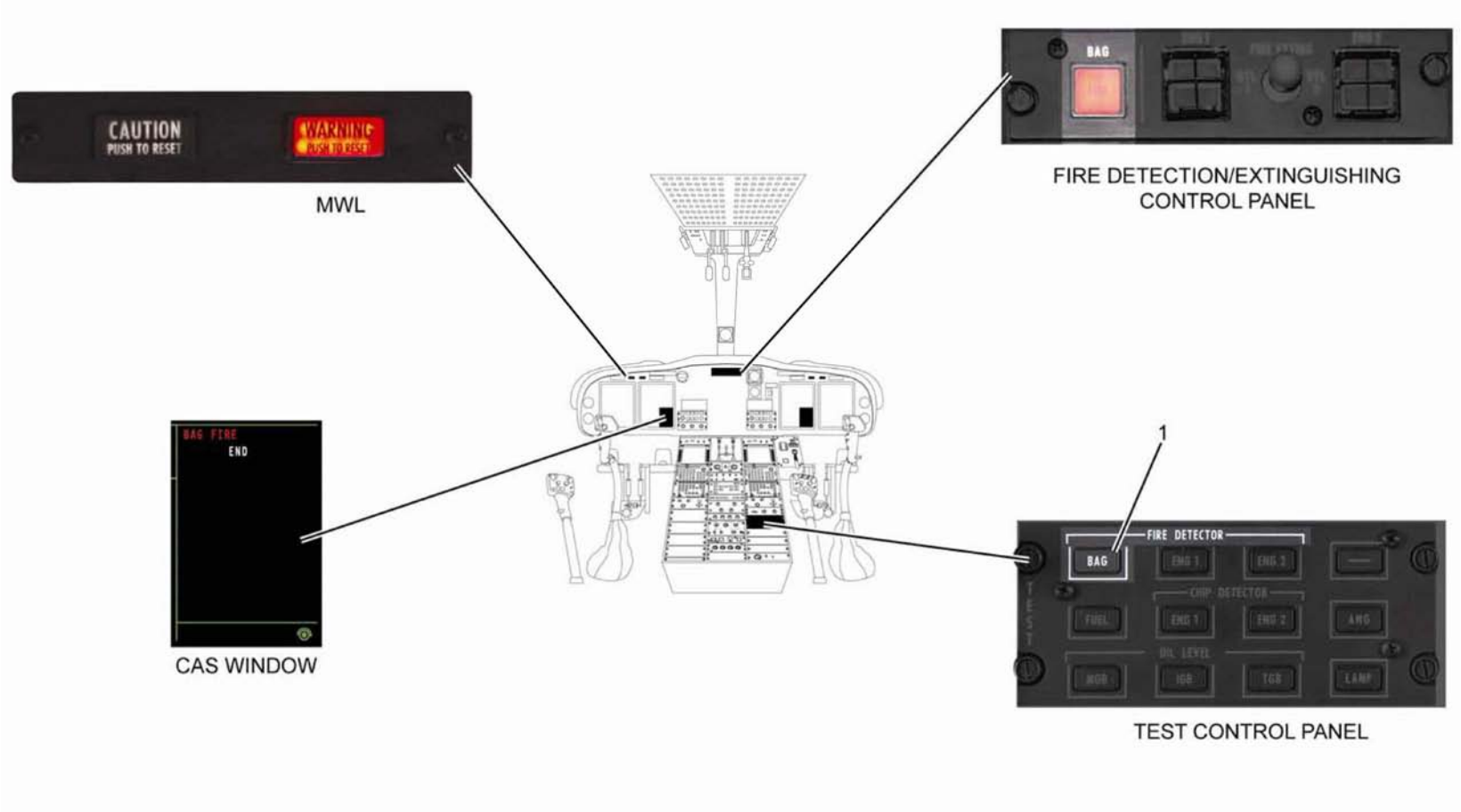
BAGGAGE FIRE DETECTION – TEST

1 FIRE DETECTOR BAG push-button

- pressed .....
  - MWL illuminate
  - the message BAG FIRE is displayed in the CAS window
  - the BAG indicator illuminates red on the FIRE EXTING control panel

NOTE C

When the push-button is released all the above lights and messages go OFF. See AW139-RFM-4D Section 2 Normal Procedures COCKPIT/ENGINE PRE-START CHECKS.

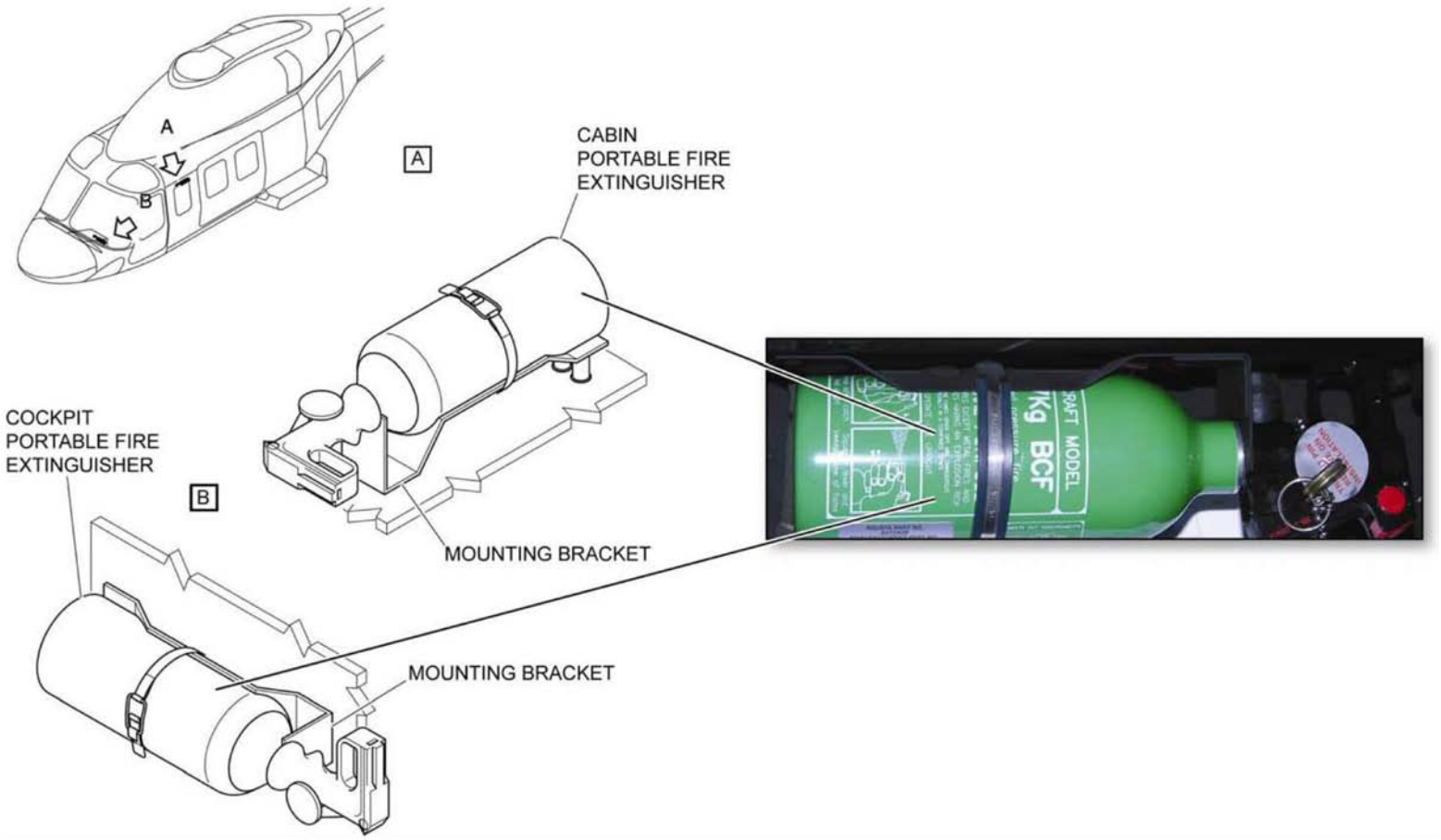


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BAGGAGE FIRE DETECTION – TEST

PORTABLE FIRE EXTINGUISHER – GENERAL

One portable fire extinguisher is installed in the cockpit between the seat of the pilot and the co-pilot.
A quick release mounting bracket allows for rapid removal of the fire extinguisher in case of fire. The extinguishing agent is HALON 1211 and the portable fire extinguisher can be used against small carbonaceous fires, flammable liquid fires and electrical fires.
A similar portable fire extinguisher is located into the cabin.



ICN-39-A-260000-G-A0126-01771-A-01-1

PORTABLE FIRE EXTINGUISHER INSTALLATION

FIRE DETECTION – CAS WARNING MESSAGES	
CAS CAPTION	**MESSAGE DESCRIPTION**
1(2) ENGINE FIRE	Fire is detected in the ENG 1 (2) bay or during the test When this warning is detected the AWG provides two pairs of tones + ENGINE 1 (2) FIRE - ENGINE 1 (2) FIRE aural message This sequence is continuously repeated until the failure is corrected or the reset input activated **CAUTION** IN CASE OF A SUBSEQUENT FIRE IN THE OTHER ENGINE BAY THE INITIAL ARM 1(2) PUSHBUTTON MUST BE DESELECTED TO ALLOW OPERATION OF THE ARM 2(1) PUSHBUTTON
BAG FIRE	Smoke is detected into baggage compartment or during the test When this warning is detected the AWG provides no tone + WARNING - WARNING aural message

FIRE DETECTION – CAS CAUTION MESSAGES	
CAS CAPTION	**FAILURE DESCRIPTION**
1(2) FIRE DET	Associated engine fire detect system not operational

FIRE EXTINGUISHING SYSTEM INDICATING

The maintenance provided on the MFD is as follows:

1	1(2) FIRE EXTING EMPTY …	when the extinguisher bottle N.1(2) is empty or its internal pressure is dropped below a predetermined value

NOTE D

This message stored in the fault log is announced by the MAINTENANCE statud message displayed on the MFD.

LEADING PARTICULARS

- Extinguisher bottle

Charge pressure	600 +25/-0 psing at 21°C (70°F)
Primary safety relief pressure	2311 to 2511 psig at 110°C (230°F)
Scendary safety relief pressure	3134 to 3434 psig at 110°C (230°F)
Agent	bromotrifluoromethane (CB_rF_3)
Pressurizing media	nitrogen
Internal volume	72 cubic inches
Ambient temperature range	-62 to +110°C (-82 to +230°F)
Charged bottle weight	5.90 lbs max
Power supply for bottle N.1	28VDC (ESS BUS NO.1)
Power supply for bottle N.2	28VDC (ESS BUS NO.2)

CHAPTER

28

FUEL

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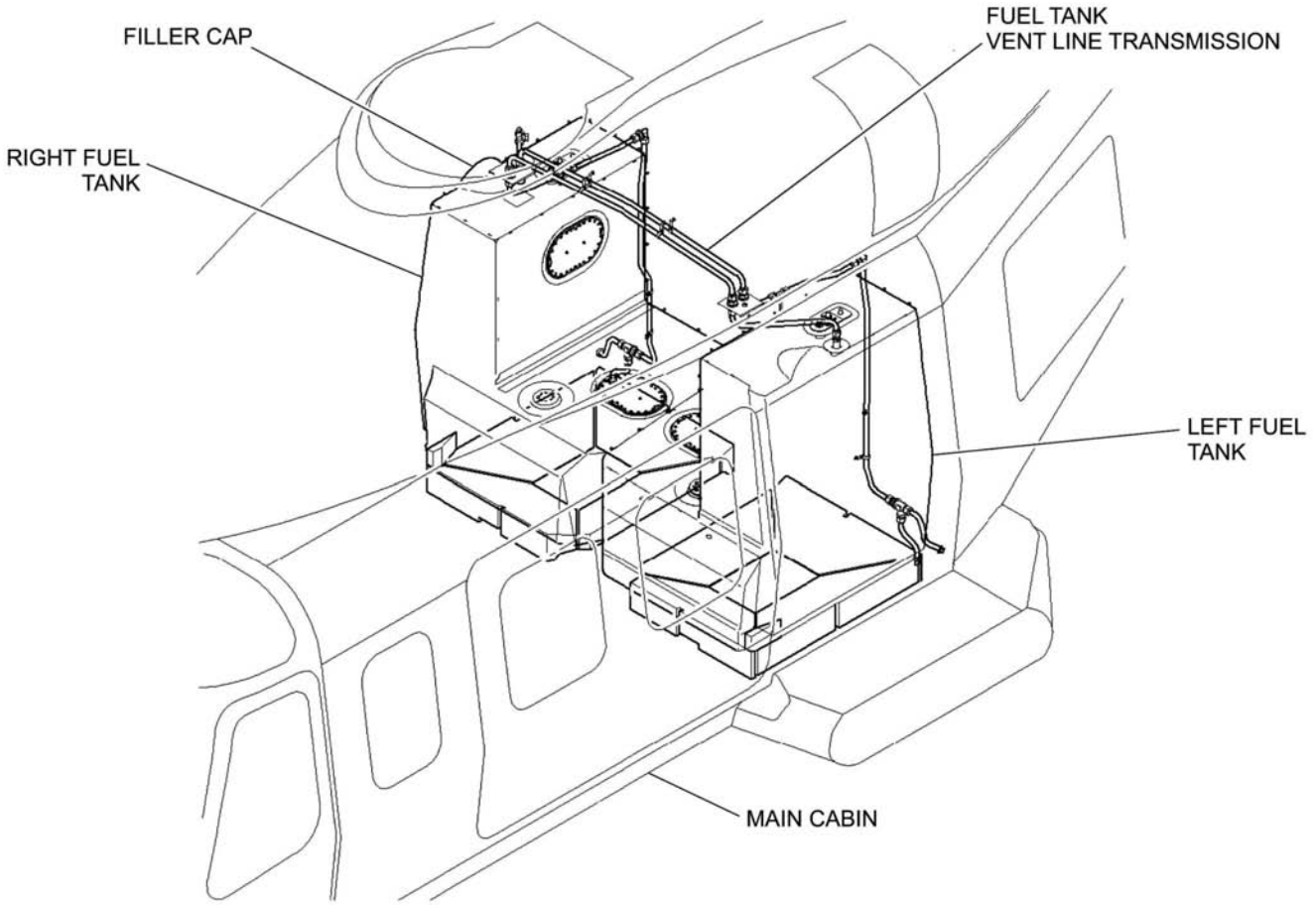
SECTION 10 - STORAGE

DESCRIPTION

The storage system is composed by two main left and right tanks located on the rear side of the cabin between the engine compartment floor and the aircraft underside.
Fuel tanks are bladder type made in rubber impregnated fabric and qualified to meet the crash resistant requirements for flexible tanks.
The two tanks are connected together through a flange located on the lower side of the tank at a specific height thus to allow a quantity of fuel (about 228 Kg) usable from only one tank.
In correspondence of the bottom sump plate the tank is installed on the structure by using four clips which in case of crash landing break away from their attachments allowing the tank, together with the sump, to collapse internally in the bay avoiding any damage which can cause fuel spillage.
Both tanks are gravity refilled through a filler cap located on top of the right tank.
Main tanks are provided with caps for connection with the auxiliary tank.
At the bottom of each tank is located a sump plate where are installed a fuel booster pump, a secondary quantity probe with low level sensor, a fuel draining valve and a water drain valve.
The electrical component cables are routed from the sump plate.
The water drain valve can be opened electrically with a switch located in the main landing sponson; a plunger on the valve allows also the manual operation.
To ensure an equalisation of pressure between the tanks and existing ambient pressure each tank is provided at the top with a venting pipeline routed to the opposite side to prevent post-crash spilling after roll-over.
The height and routing of the pipeline prevents excessive spilling or fuel collection during aircraft manoeuvres.
The venting pipeline ends at the bottom of the helicopter with two outlets.
One vent goes externally, the other goes internally into a fairing at the rear of the main landing gear bay to provide protection in the event that the external vent becomes blocked with ice or other contamination.
A flame arrestor in each venting pipeline prevents a flashback to the tanks in the event of fire hazards, such as lighting strikes.
At the top of each main tank is located a plate where are installed a tank supply hose, a pressure switch, a fuel return to tank from engine and an electrical connector.
The storage system is designed to provide the required safety conditions, including a crash situation, by:

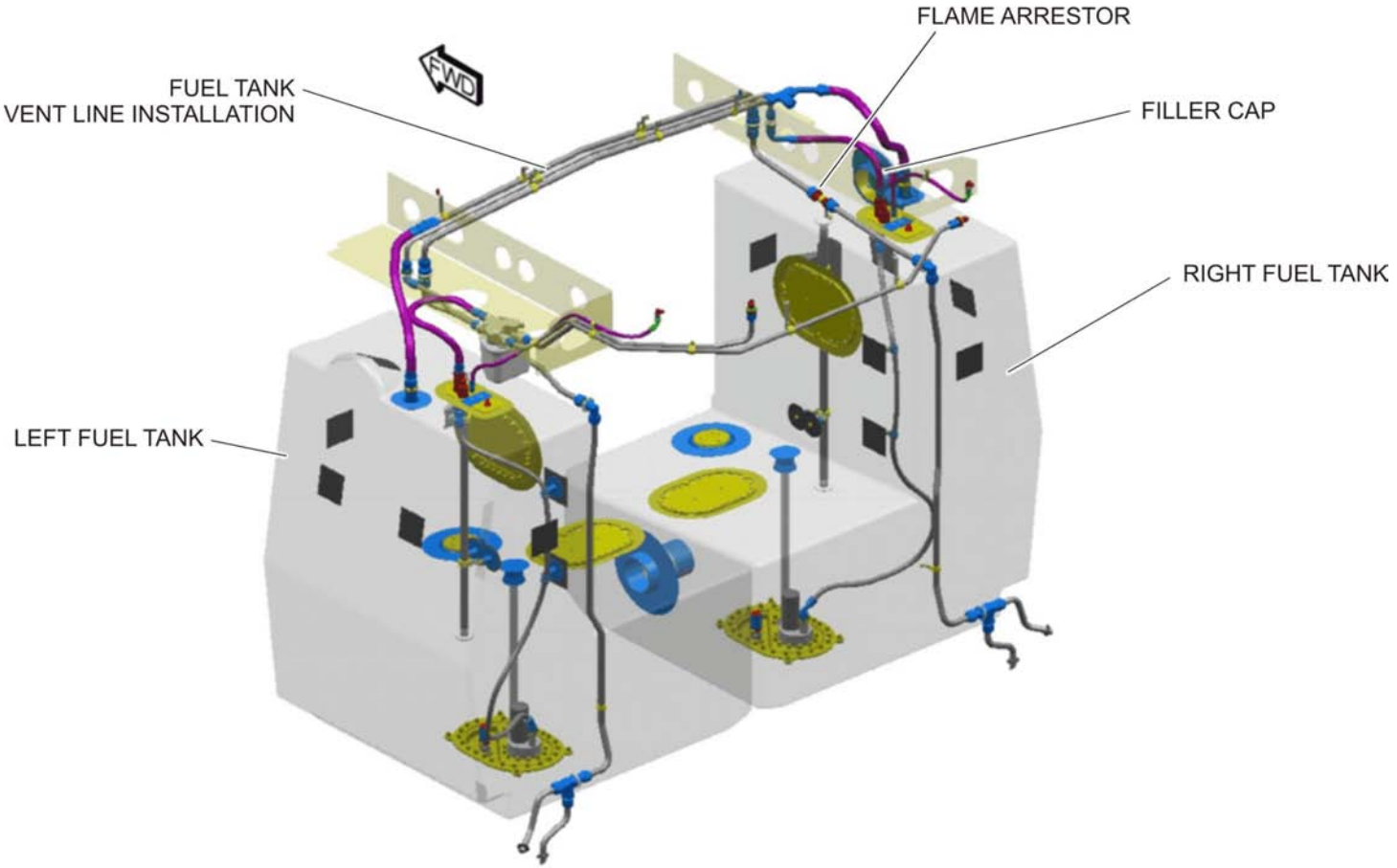
- Tank deformation and elongation to take up the structure shape and capability to breakaway its attachments to the airframe without loosing its integrity.
- Crossover vent system to prevent fuel spillage through the vent pipes in the event of the aircraft rolling on its side.
- Part of the tank fuel content is available to be used from a single source.
- Flame arrestors in the venting lines prevent flashback of flame into a tank.
- Controls and linkage are located clear of fuel tanks.

Main landing gears would collapse upwards in case of crash condition away from fuel tanks.



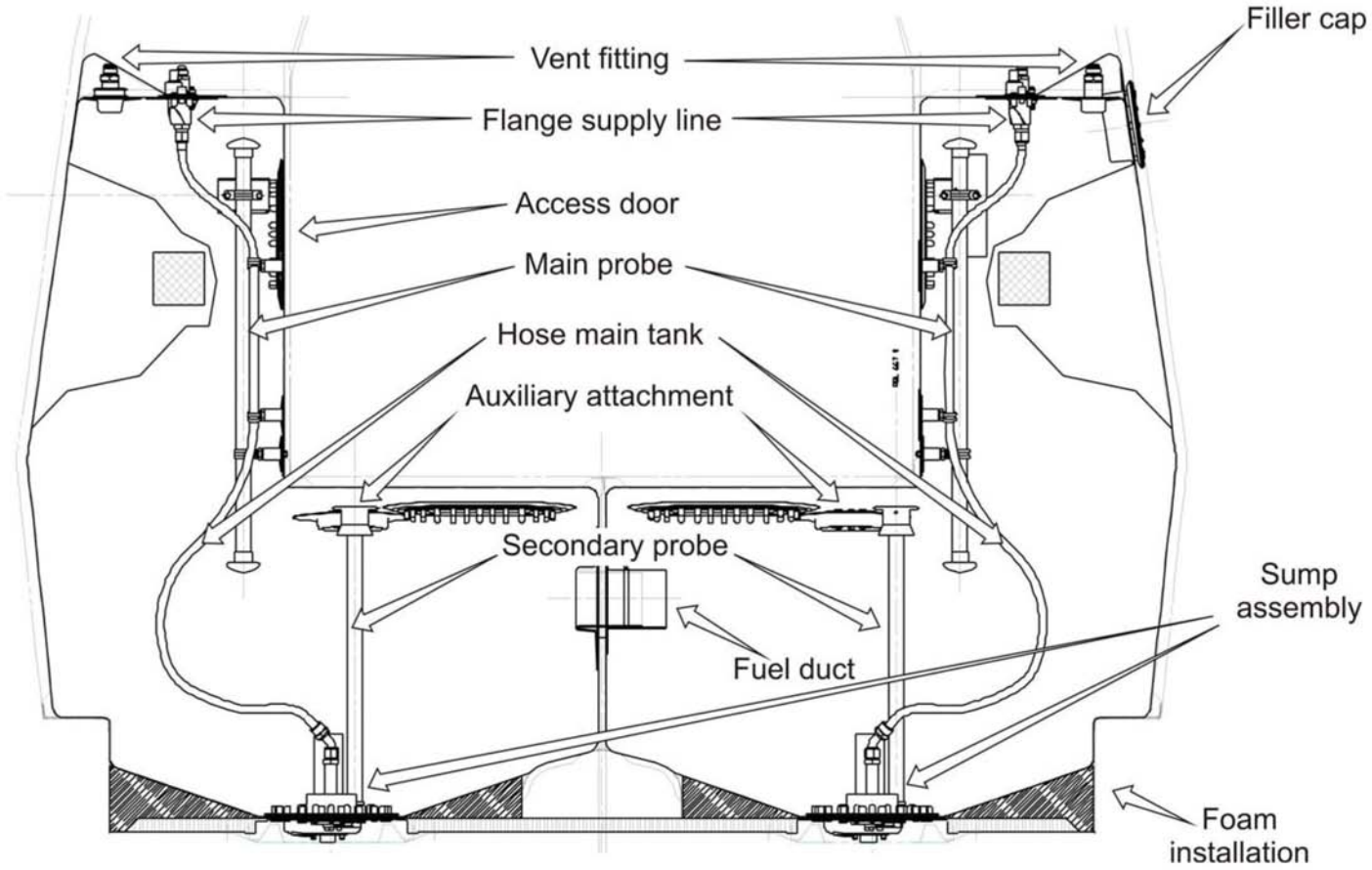
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FUEL STORAGE LAYOUT



ICN-39-A-281000-G-A0126-00424-A-01-1

FUEL STORAGE LAYOUT



ICN-39-A-281000-G-A0126-00425-A-01-1

FUEL TANKS SECTION

PRIMARY COMPONENTS

Filler Cap

The filler cap is installed on the external right side of the Number 2 tank. A plastic lanyard safeties the filler cap. It attaches the cap to the helicopter structure thus the cap cannot become loose.
The filler cap is crash-resistant and is made of stainless steel. It is red and has white labels with the open-close procedure operation.

Filler Neck

The filler neck is installed on the right side of the helicopter, at STA X6082.9, in the Number 2 tank.Twelve screw attach the filler ring of the filler neck to the helicopter structure. A packing is installed to seal the area.
Refuel of the Number 1 – Number 2 tanks is provide by the filler neck.
The filler cap closes the filler neck and seals the tank.

Water Drain Valve (Number 1 – Number 2)

The water drain valve is installed in the lowest point of the sump plate of the tank by two screws and two washers.
The valve drains the unwanted water. The valve can be operated both manually and electrically. A push button on the hexagonal head of the valve lets you operate the valve manually.

Drain Valve (Number 1 – Number 2)

The drain valve is installed on the sump plate of the tank, approximately at STA X6180. The drain valve has a threaded end fitting which engages in a housing in the sump plate. It also has a groove that contains a packing that seals the installation.
A wire safeties the Drain valve to the Water Drain Valve.
The drain valve is used to drain the bladder tank fully.
A safety system prevents the operation of the drain valve, thus the tank can be drained only through a controlled procedure.

Access Panel (Number 1 – Number 2)

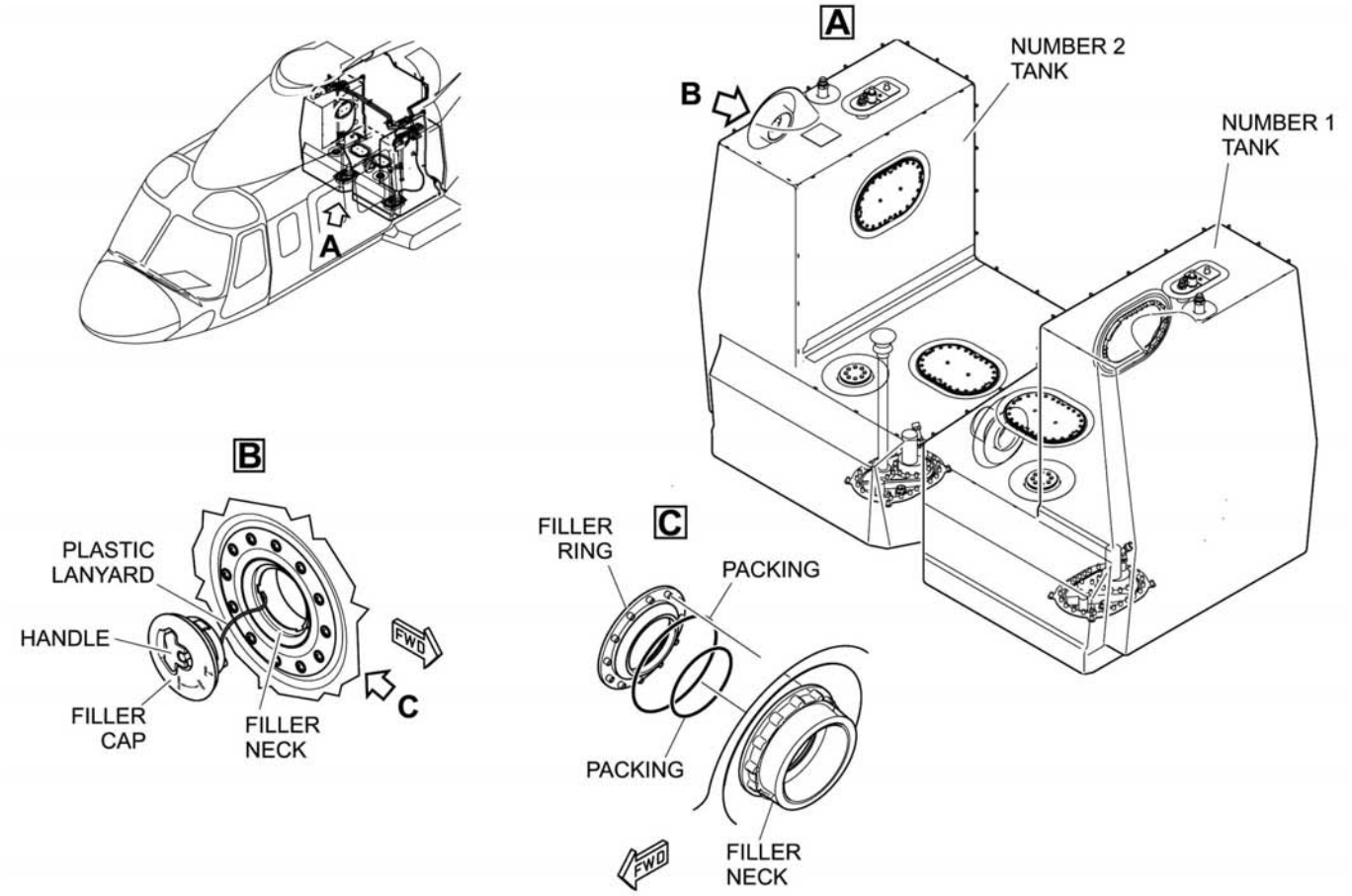
The side-access panel is installed on the vertical wall of the tank. It is attached to a special flange with 24 washers and 24 screws. The flange has a groove for the packing. The top access panel can be removed to do the inspection of the components installed in the tank.

Tank Sump (Number 1 – Number 2)

The sump is installed on the bottom of the tank. It is attached to its flange with 25 washers and 25 screws.
The capacity of the tank sump is approximately 1.1 litres of fuel.

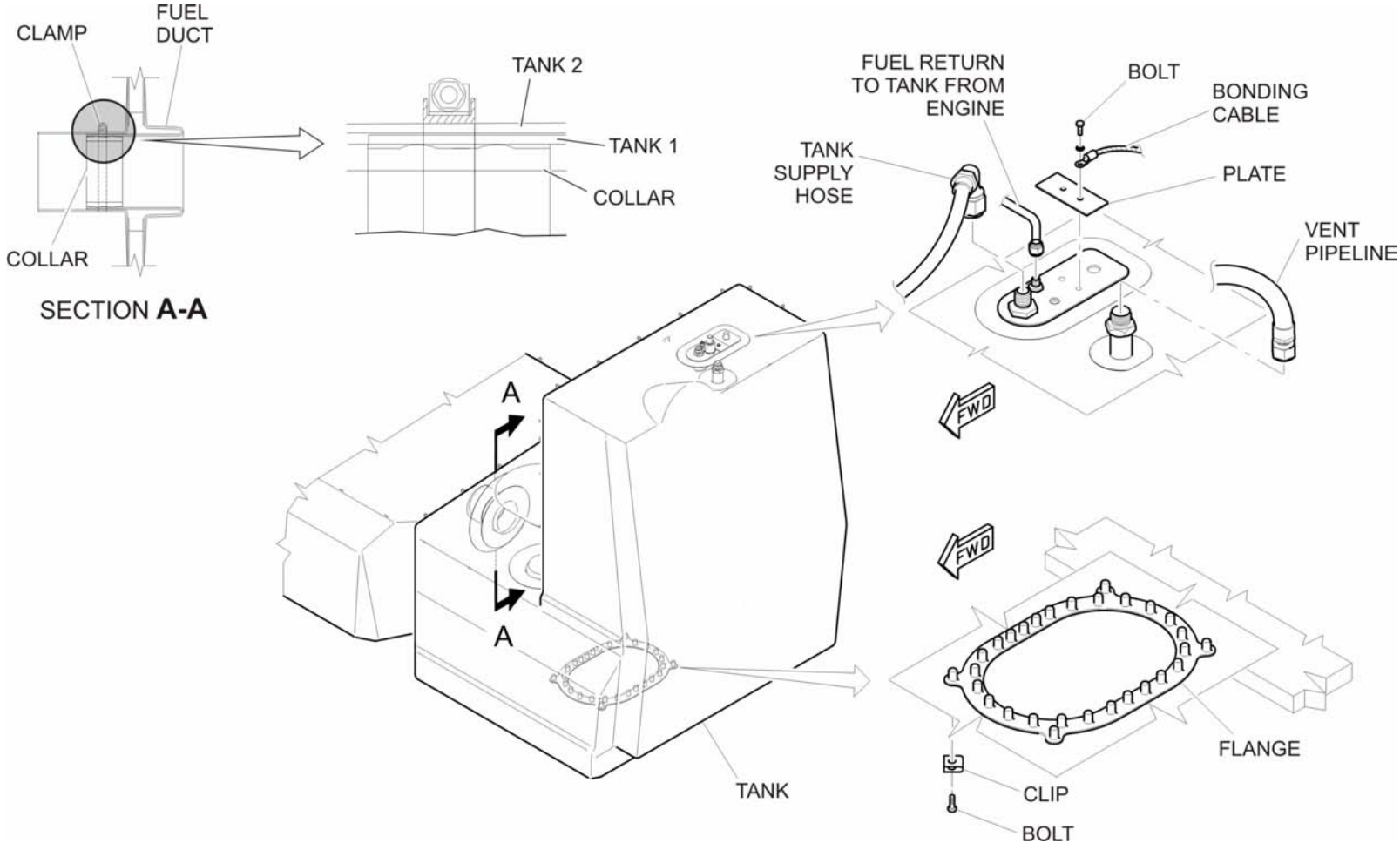
Foam Installation

The foam are installed on all sides of the main tanks in the rear cabin. They keep the tanks in the correct position, thus no tank loads are applied to surfaces that have no support below them.
The foam have openings which let you drain the tank bay. These openings also let a sufficient flow of air go through the tank bay.



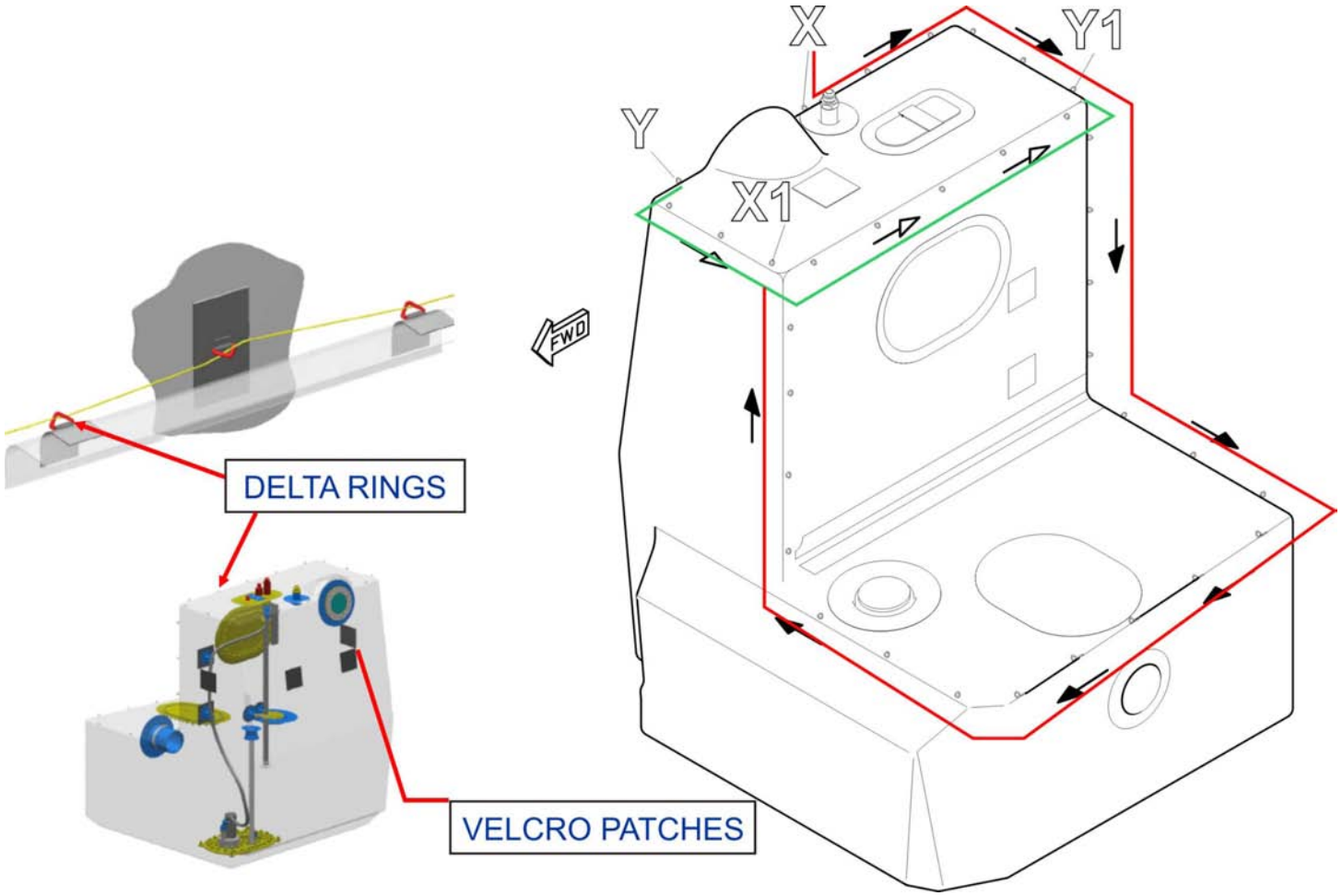
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FUEL TANK INSTALLATION (SHEET 1 OF 3)



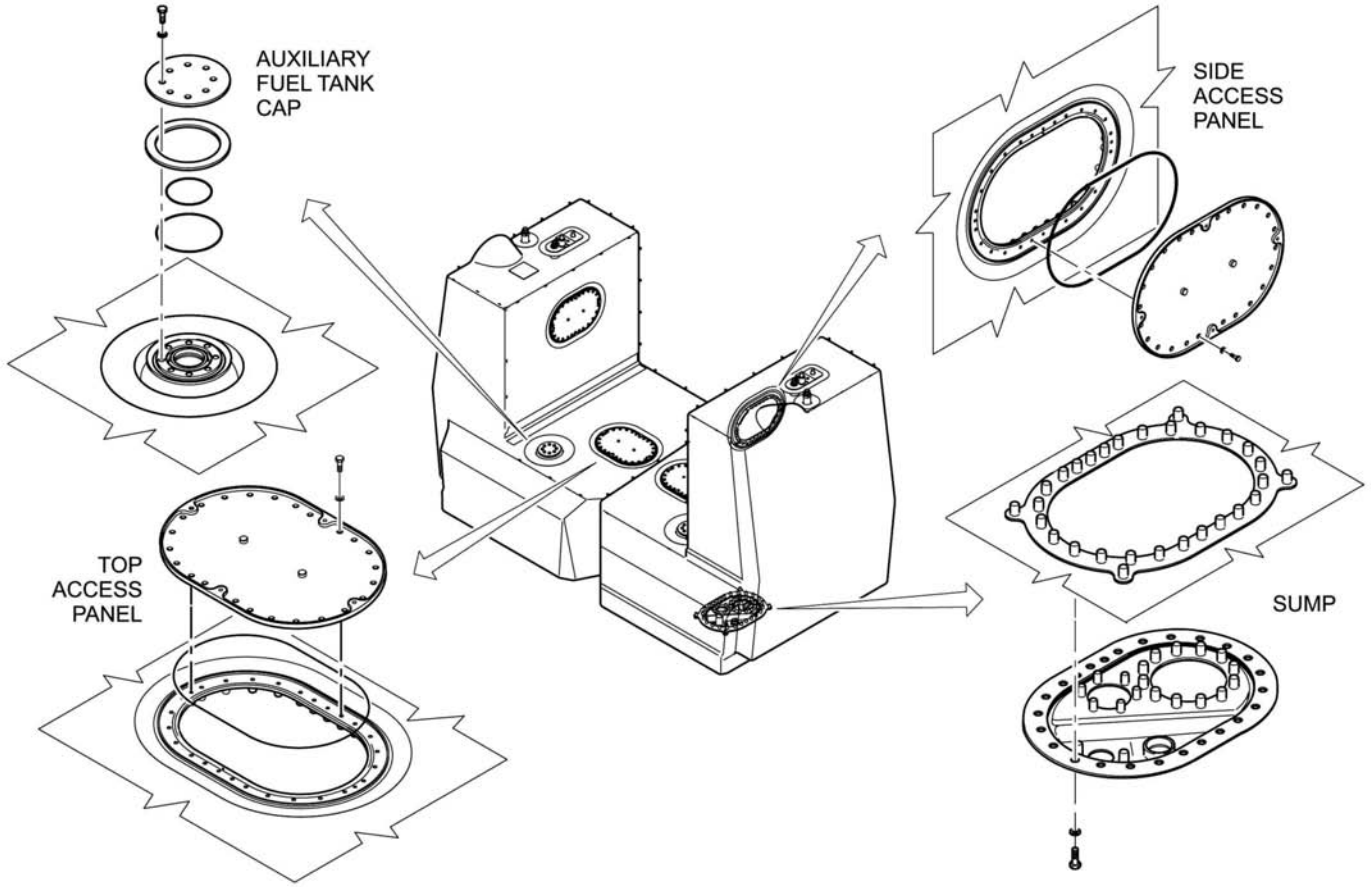
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FUEL TANK INSTALLATION (SHEET 2 OF 3)



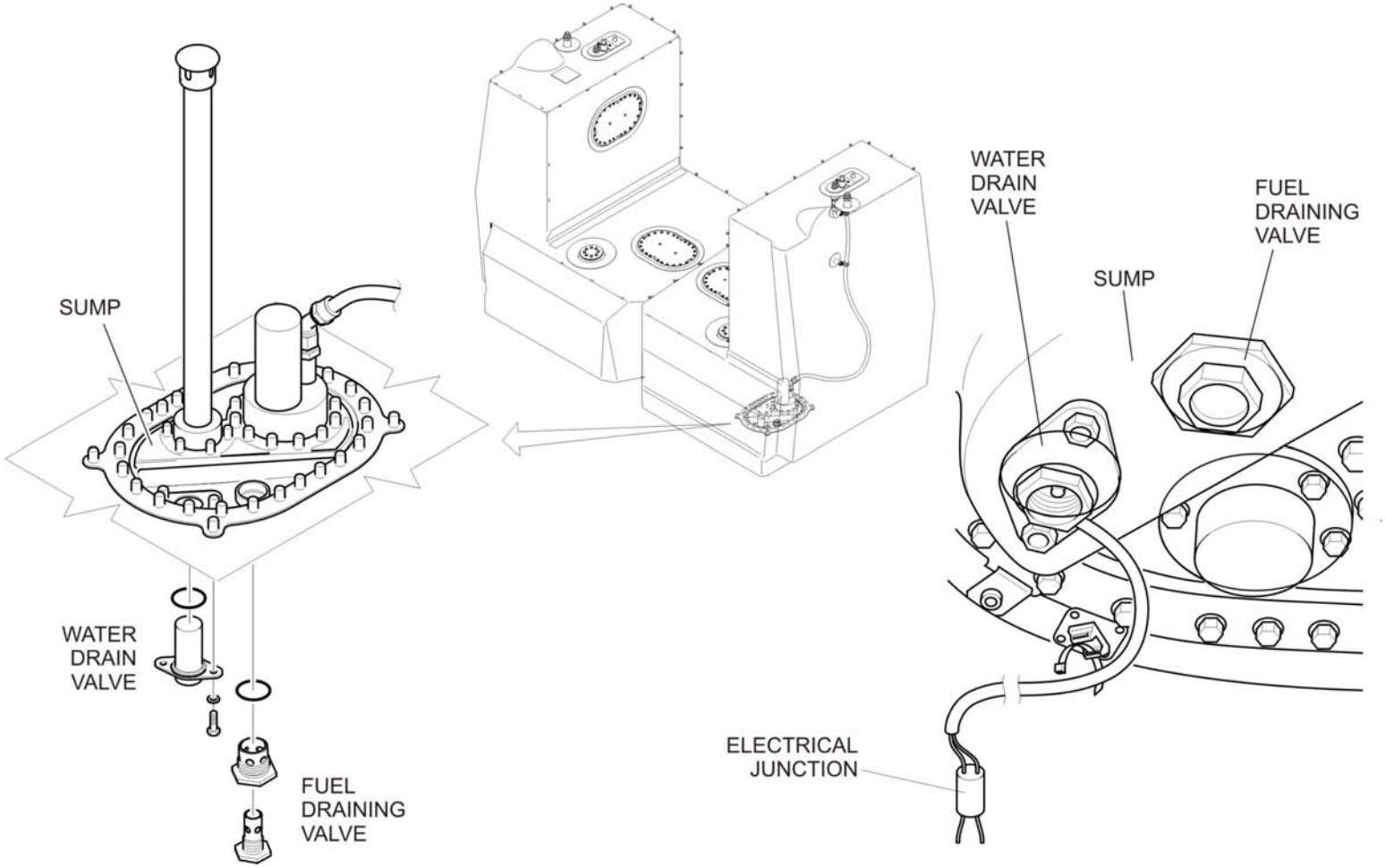
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FUEL TANK INSTALLATION (SHEET 3 OF 3)



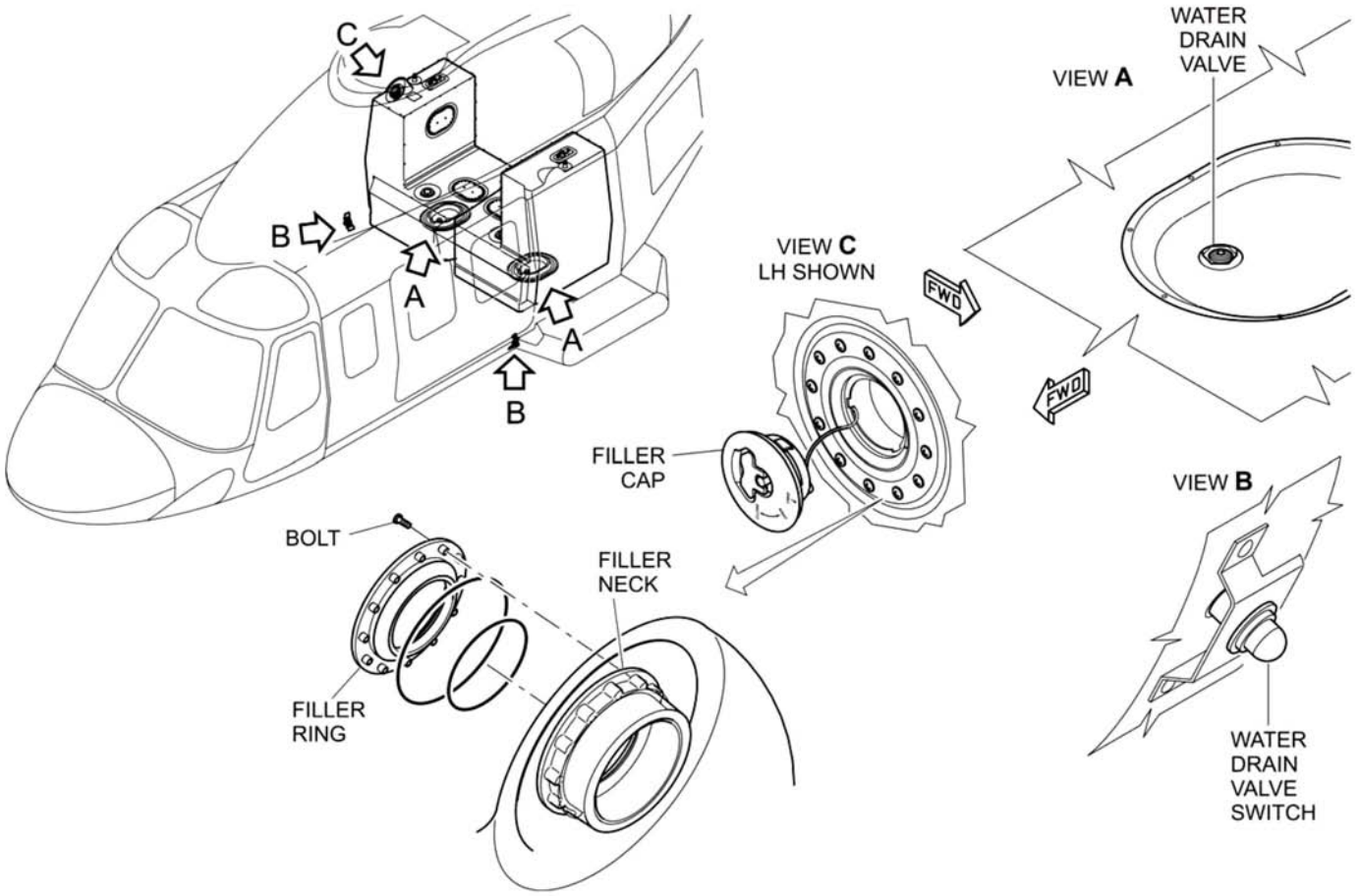
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FUEL TANK ACCESS POINTS



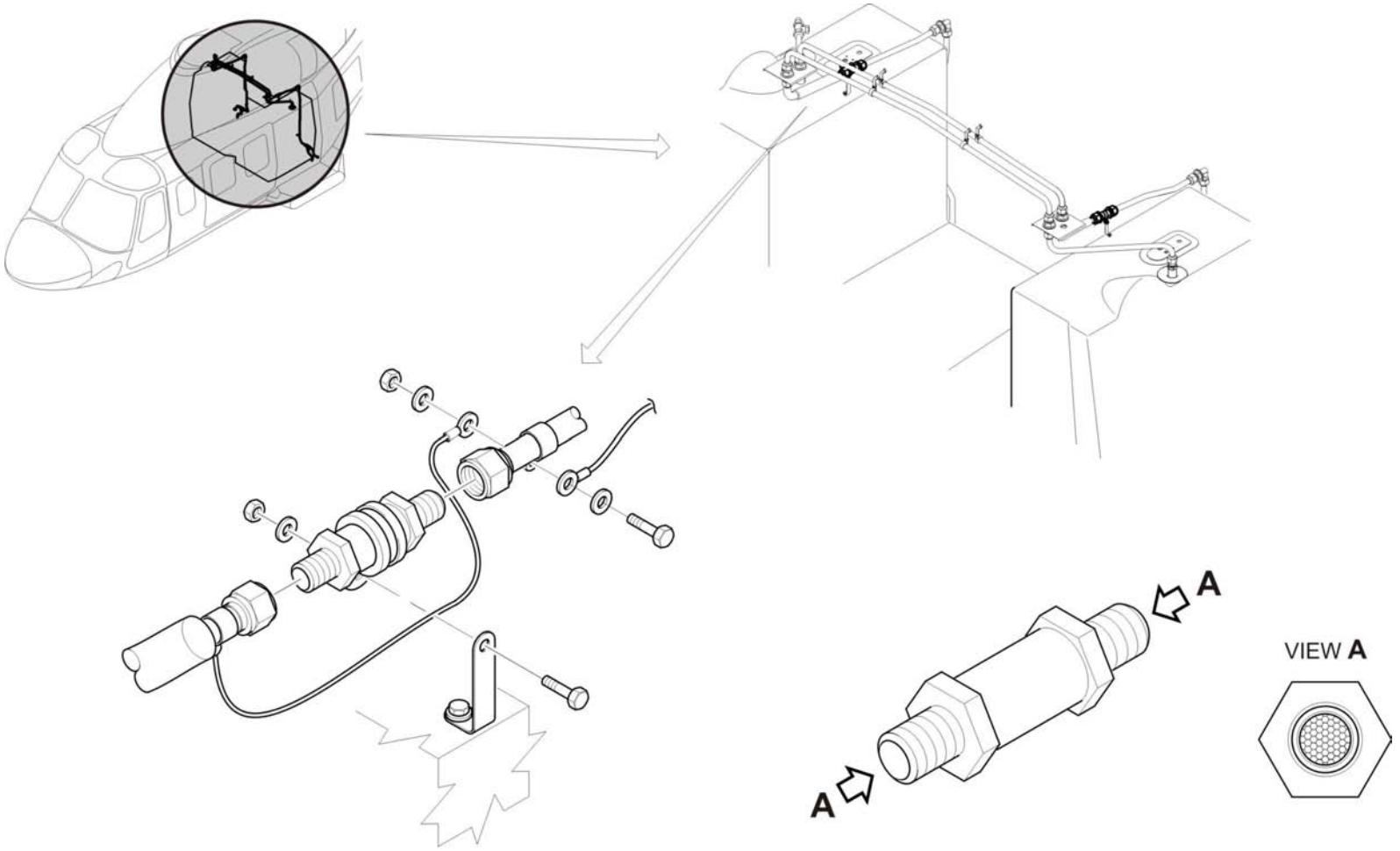
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DRIVE VALVES INSTALLATION



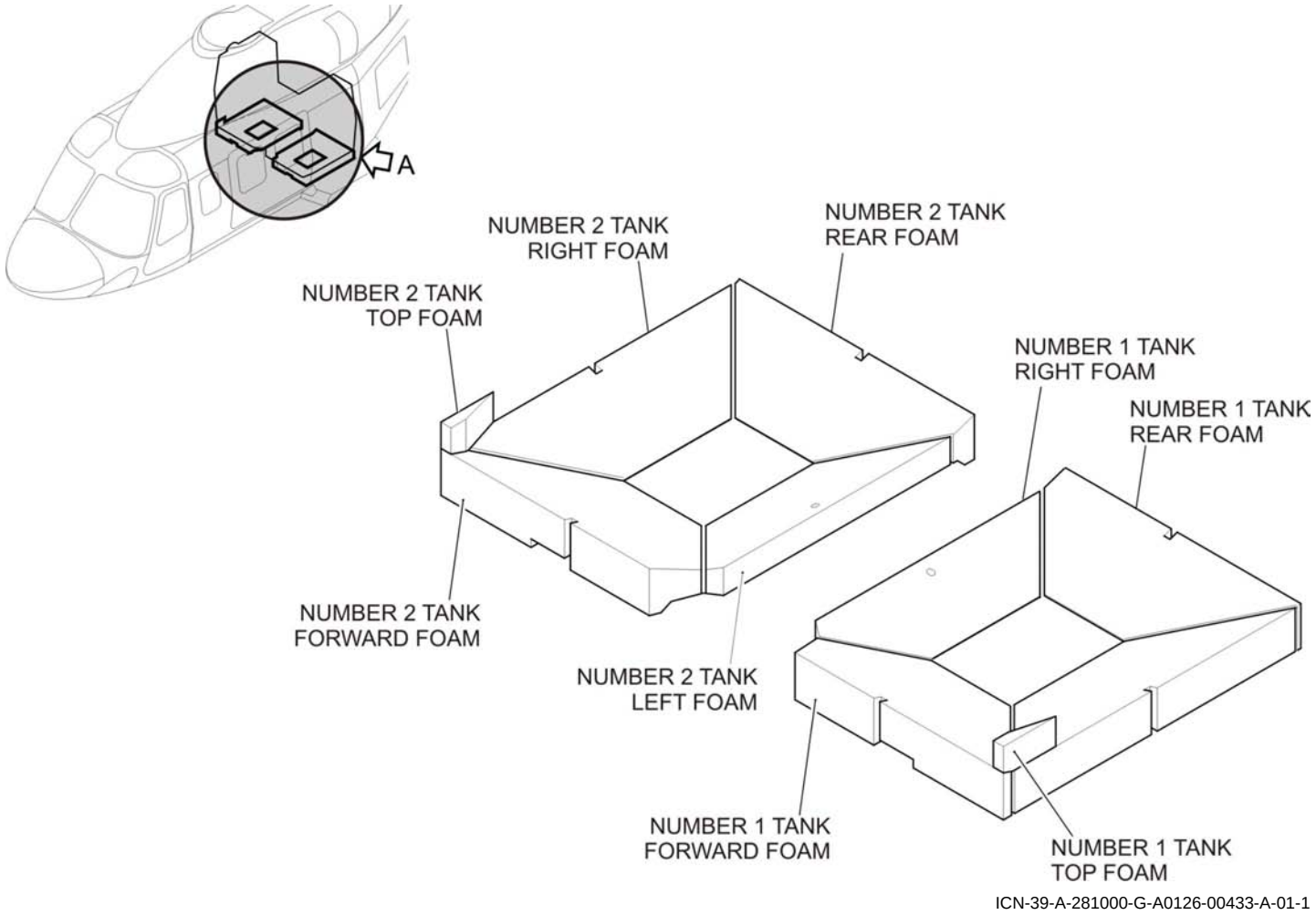
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WATER DRAIN VALVE OPERATION



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FLAME ARRESTOR



ICN-39-A-281000-G-A0126-00433-A-01-1

FOAM INSTALLATION

MAINTENANCE INFORMATION

- The fuel tank surface is easily damaged.
  Do not use pointed or sharp edges tools when you touch or move the fuel tank.
  Do not put the fuel tank on surfaces which have sharp edges.
- The fuel tank material is sensitive to light and low temperature conditions.
  Do not let the fuel tank stay in heavy light (such as sun rays) more than necessary.
  Do not touch the fuel tank when the environmental temperature is low.
- Before tank removal, it's required to remove the sump, the booster pump hose and the primary probe.
- Before water drain valve removal it's required to record the position of the electrical wires.
- After tank removal, it's required to replace the installation nylon cord.

LEADING PARTICULARS

- Total capacity..... 1270 Kg (1588 l)
- Unusable fuel max..... 8 Kg (10 l)
- Unusable fuel when cross feeding..... 228 Kg (285 l)
- Low fuel caution..... below 92 Kg (112 l)
- Authorized fuels..... see following table
- Water drain valve electrical supply..... 28 VDC NON ESS BUS1
- Tank weight..... 16.4 Kg (36.1 lb)

Fuel Type	Applicable Specification
JET A JET A-1	ASTM D1655 ASTM D1655
JP5	DEF STAN 91-86 AVCAT/FSII MIL-PRF-5624F NATO CODE F-44
JP8	DEF STAN 91-87-2002 AVTUR/FSII MIL-T-83138D NATO CODE F-34
JP8+100	Aeroshell Performance Additive 101

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SECTION 20 - DISTRIBUTION

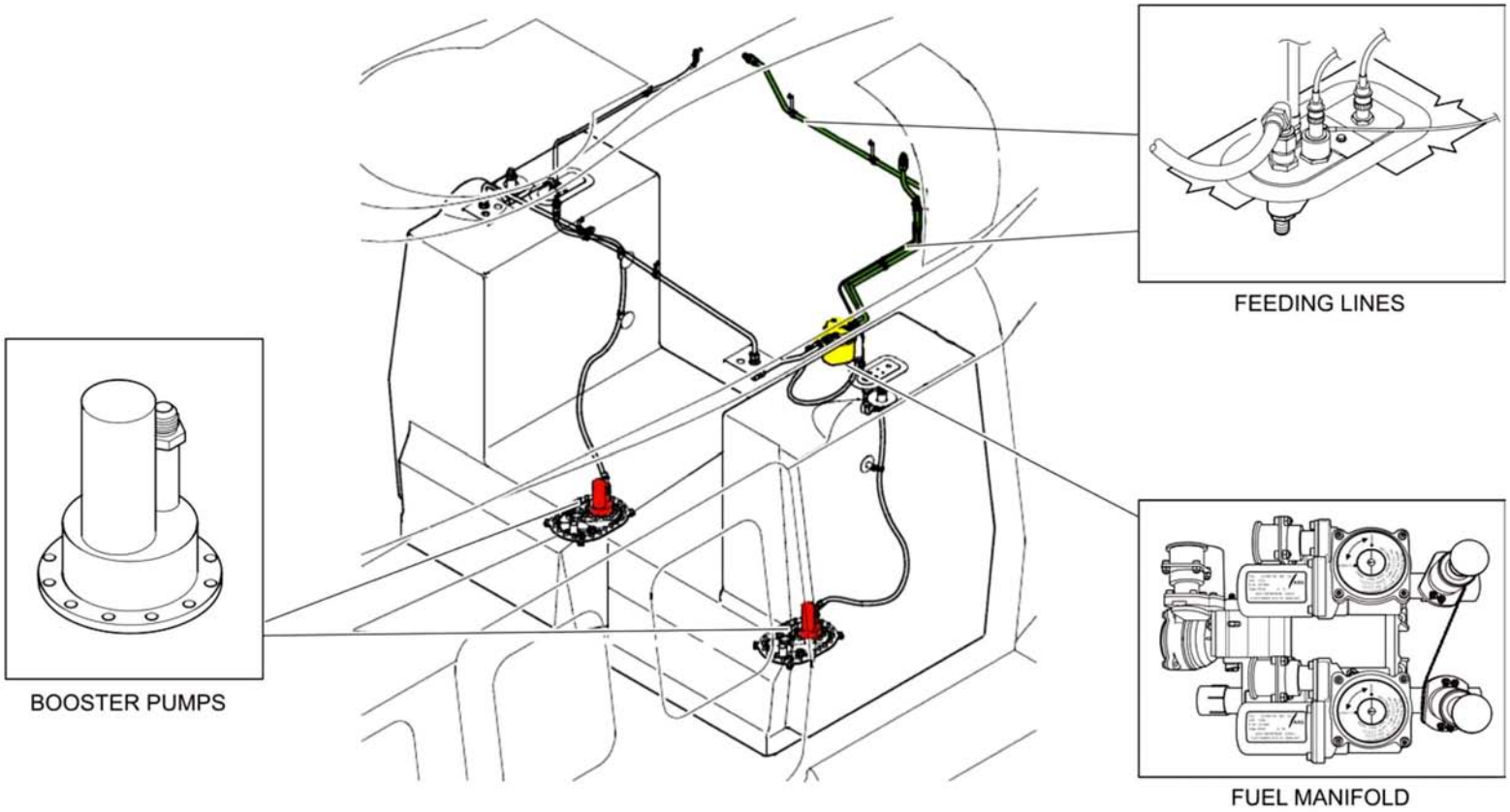
DESCRIPTION

The fuel distribution system is designed to provide the required safety conditions by:

- self sealing frangible couplings on the engine inlet pipelines at engine firewalls and on the hoses at the fuel tanks output.
- engine feed lines are double wall flexible hoses
- lines inside the tanks are flexible to allow elongation and avoid tank puncture
- fuel components and pipelines are routed away from the bottom aircraft around tank area.

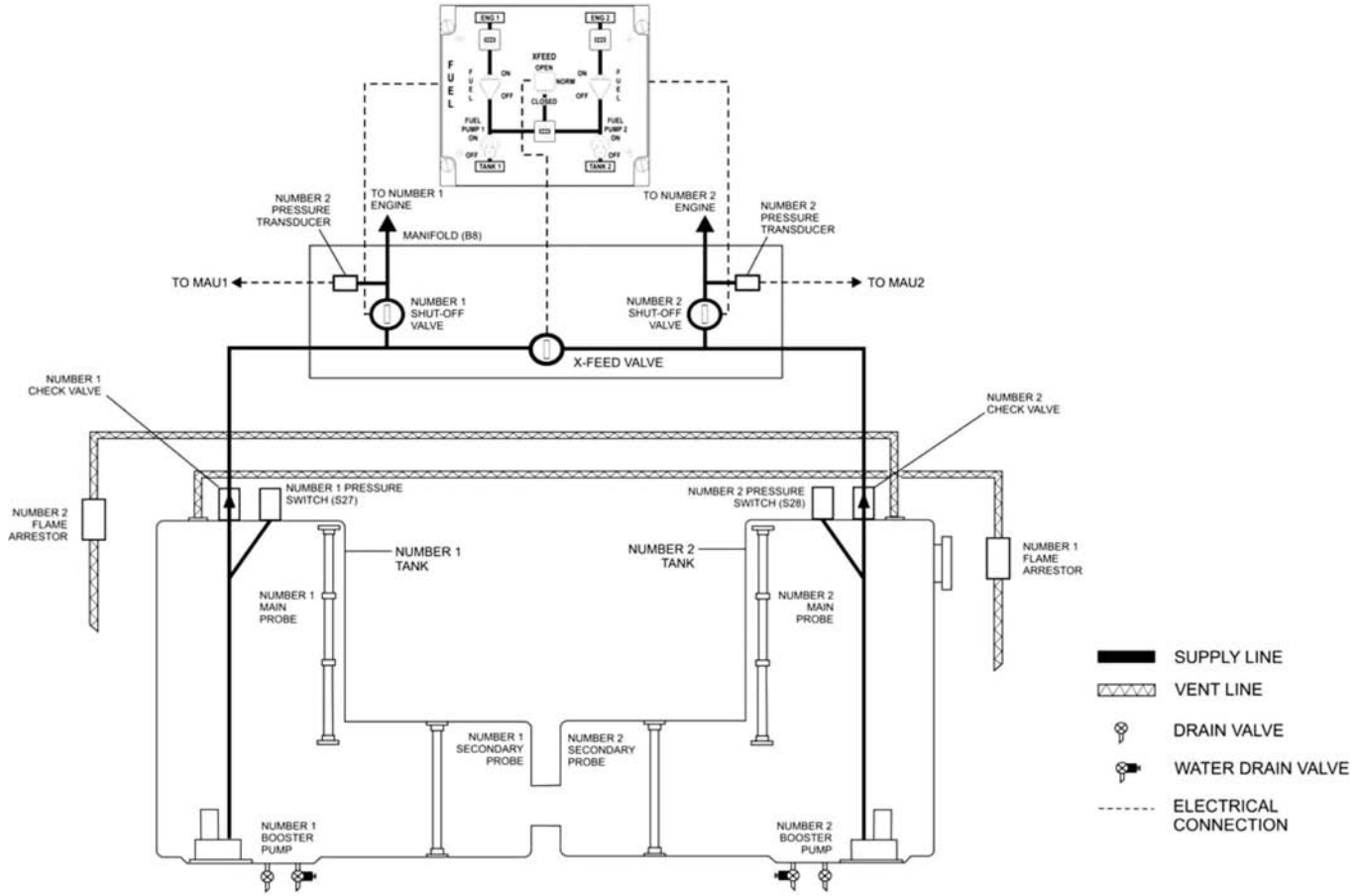
The distribution system is composed by:

- booster pump
- feeding line
- fuel selector manifold.



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FUEL DISTRIBUTION LAYOUT



ICN-39-A-282000-G-A0126-00644-A-01-1

FUEL DISTRIBUTION SYSTEM SCHEMATIC

CONTROLS AND INDICATORS – FUEL CONTROL PANEL

The fuel distribution controls are housed on the FUEL CONTROL PANEL and is installed on the console.
The operative functions and the malfunctions are displayed in the CAS message windows of the MFD.
The Fuel Control Panel provides the following functions:

1 Fuel pump 1 and 2 Switches:

- OFF/ON ..... allows the fuel booster pump 1 or 2 operational conditions.

2 Fuel Shut-Off Valve 1 and 2 switch:

- OFF/ON ..... allows the shut-off valve 1 or 2 operational conditions.

3 SHUT-OFF Valve indicators:

- Is black when the helicopter has no electrical power supply.
  The white flow line is vertical when the shut-off valve is open.
  The white flow line is horizontal when the shut-off valve is closed.

4 XFEED Switch:

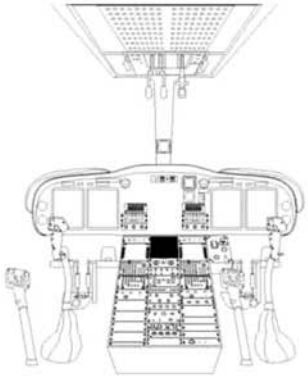
- CLOSED: ..... allows to close the crossfeed valve.
- NORM: ..... allows to open the crossfeed valve when the pressure switch in the feeding line detect a low pressure condition.
- OPEN: ..... allows to open the crossfeed valve.

NOTE A

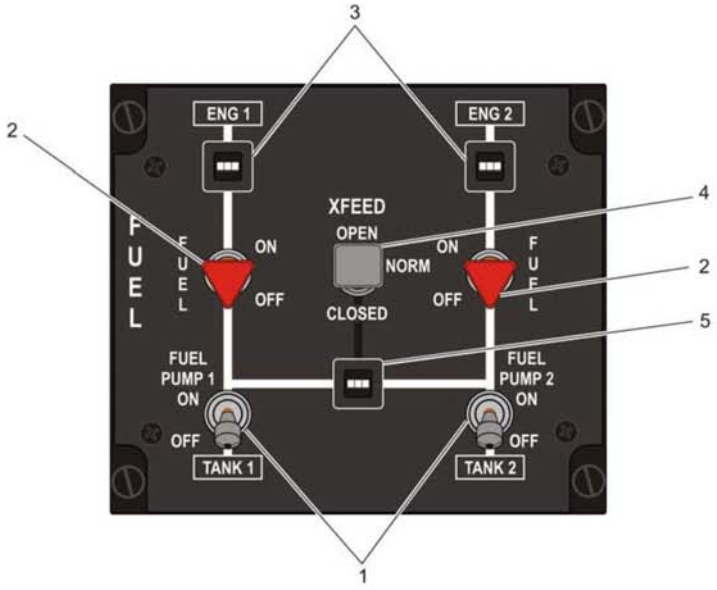
In case of fire, the shut-off valve 1 or 2 is automatically closed when the relative ENG 1 or ENG 2 push-button is pressed (to arm the EXTINGUISHER) on the FIRE DETECT/EXT CONTROL PANEL.

5 XFEED indicator:

- Is black when the helicopter has no electrical power supply
  The white flow line is vertical when the crossfeed valve is closed.
  The white flow is horizontal when the crossfeed valve is open.

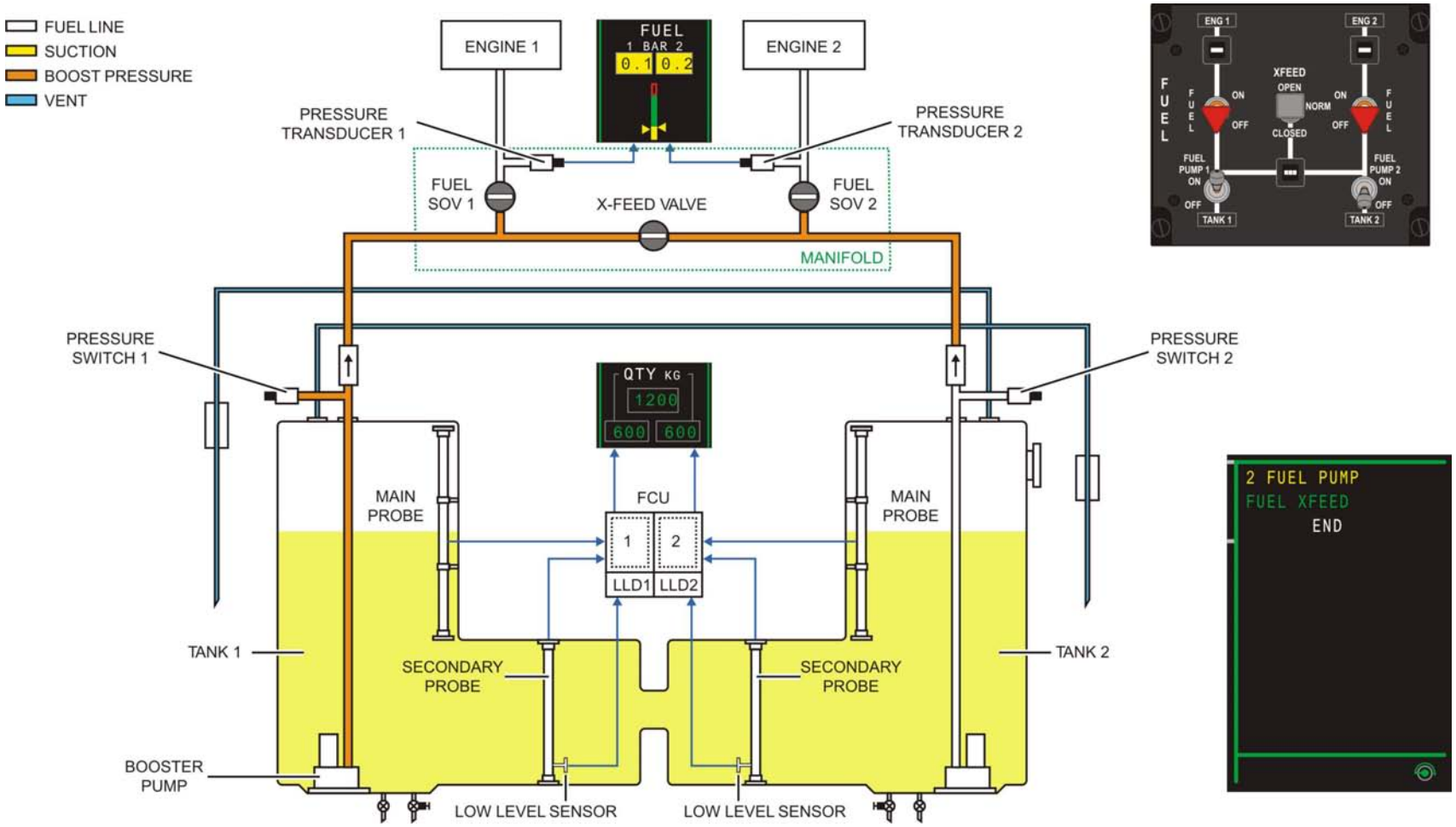


1. FUEL BOOSTER PUMP 1 and 2 switch
2. SHUT-OFF VALVE 1 and 2 switch
3. SHUT-OFF VALVE 1 and 2 indicator
4. CROSS FEED VALVE switch
5. CROSS FEED VALVE indicator



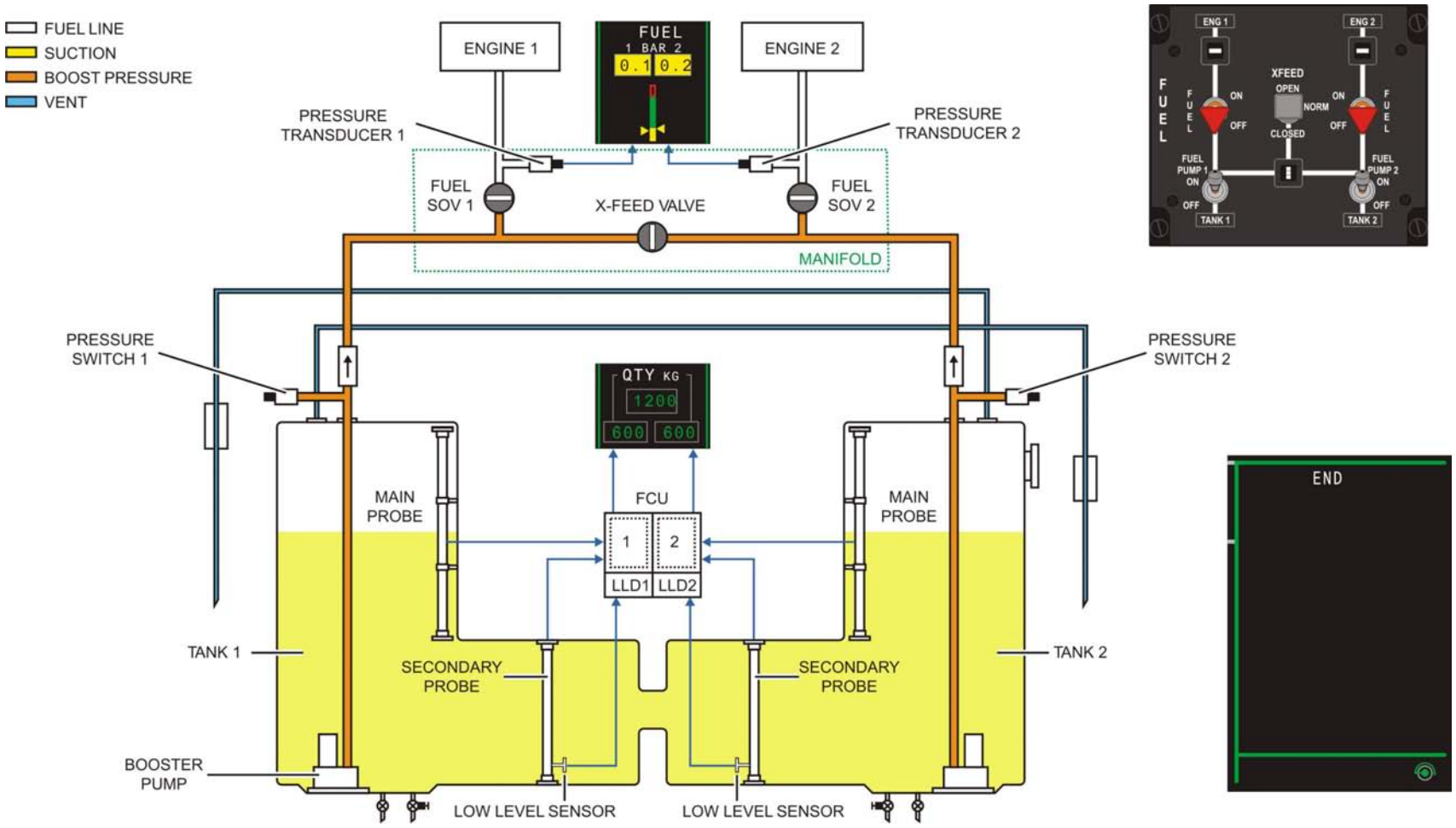
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FUEL CONTROL PANEL



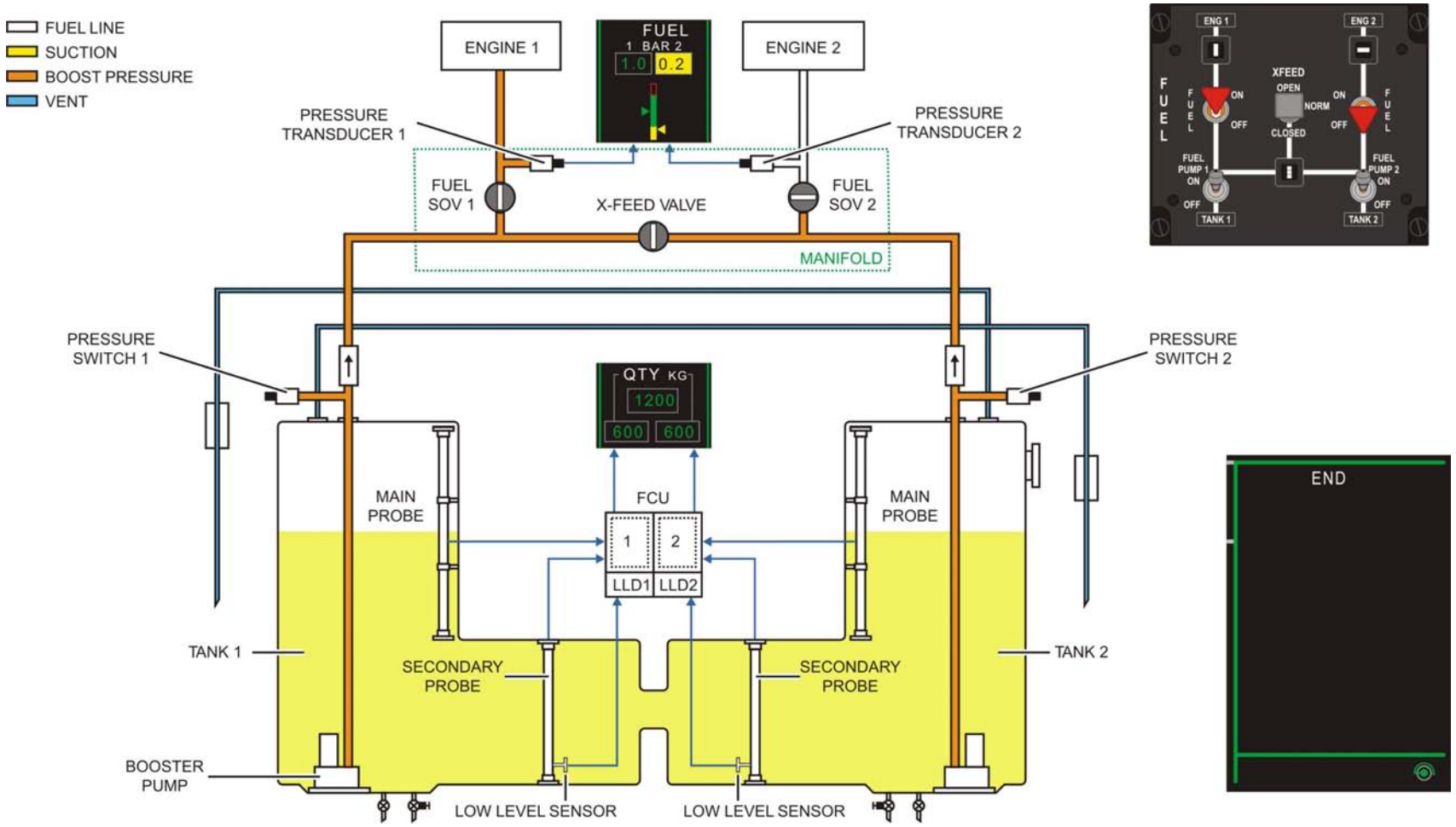
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BOOSTER PUMP NO.1 ON



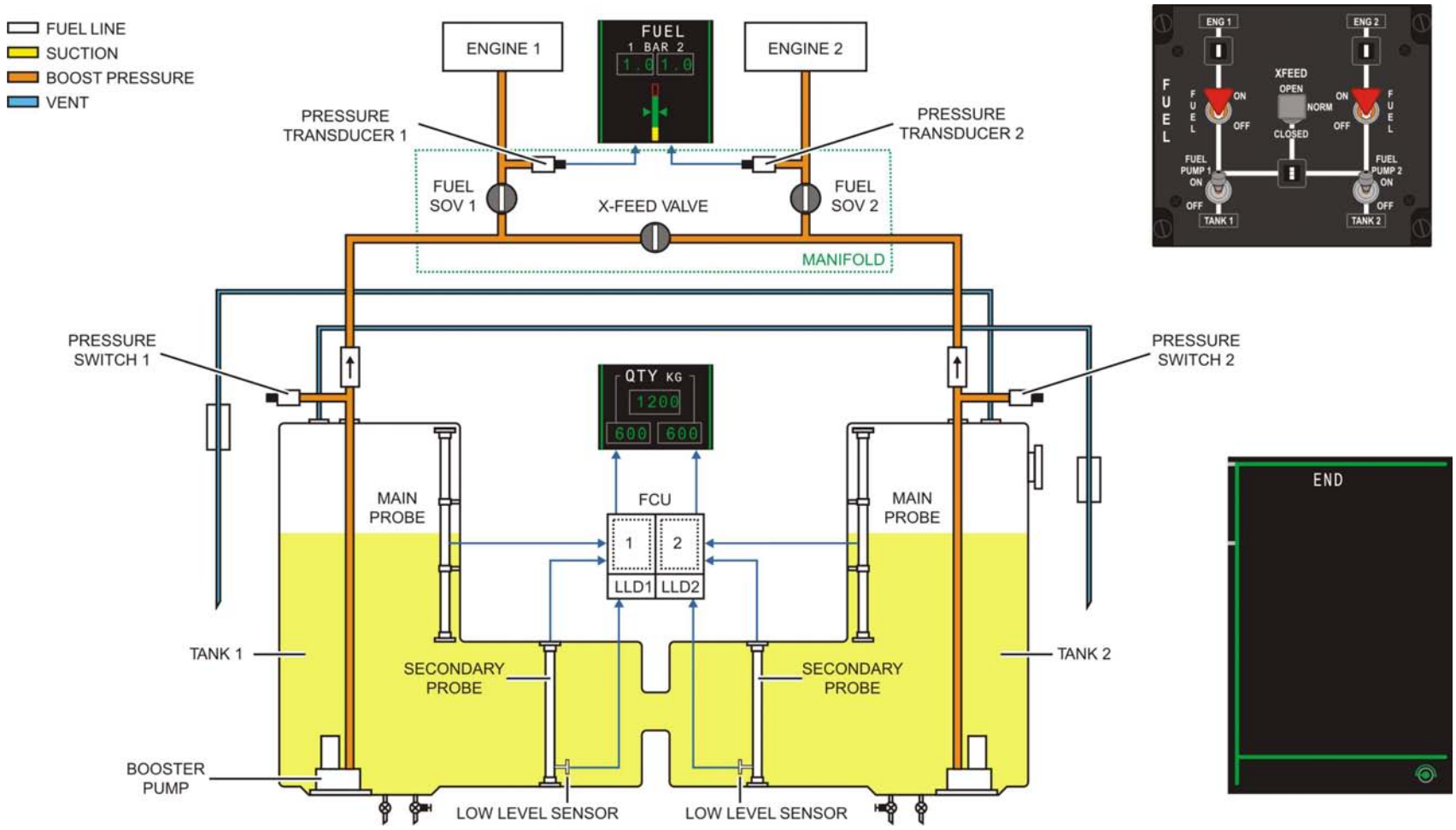
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BOOSTER PUMP NO.1 AND NO.2 ON



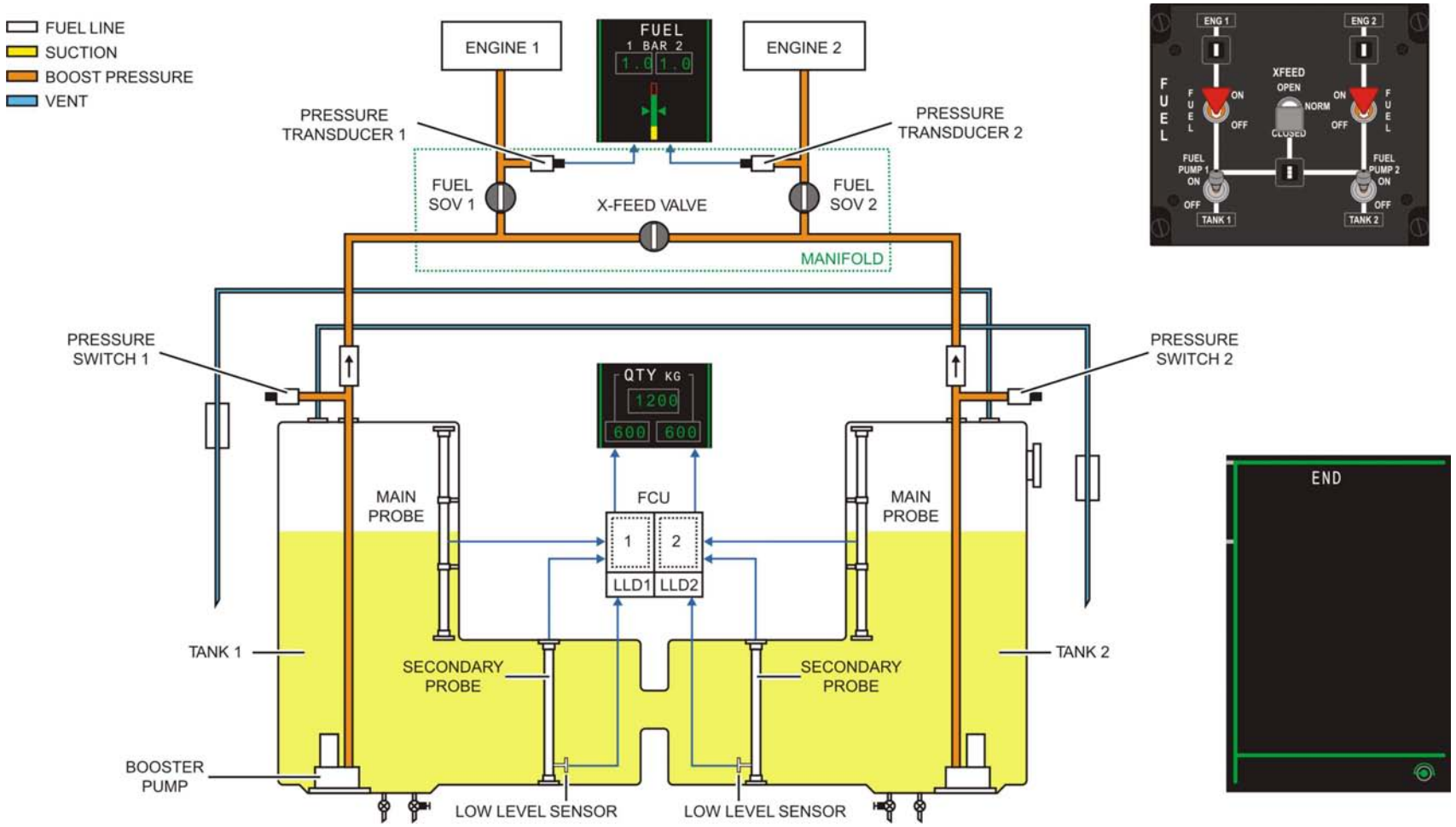
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BOOSTER PUMP NO.1 AND NO.2 ON AND FUEL SOV NO.1 ON



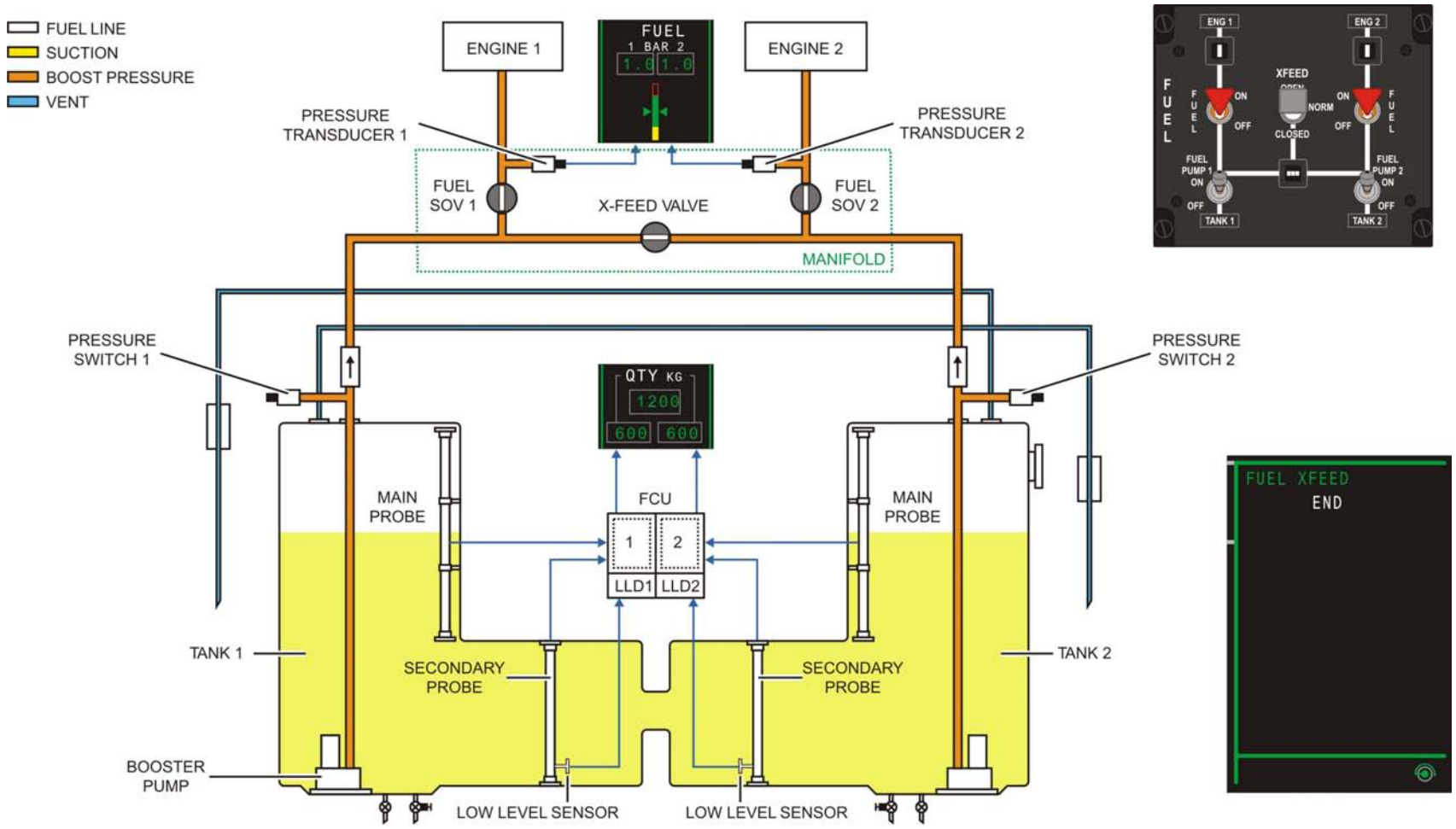
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BOOSTER PUMP NO.1 AND NO.2 ON & FUEL SOV NO.1 AND NO.2 ON



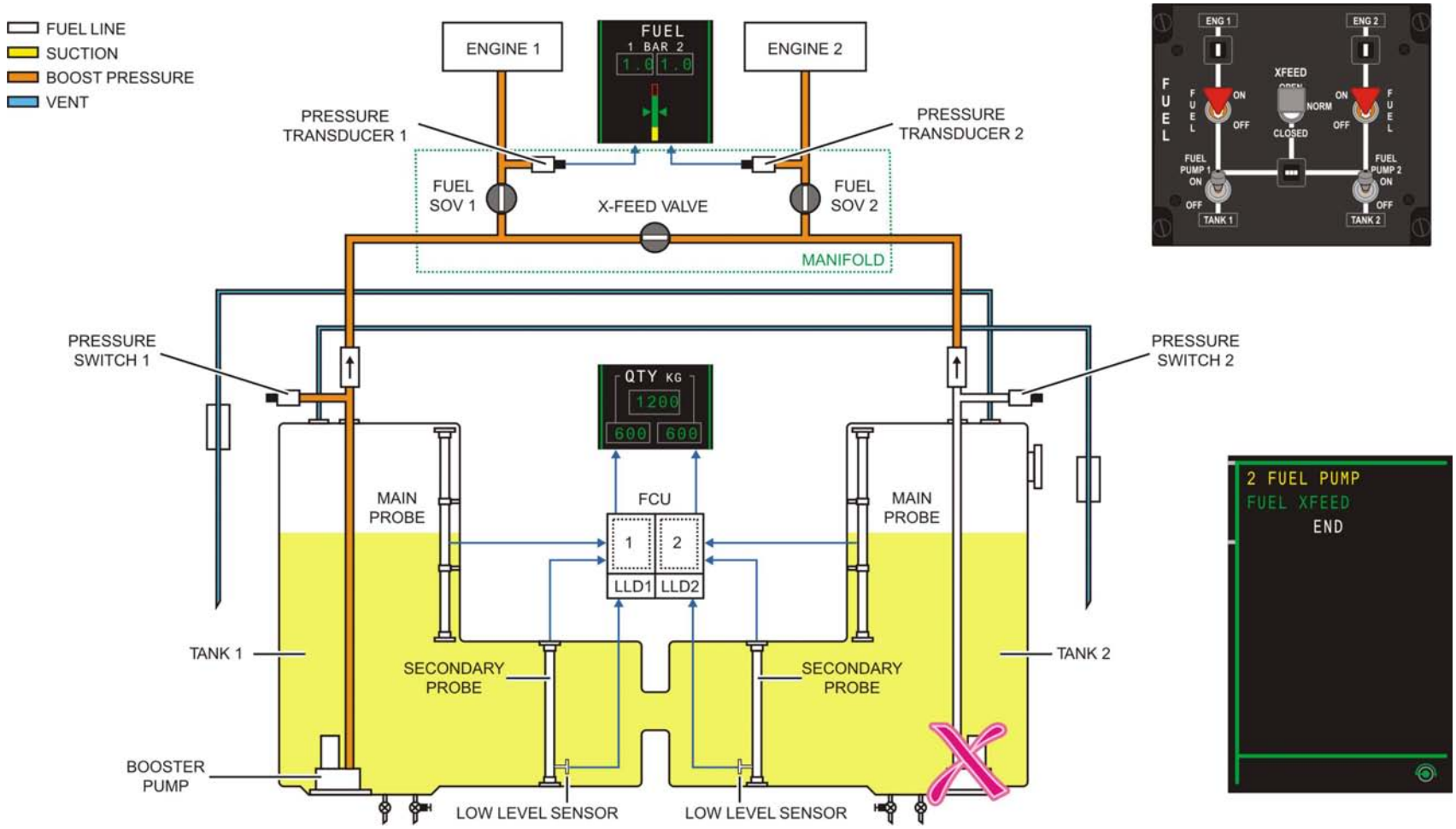
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XFEED MANUALLY CLOSED



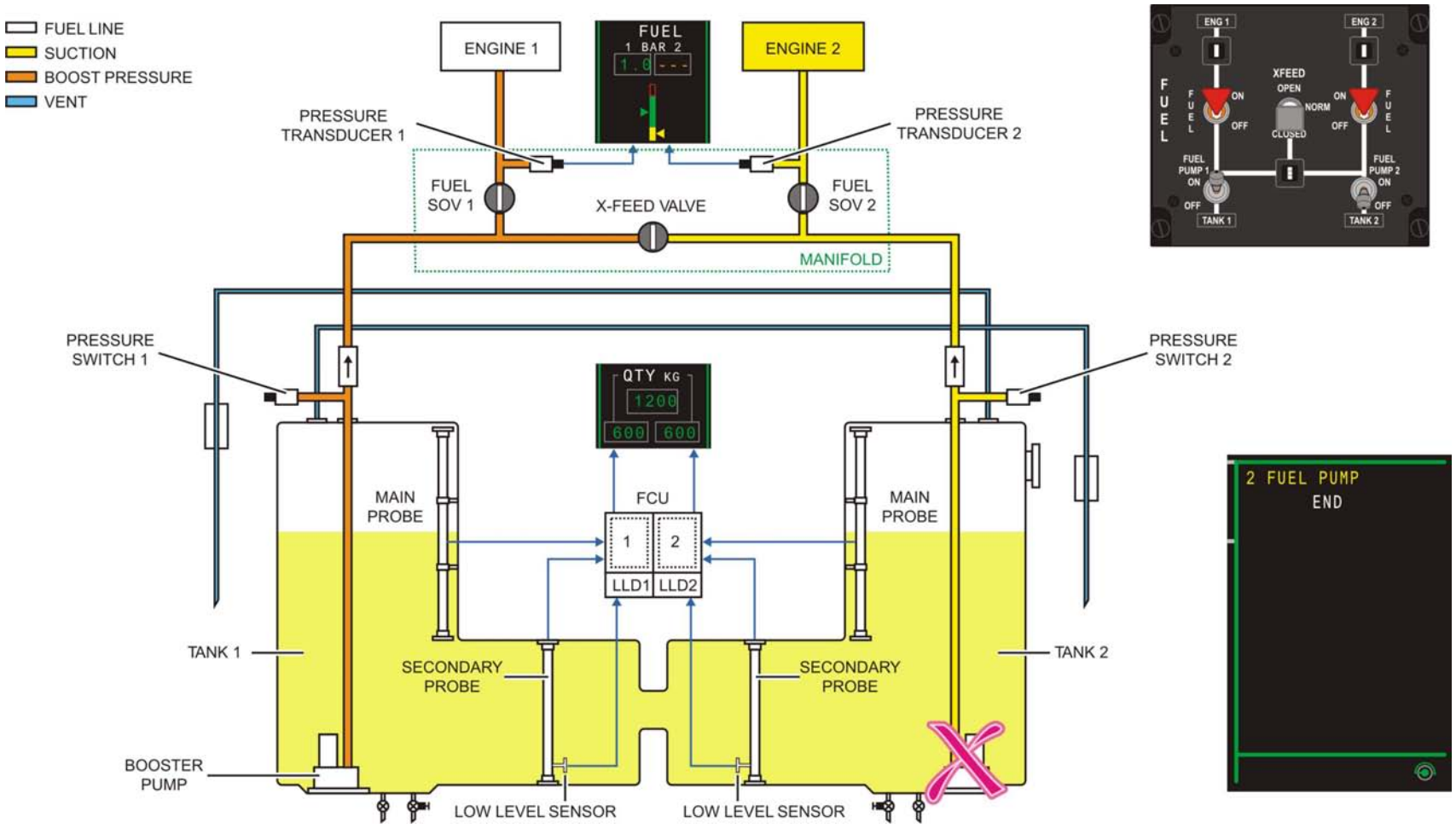
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XFEED MANUALLY OPEN



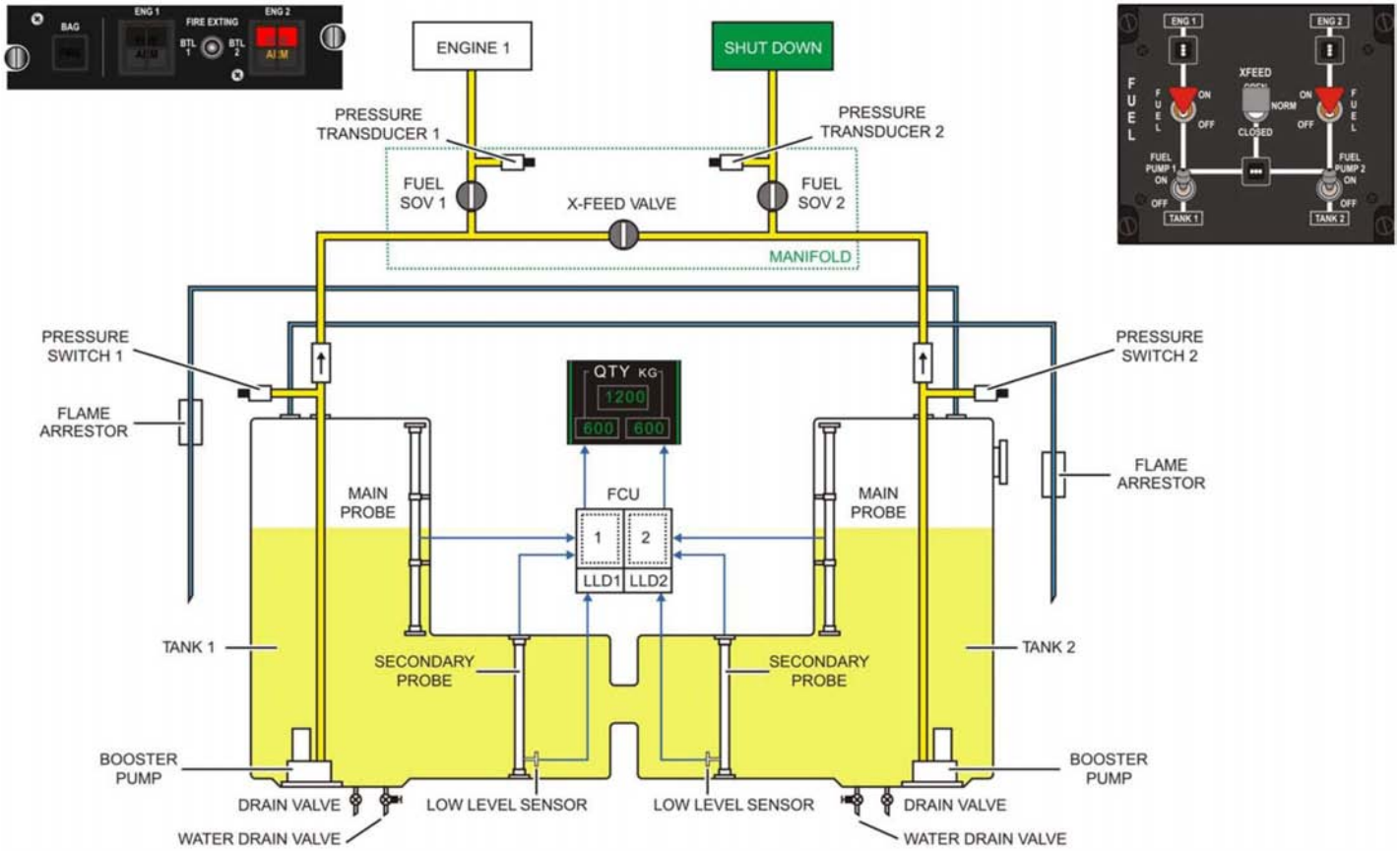
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XFEED AUTOMATICALLY OPEN (PRESSURE RESTORED)



ICN-39-A-282000-G-A0126-00659-A-01-1

XFEED MANUALLY CLOSED (SUCTION MODE FOR ENGINE NO.2)



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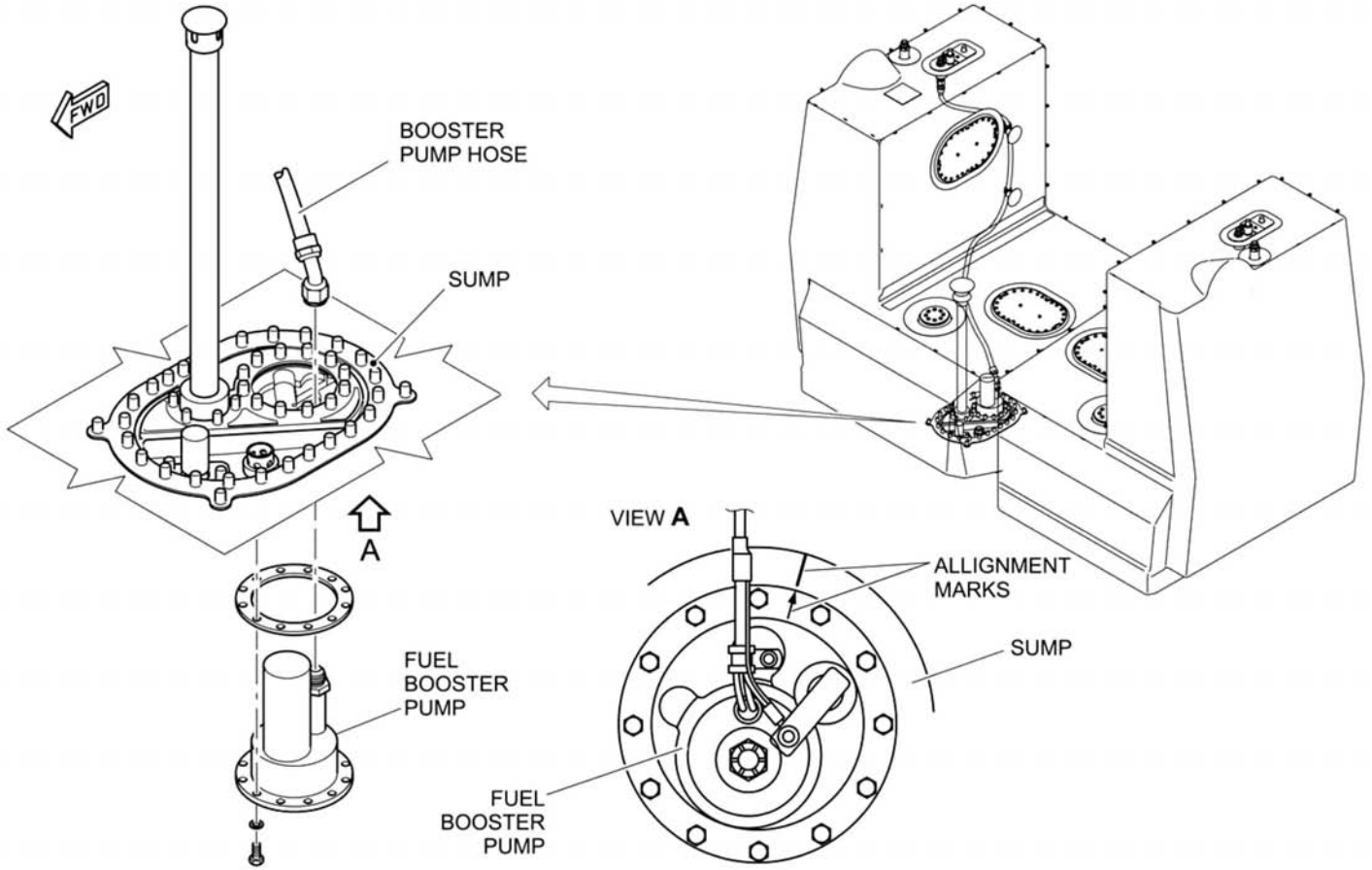
ENGINE FIRE CONDITION (ENG 2 FIRE)

BOOSTER PUMP

The DC coaster pump is installed on the sump plate at the bottom of each tank.
A slope at the bottom of the tank ensures the sump assembly be wet whenever fuel is available in the tank.
A screen on the pump housing protects the inlet from large contaminating particles.
The booster pump is provided with an integral overheat protection switch which turns off power to the motor whenever an overheat conditions exists (power is restored when the overheat is over).
The booster pump includes a cartridge which can be replaced without draining the fuel tank.
The booster pump supplies fuel through a hose inside the tank to a plate installed at the top.

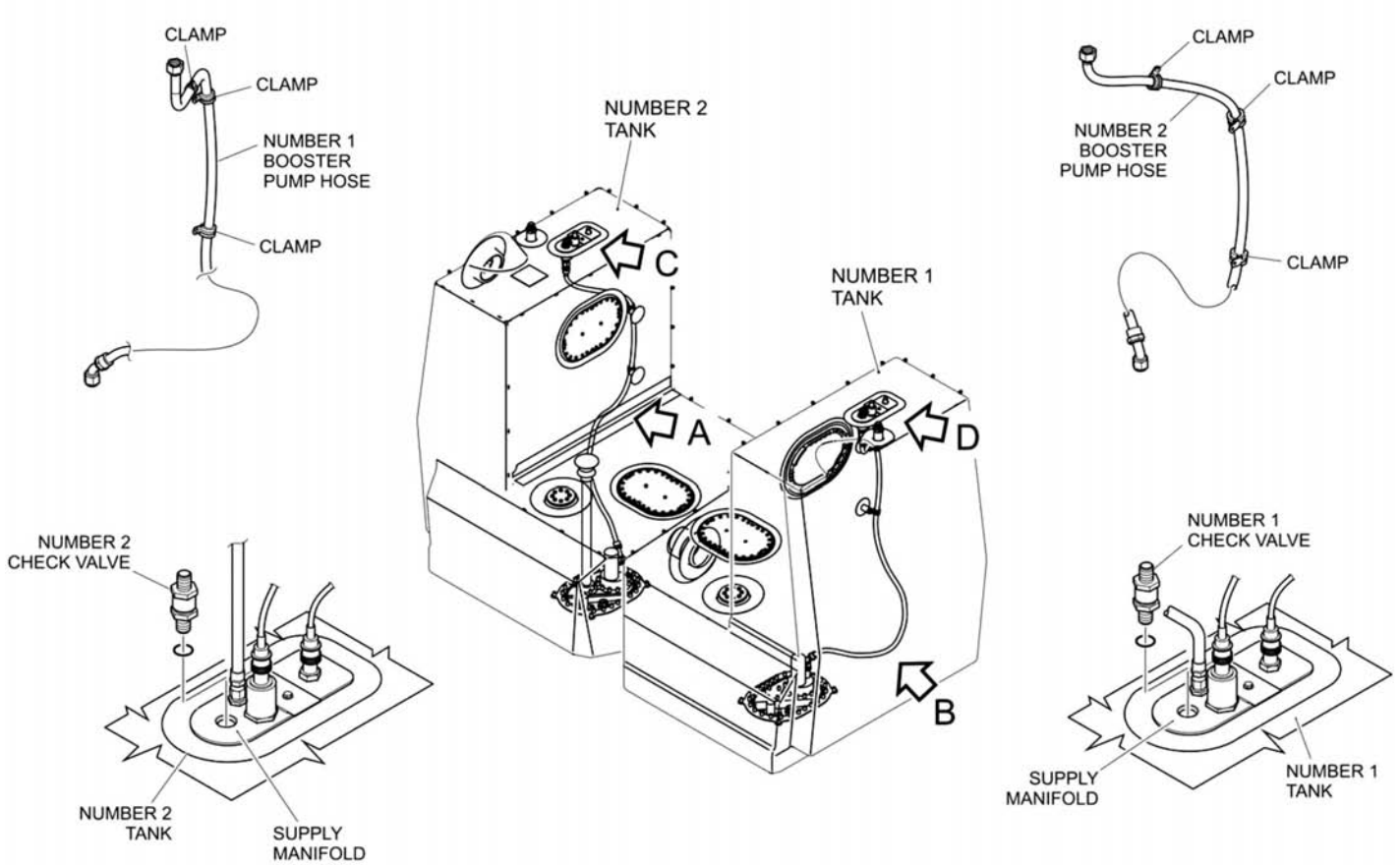
FEEDING LINE

Feeding line outlet on the top of the tank is provided with a frangible coupling.
A branch of the feeding line output on the plate goes to a pressure switch which senses pump output low pressure.
On the plate on the top of the tank is also installed a return line from the engine.
The feeding line goes through a hose to the fuel selector manifold where is installed a check valve.
In the event of failure of a pump the check valve prevents a reverse flow to the relative feeding line if the cross-feed valve is open.
The check valve incorporates a drain used to discharge over pressure applied in the reverse direction when the valve is set closed.



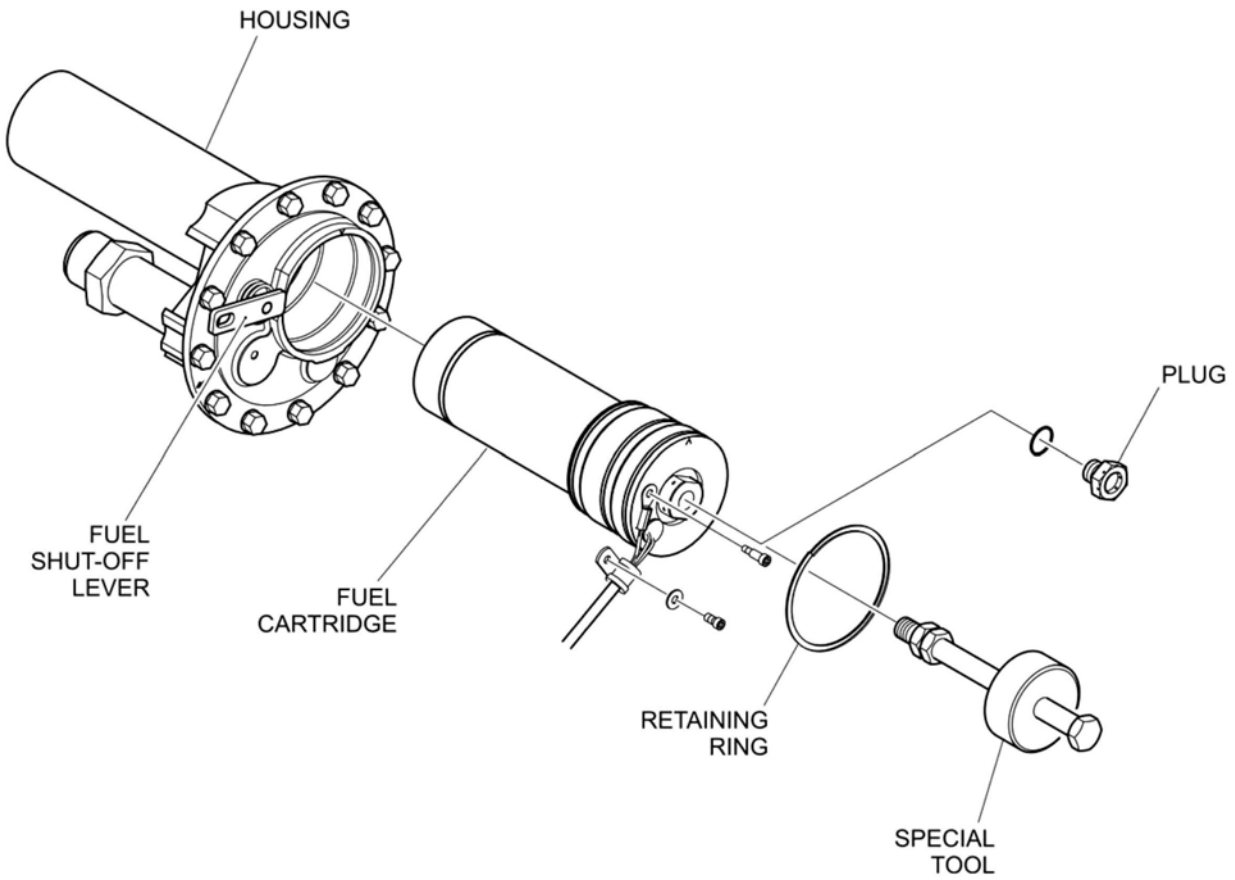
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FUEL PUMP INSTALLATION (SHEET 1 OF 2)



ICN-39-A-282000-G-A0126-00662-A-01-1

FUEL PUMP INSTALLATION (SHEET 2 OF 2)



ICN-39-A-282000-G-A0126-00663-A-01-1

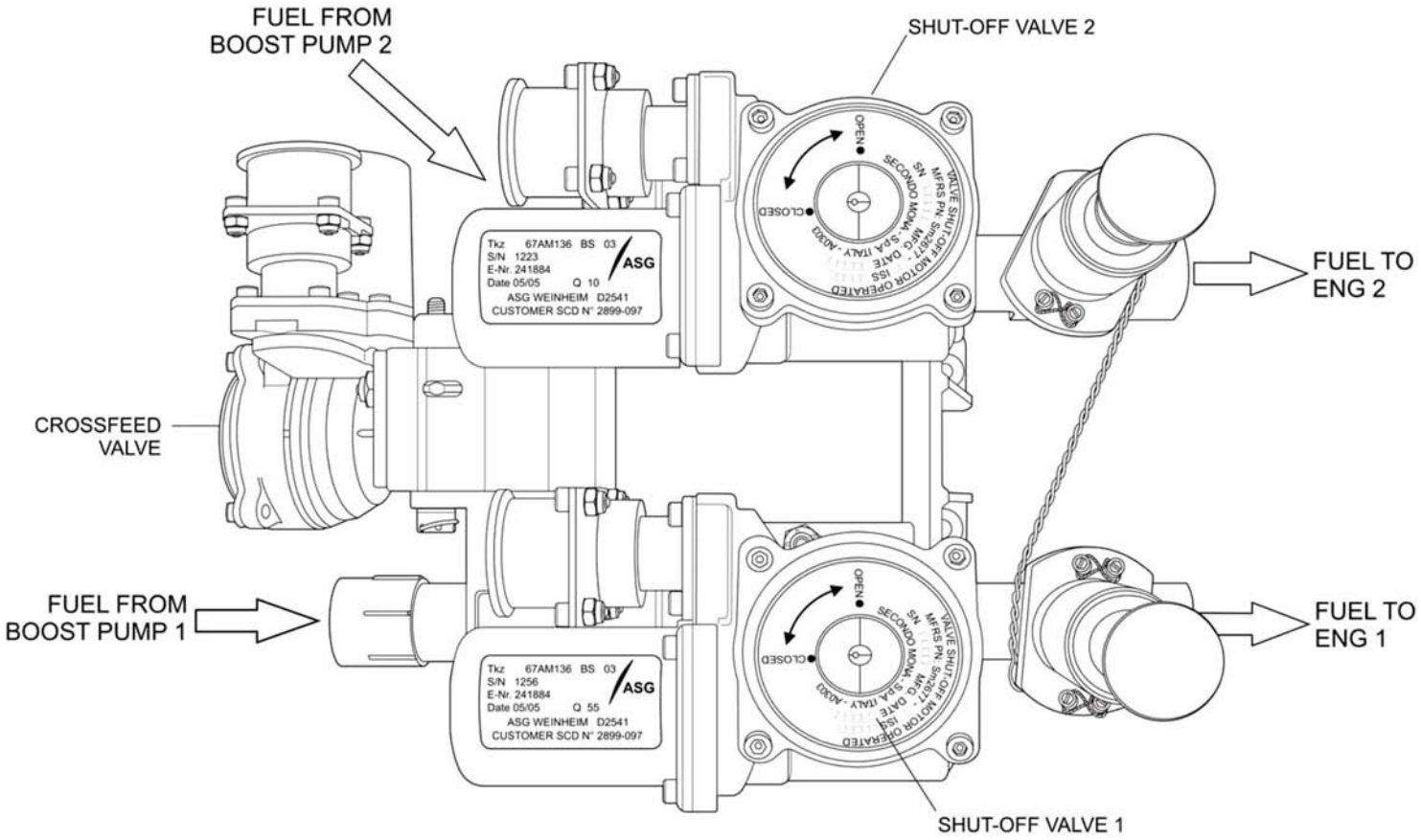
FUEL PUMP REMOVAL INSTALLATION (SPECIAL TOOLS)

FUEL MANIFOLD (SECONDO MONA)

The fuel selector manifold is composed of a DC electrically operated shut-off valve and pressure transducer for each feeding line and DC electrically operated crossfeed valve.
Two electrical connectors are located on the back side of the fuel selector manifold.
The fuel selector manifold is located on the left side of the cabin, near the horizontal engine firewall.
From the fuel selector manifold pipelines are routed to each engine firewall where are installed using a frangible self-sealing coupling.
The manifold is an assembly composed by:

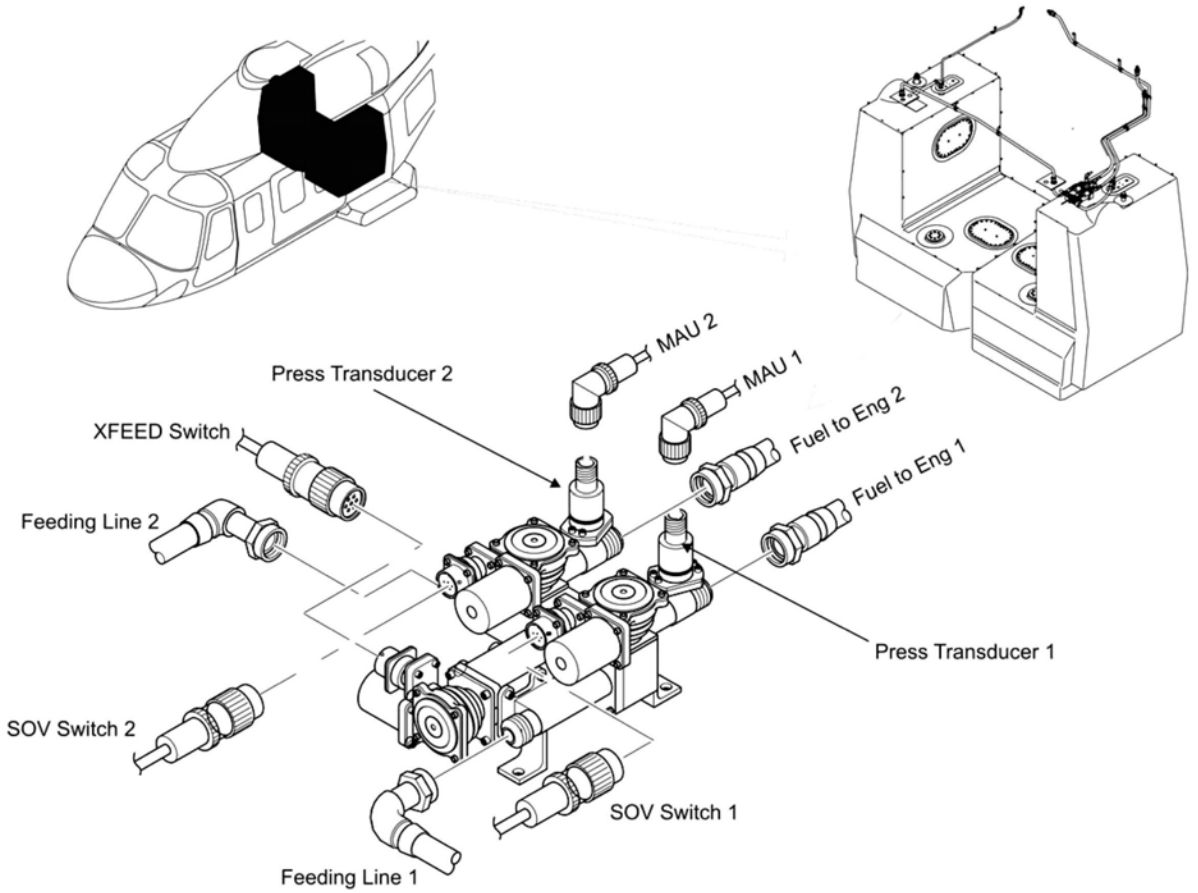
- Valves, Fuel shut-off, Motor operated
- Pressure transducer
- Machined parts for fuel lines interconnection and supports.

The valve is a “ball type” operated at 28 VDC. It has two ports and two positions.
The valve is on-line mounted.
The actuators is a high speed rotary type powered by a permanent magnet.
The actuator embodies miniature microswitches for end of travel cut off.
These provides signals for valve status indication. The rotary direction is controlled by a micro relay.
each valve is provided with a relief pressure device in order to discharge the fuel overpressure from manifold outlet port ENGINE 1 or ENGINE 2 to opposite inlet TANK 1 or TANK 2.



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FUEL MANIFOLD (SECONDO MONA)



ICN-39-A-282000-G-A0126-00665-A-01-1

FUEL MANIFOLD INSTALLATION

OPERATION

- Booster pumps are operated by a switch located on the Fuel Control Panel in the cockpit.
- Shut-off valves are also operated by a switch located on the Fuel Control Panel.
- A visual indicator on the Panel provides the valve condition.
- Engine fuel feeding is normally provided by the booster pump in the on-side tank.
- The supply from a tank can be isolated as a result of the shut-off valve selection.
- The shut-off valve can be also automatically closed as a result of an engine fire extinguishing arming.
- In normal operation the crossfeed valve is closed and the engines are supplied from their on side booster pumps.
- In the event of a specific failure condition the valve can be manually or automatically opened to crossfeed a tank supply to an opposite engine.
  A visual indicator on the Fuel Control Panel provides the valve condition in addition to the advisory FUEL XFEED to the MFD.
- The crossfeed valve can be selected open or closed at any time by setting the switch to OPEN or CLOSED.
- If the switch is set to NORM the crossfeed valve is automatically opened by a signal coming from the pressure switch.
  If the pressure switch detects a loss of pressure in one feeding line, the remaining booster pump supplies also the opposite engine without aircraft operation limitations.
- The pressure transducer in the manifold of the failed supply lines still provides the operating pressure supply when crossfeed conditions are applied.

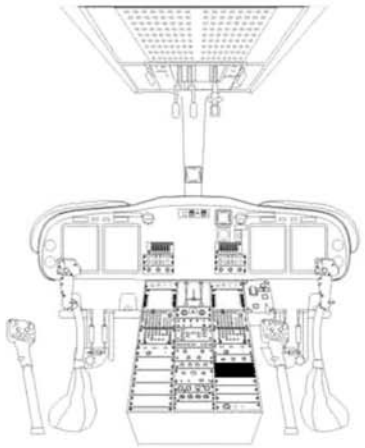
CONTROLS AND INDICATORS – TEST CONTROL PANEL

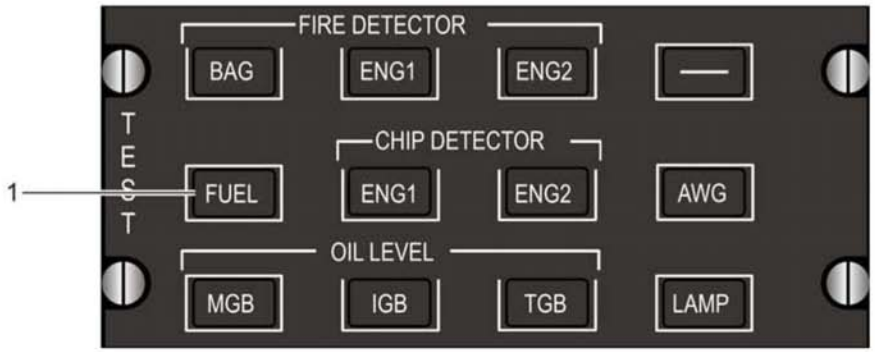
The fuel system test push-button is housed on the TEST CONTROL PANEL and is installed on the console.
The Test Control Panel provides the following function:

1 FUEL push-button ..... when pressed the TEST function takes place.
On the PFDs the white legend FUEL reverts to amber TEST and back to FUEL at the end of test. If the test fails, see the relative caution on the FUEL INDICATING section. The test duration is about 6 seconds.

NOTE B

At the power-up the test is automatically performed and the amber TEST appears.





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CONTROLS AND INDICATORS – TEST CONTROL PANEL

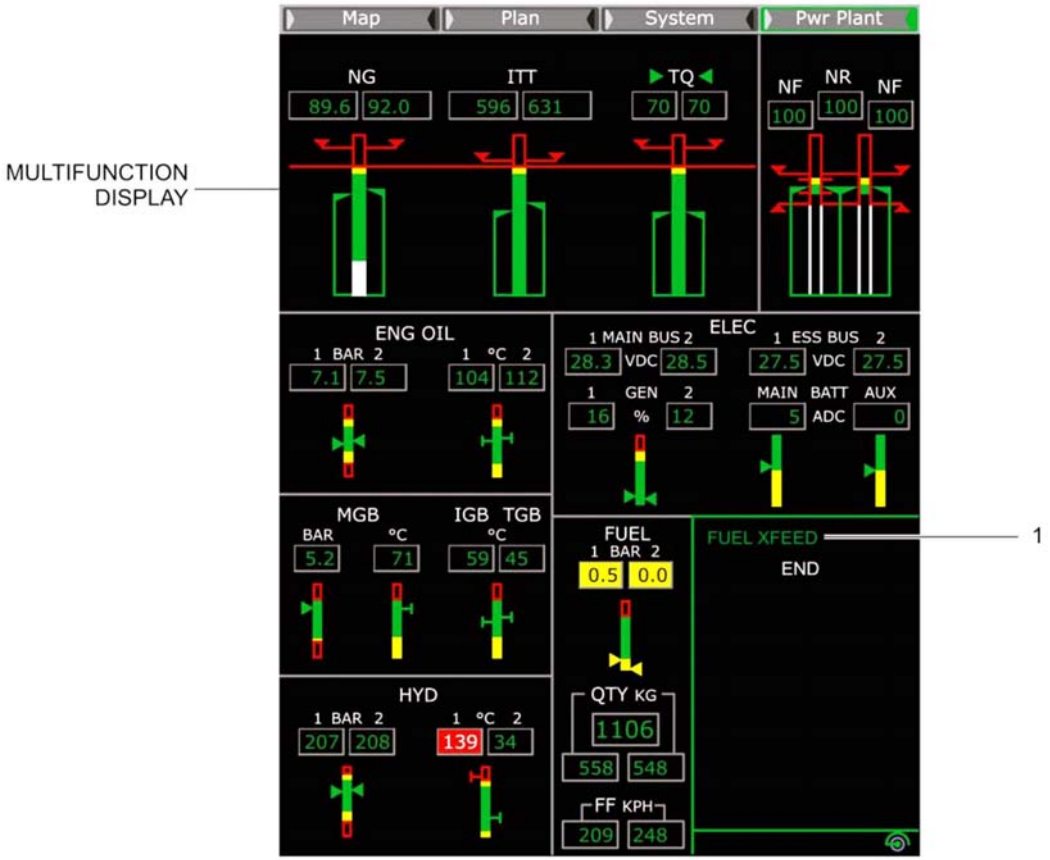
CAPTIONS

The advisory provided in the CAS message window is as follows:

1 FUEL XFEED ..... when the crossfeed valve is open (either manually or automatically).

NOTE C

When the crossfeed valve opens due to a pump failure, the relative tank quantity digital readout becomes grey when the fuel quantity drops below 228 Kg (503 lb).



ICN-39-A-282000-G-A0126-00667-A-01-1

CAPTIONS

MAINTENANCE INFORMATION

- The position of the booster pump feeding hose in the fuel tanks must be recorded before removal.
- Before booster pump removal it's required to record the position of the electrical wires.
- The booster pump cartridge is removed using a special tool.
- The fuel shut-off lever must be rotated of 180° before cartridge removal.
- Fuel spilling upon removal of the booster pump cartridge (about 0.25 l) and from the cartridge plug boss must be collected into a container.
- The position of tank supply hoses to the manifold must be recorded before their disconnection.
- Booster pump and booster pump cartridge must be installed by aligning a reference mark.
- Fuel spilling from the manifold and hoses during their removal must be collected into a container.

LEADING PARTICULARS

- Booster pump flow supplying two engines at take off power..... 634 Kg/h (1400 lb/h)
- Left booster pump (1) electrical supply..... 28 VDC ESS BUS 1
- Right booster pump (2) electrical supply..... 28 VDC ESS BUS 2
- Left shut-off valve (1) electrical supply..... 28 VDC ESS BUS 1
- Right shut-off valve (2) electrical supply..... 28 VDC ESS BUS 2
- Crossfeed valve electrical supply..... 28 VDC ESS BUS 2

SECTION 40 - INDICATING

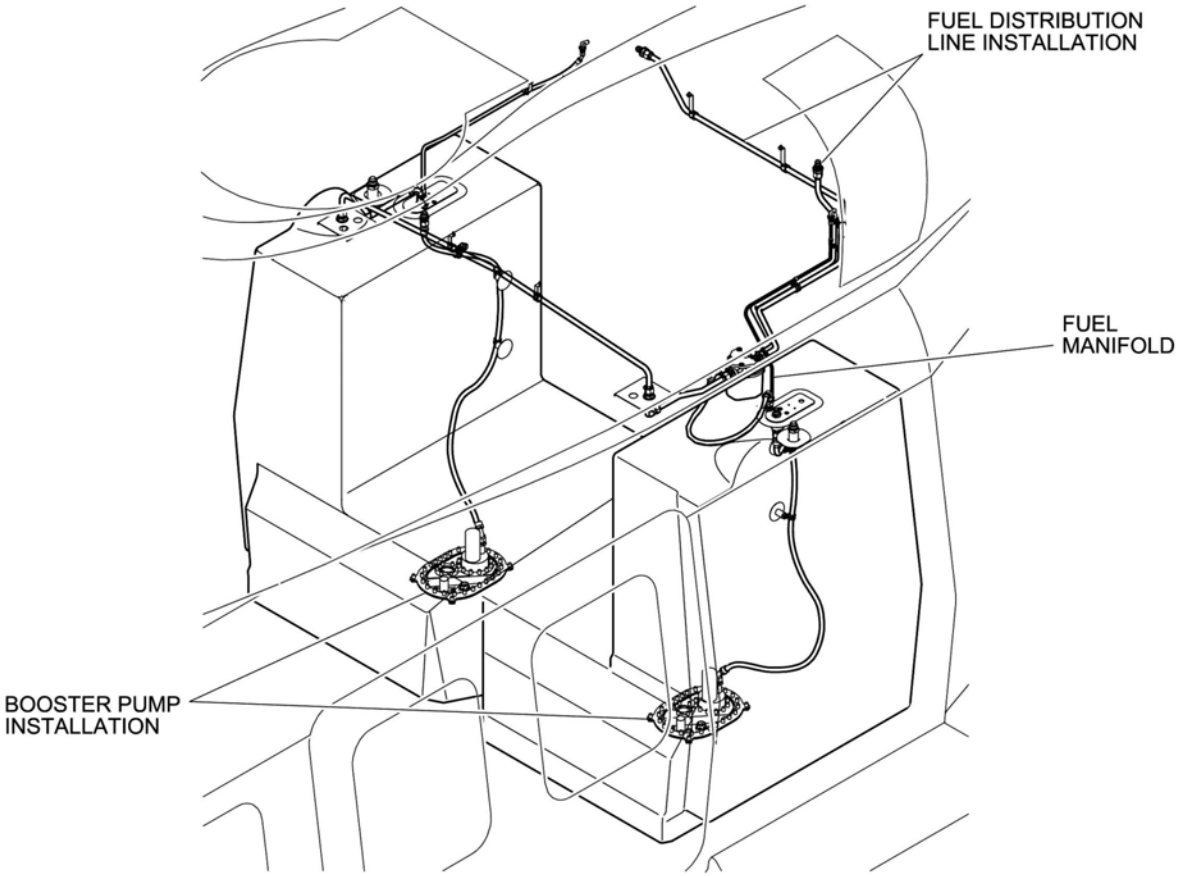
DESCRIPTION

The fuel indication system is composed of:

- Primary quantity probe
- Secondary quantity probe
- Low level sensor
- Fuel control unit
- Pressure switch
- Pressure transducer

The fuel indicating system is interfaced with:

- Central fuselage
- Electrical system
- Fuel distribution system
- Fuel storage system
- Modular Avionics Unit

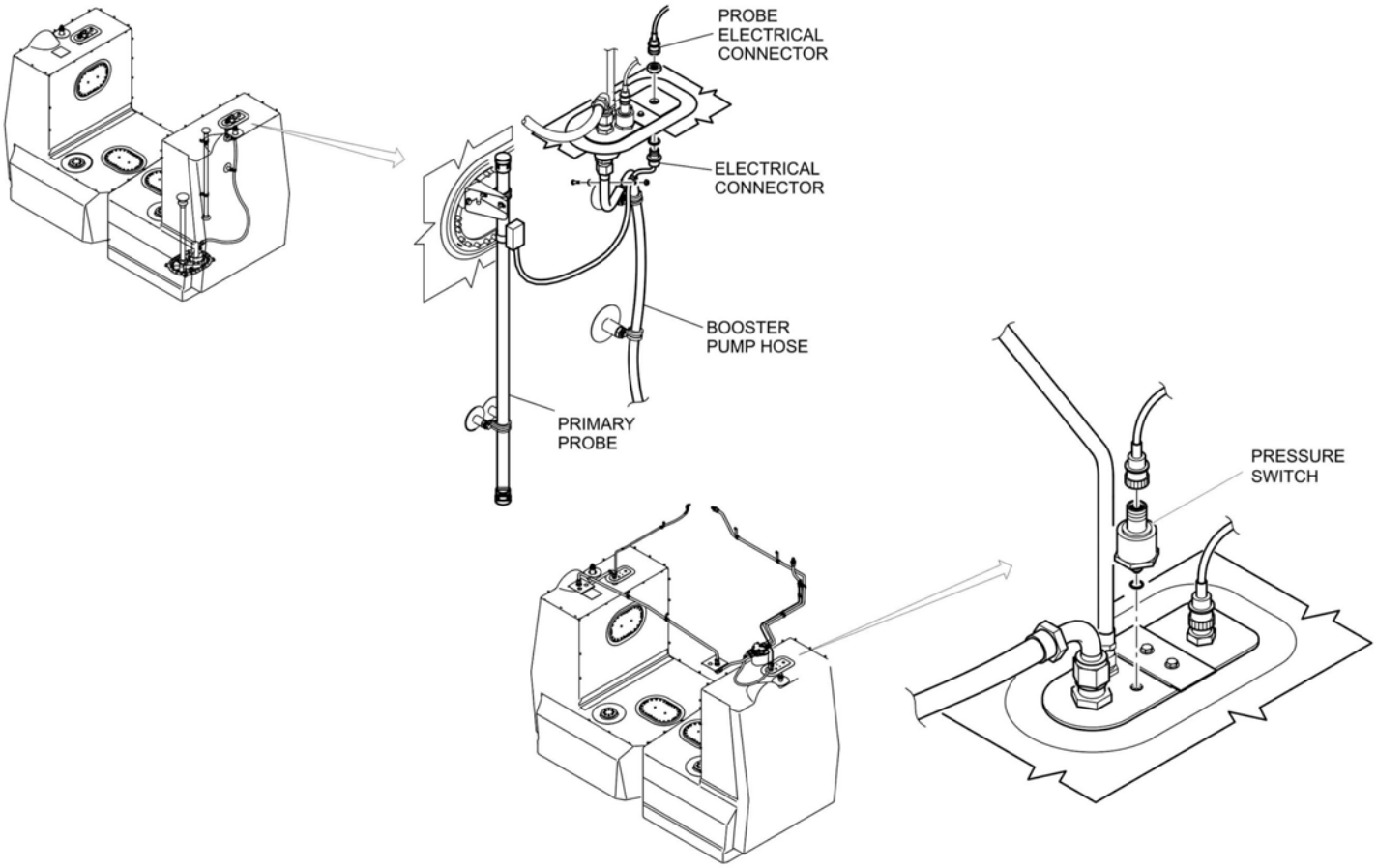


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FUEL INDICATING LAYOUT

PRIMARY QUANTITY PROBE

Inside each main tank are located one capacitance type primary probe installed on the side of the tank and a capacitance type secondary probe installed on the sump plate at the bottom of the tank.
The primary quantity probes are installed in the main tanks; the probe are attached, to a bracket on the internal side of the tank vertical wall, by a clamp, two washers and two bolt.
The bottom end of the probe is attached to a self locking insert.

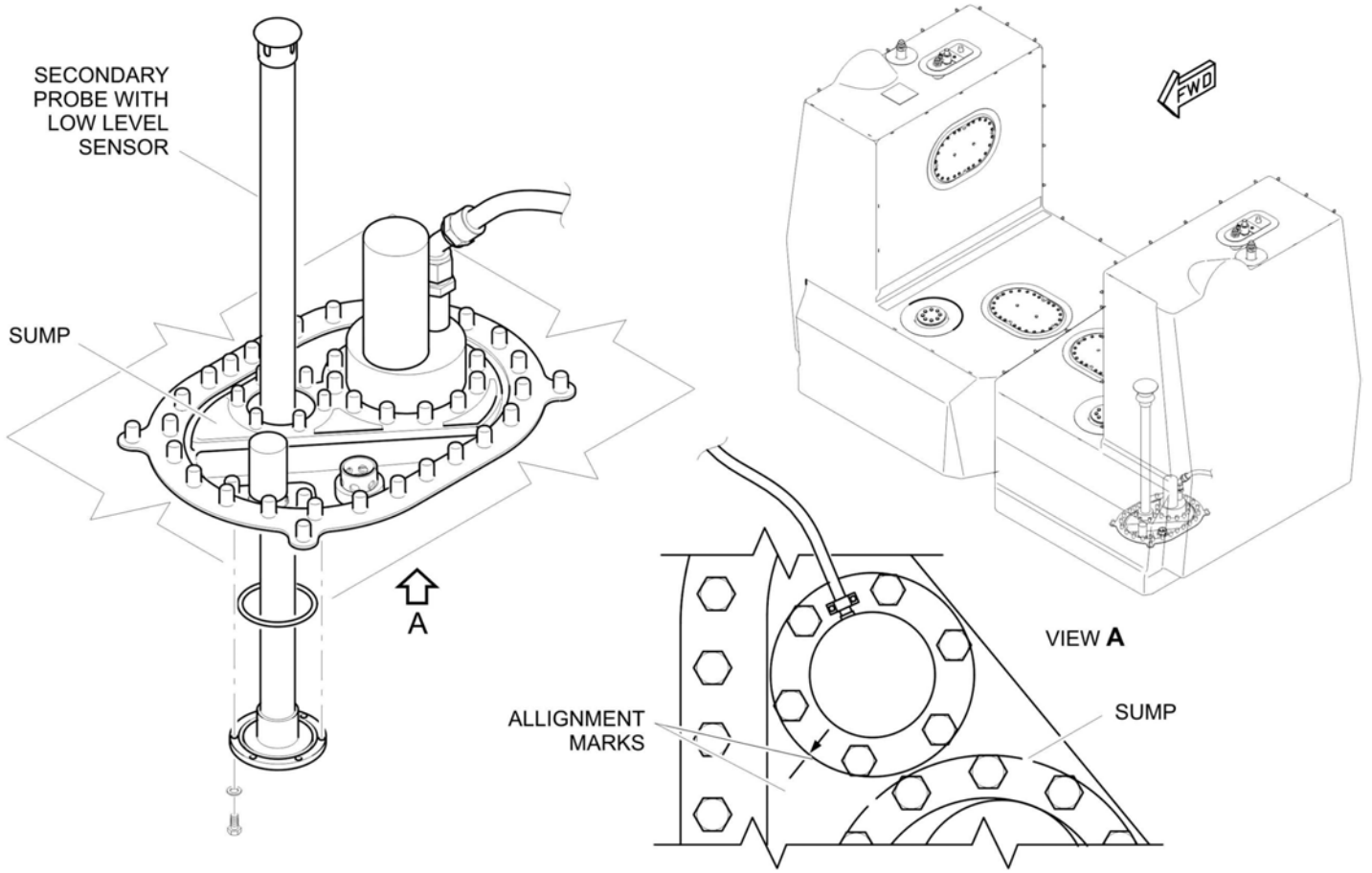


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PRIMARY PROBE INSTALLATION

SECONDARY QUANTITY PROBE

The secondary probes are installed on the sump plate of the main fuel tanks.
The probes are attached to the sump plate by six washers and six screws.
The secondary probes includes a low level sensor that provides a caution when the fuel level in the tank falls below 100 Kg.



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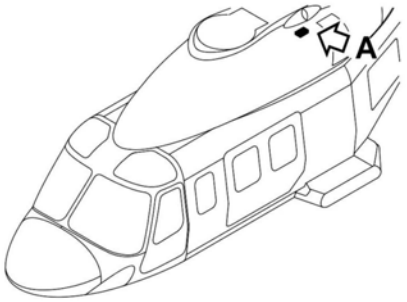
SECONDARY PROBE INSTALLATION

FUEL CONTROL UNIT

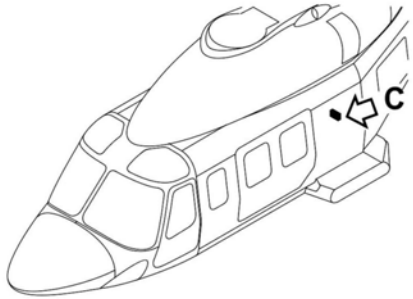
The Fuel Control Unit (FCU) is located in the right avionic bay in the rear side of the central fuselage (STA X7040) and coverts fuel height into fuel mass.
The tank geometry is part of the Fuel Control Unit software.
Specific fuel density compensation is calculated as a function of the fuel temperature.
The Fuel Control Unit is an electronic device that interfaces with the fuel quantity probes and low level sensors; the output is provided to the Modular Avionics Unit.
The FCU function are below:

- Reads and interprets fuel quantity data from the fuel probes and transmit digital data via ARINC429 serial data bus to the MAUs.
- Provides an excitation current to the capacitance probes.
- Reads and interprets low level sensor impedance and transmits digital data via 429 and a discrete signal to the MAUs.
- Provides an excitation events to the thermistor low level sensors.
- Performs a BIT at power up.
- Performs a continuous BIT.
- Performs a BIT when manually initiated.

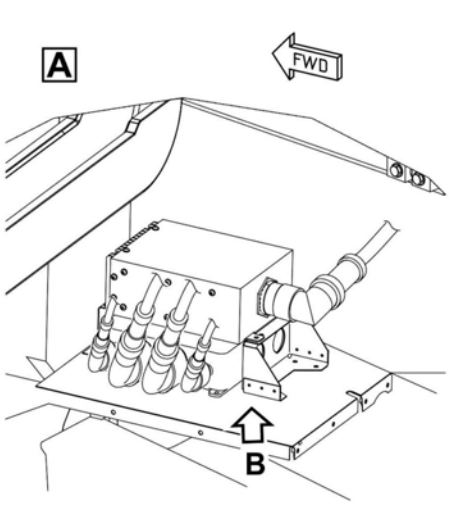
The Fuel Control Unit is designed to have two independent and separated channels with data link interfacing so that comparison can be made between channels and any failure of one does not affected the proper operation of the other.
The fuel indicating also includes the pressure switch located on the engine feeding line and the pressure transducers (located on the manifold).
They are interfaced with the Modular Avionics Unit.

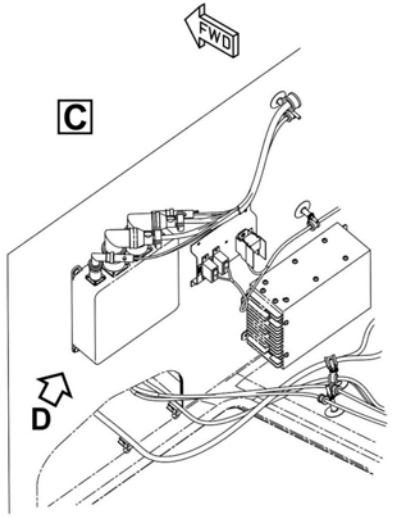


HELICOPTER (1J)



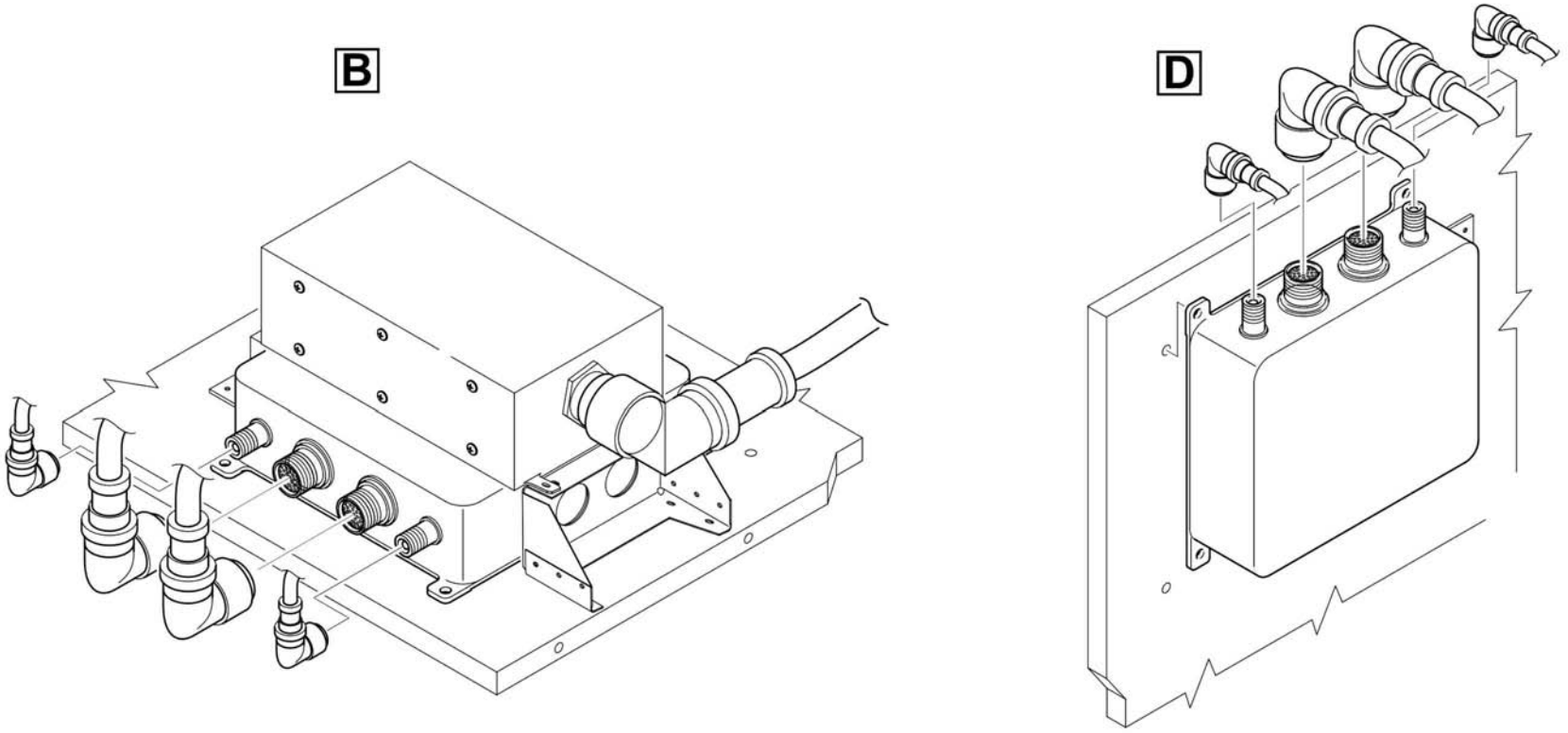
HELICOPTER (1L)





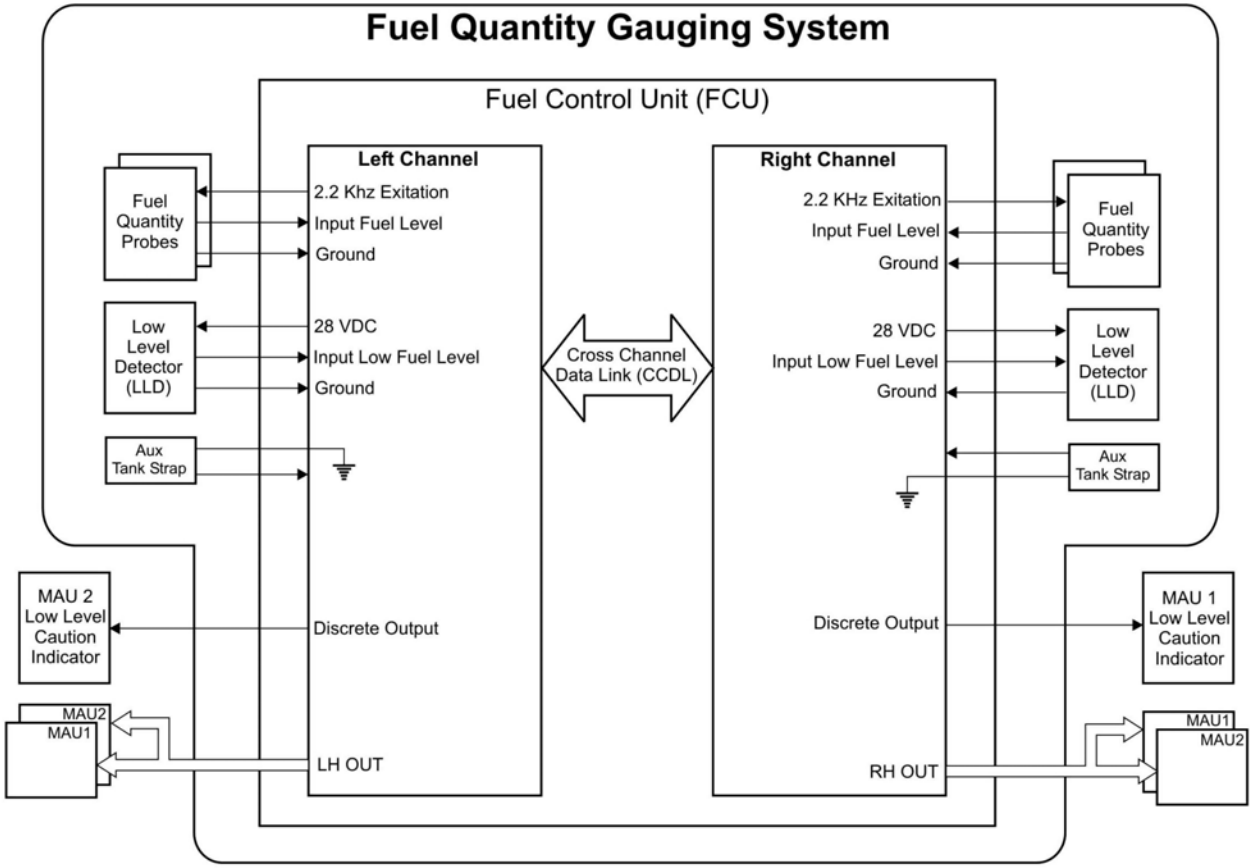
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FCU INSTALLATION (SHEET 1 OF 2)



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FCU INSTALLATION (SHEET 2 OF 2)

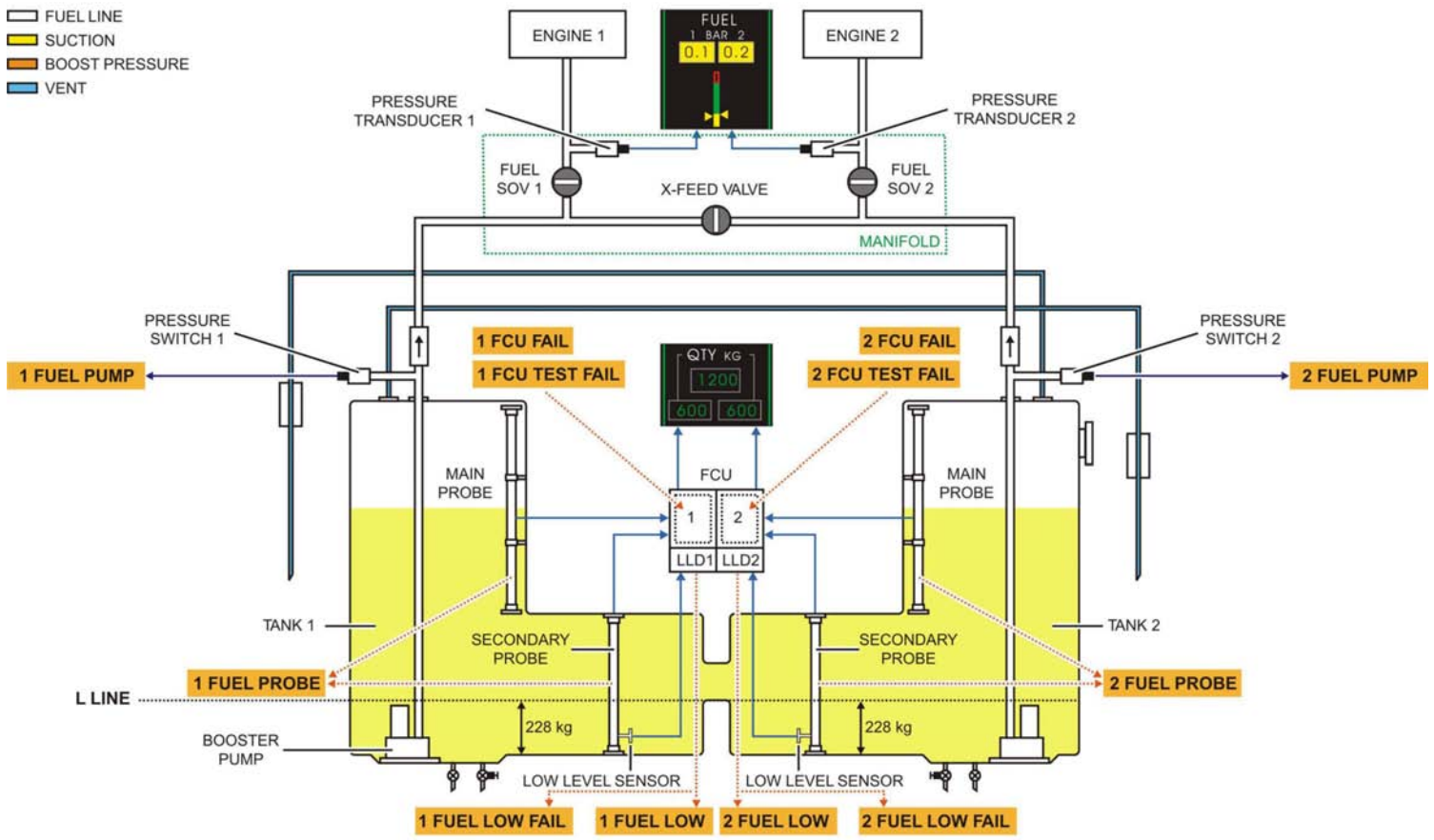


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FCU INTERFACE

OPERATION

- When the primary probe is immersed in fuel it provides the height of fuel to the FCU and then converted in fuel mass to the MFD.
- When the primary probe loses contact with fuel, the measurement is provided by the secondary probe.
- When a primary probe fails to provide a signal when measuring, the FCU signals the loss of the probe as a failure condition providing the caution 1 (2) FUEL PROBE to the MFD and use the secondary probe in the same tank to provide the fuel quantity indication.
- If both primary probes fail the cautions 1 (2) FUEL PROBE are provide a fuel quantity indication relative to their maximum height (lower than actual fuel quantity).
- When a secondary probe fails to provide a signal when measuring, the relative fuel quantity indication is set to zero.
- If both secondary probes fail the cautions 1 (2) FUEL PROBE are provided to the MFD and the fuel quantity indication is zero.
- Because of the interconnect between the two tanks, when the fuel level falls below the secondary probes provide unique information in case of failure.
- If a secondary probe fails in the right (left) tank while fuel level is below the interconnect the quantity of fuel reported is zero while the secondary probe in the left (right) tank reports its actual fuel level.
  The fuel quantity indication in this failure condition is then less than the actual.
- The FCU uses the roll attitude compensation in each channel to determine the fuel quantity calculation.
- When the fuel quantity falls below the low sensor the caution 1 (2) FUEL LOW is provided to the MFD.
- Failure of the low sensor provides the caution 1 (2) FUEL LOW FAIL.
- If the BIT of the Fuel Control Unit detects internal left/right channel failure, the caution 1 (2) FCU FAIL is provided to the MFD.
- A manual test of the FCU can be operated (only on ground) by selecting the pushbutton FUEL in the Test Control Panel located in the console. If the result is unsuccessful the caution 1 (2) FCU TEST FAIL is provided to the MFD.
- The pressure transducer in the manifold provides continuous pressure monitoring to the MFD.
- In the event of fuel boost pump low pressure, a pressure switch provides the signal for the caution 1 (2) FUEL PUMP to the MFD.



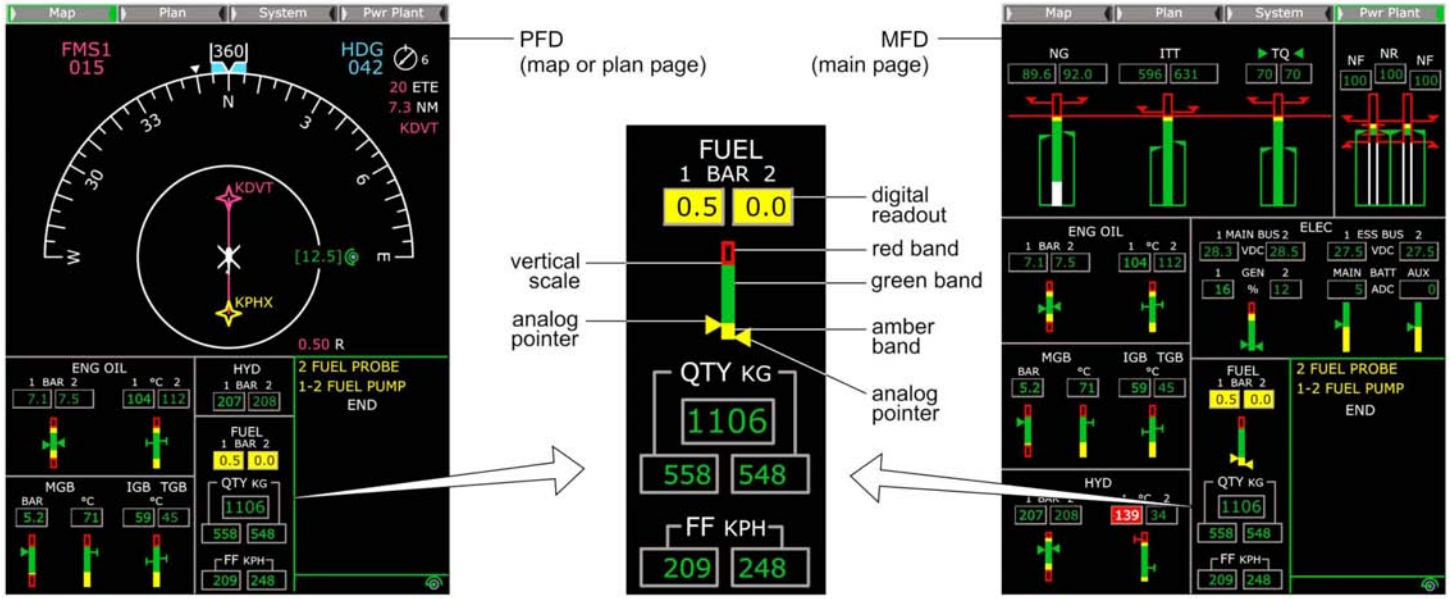
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FUEL QUANTITY INDICATING PHILOSOPHY (SHEET 1 OF 2)

FUEL QUANTITY INDICATING PHILOSOPHY (SHEET 2 OF 2)				
FUEL LEVEL	**PROBE FAILURE**	**TOTAL FUEL**	**ROLL COMP. (between ± 5°)**	**NOTE**
above Q	no failures	M1 + M2	YES	
below Q	no failures	S1 + S2	YES	
above Q	M1	S1 + M2	default to 0°	LH tank shows a fuel level lower than effective
above Q	M2	M1 + S2	default to 0°	RH tank shows a fuel level lower than effective
above Q	S1	M1 + M2	YES	
above Q	S2	M1 + M2	YES	
below Q	M1	S1 + S2	YES	
below Q	M2	S1 + S2	YES	
below Q	S1	0 + S2	default to 0°	LH tank shows a fuel level equal to zero
below Q	S2	S1 + 0	default to 0°	RH tank shows a fuel level equal to zero
above Q	M1 and M2	S1 + S2	YES	RH / LH indications lower than effective
above Q	S1 and S2	M1 + M2	YES	
below Q	M1 and M2	S1 + S2	YES	
below Q	S1 and S2	0 + 0	default 0°	RH / LH tanks show zero fuel

NOTE A

IF THE FUEL LEVEL IS ABOVE THE COLLECTOR LEVER (L), the FCU will compensate for roll attitudes between ± 5°. When the fuel level is below the collector no compensation is made.

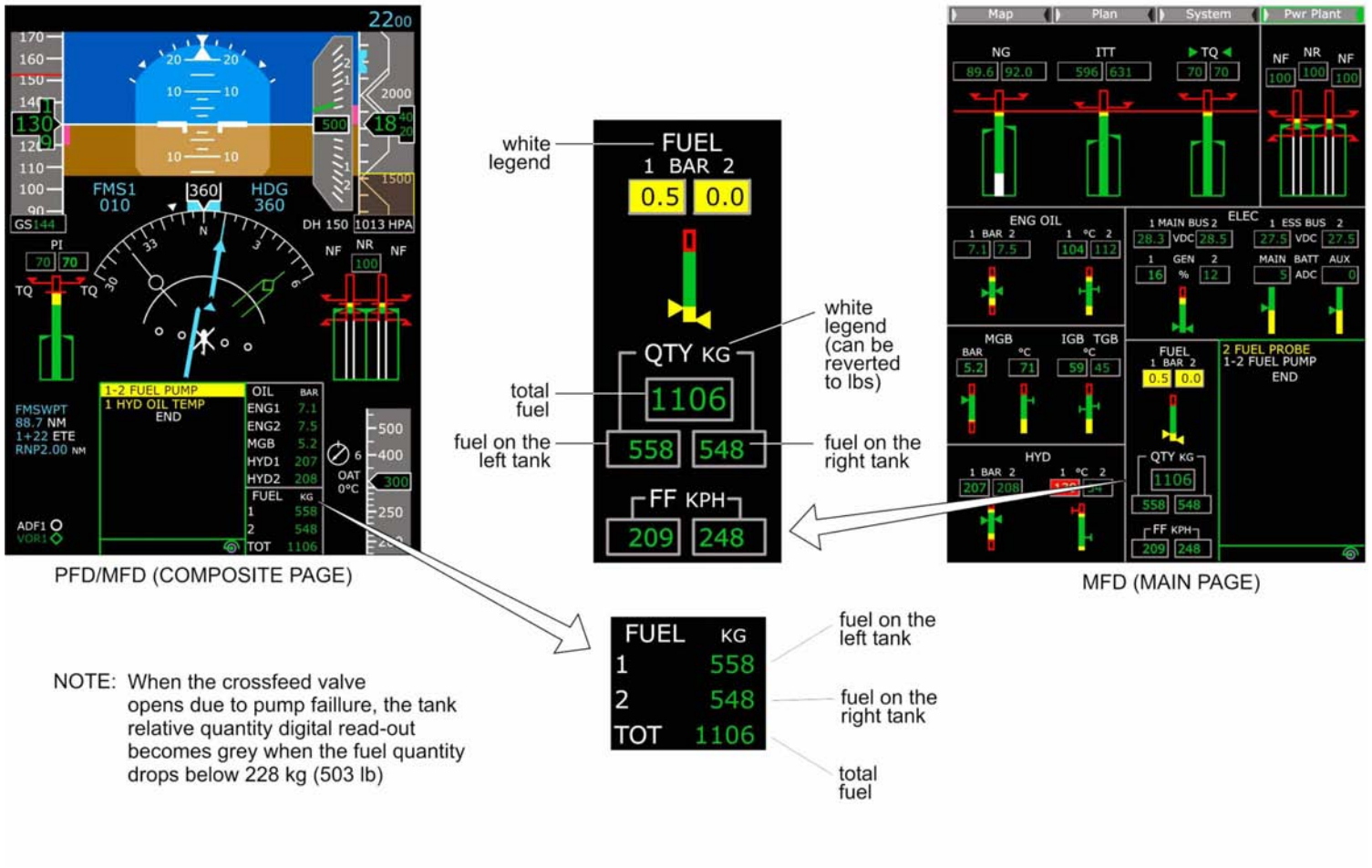


DISPLAY PRESENTATION

- One single vertical scale with one analog pointer and one digital readout per system
- range : 0 to 2,7 bar for analog scale
  : 0 to 2,7 bar for digital readout
- resolution : 0,1 bar for digital readout
- marking and range : amber band (0 - 0,5 bar)
  : green band (0,6 - 2,1 bar)
  : red line (2,2 bar)
  : red band (2,2 - 2,7 bar)

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FUEL PRESSURE INDICATION



ICN-39-A-284000-G-A0126-00571-A-01-1

FUEL QUANTITY INDICATING

MAINTENANCE INFORMATION

- The main probe must be installed with the UP mark pointing at the top.
- The secondary probe must be removed 7 installed with the helicopter lifted on jacks.
- The secondary probe must be installed with the mark aligned with the mark on the sump.
- An operational test of the quantity indicating system must be carried out after main or secondary probe and Fuel Control Unit replacement.
- An Operational test of the pressure indicating must be carried out after pressure switch and Fuel Control Unit replacement.

LEADING PARTICULARS

- Fuel pressure switch:
  - Decreasing..... below 0.6 bar (8.7 psi)
  - Increasing..... above 0.7 bar (10.1 psi)
- Fuel operating pressure:
  - Red band..... 2.2 thru 2.7 bar (32 thru 39 psi)
  - Red line..... 2.2 bar (32 psi)
  - Green band..... 0.6 thru 2.1 bar (8.7 thru 30.5 psi)
  - Amber band..... 0 thru 0.5 (o thru 7.2 psi)
- Fuel low level quantity..... below 92 Kg
- Fuel Control Unit electrical supply..... 28 VDC ESS BUS 1 and 28 VDC ESS BUS 2
- Left and right primary probe electrical supply..... FCU
- Left and right secondary probe electrical supply..... FCU
- Left and right low level sensor electrical supply..... FCU
- Left pressure transducer (1) electrical supply..... MAU1 (10 VDC)
- Right pressure transducer (2) electrical supply..... MAU2 (10 VDC)

CAS MESSAGES	
1(2) FUEL LOW FAIL	Respective Low level Sensor is faulty
1(2) FUEL LOW	Fuel level in respective tank is 92 kg or below

CAS MESSAGES	
1(2) FUEL PUMP	Respective fuel boost pump pressure drops below 0.6 bar. Goes out when above 0.7 bar.
1(2) FUEL PROBE	Respective main or secondary probe is faulty
1(2) FCU FAIL	Right or Left FCU channel has failed
1(2) FCU TEST FAIL	Right or Left cheìannel of FCU fails the BIT test
1(2) FCU TEST FAIL	Loss of communication to a single MAU is detected for FCU 1 or 2. (The maintenance message in the CMC is A429/RS 422 BUS)
FUEL XFEED	When the Crossfeed valve is OPEN (manually or automatically)

CHAPTER
29
HYDRAULIC POWER

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SECTION 10 - MAIN

DESCRIPTION

The Main Hydraulic Power System is composed of by two independent circuits.
Each system is composed of two self-regulating hydraulic pumps (one mechanical and one electrical driven from the Auxiliary Hydraulic Power System for circuit 1, two mechanical driven for circuit 2) providing pressure flow to a Power Control Module (PCM).
The mechanical hydraulic pumps are driven continuously by the Main Gearbox, the electrical hydraulic pump (when operated by the pilot) only for ground pre-flight check of the flight controls servo actuators for a limited period of time.
The circuit 1 mechanical pump (Hydraulic Power Supply 1) is located on the top of the left engine input stage of the Main Gearbox, the electrical pump is located on the upper deck in front of the Main Gearbox.
The circuit 2 mechanical pumps are located on the Main Gearbox, the Pump 2 on the top of the right engine input stage, the pump 4 on the left front side.
The Power Control Module houses all the equipment required for the hydraulic operation such as pressurized reservoir, pressure and return filters, check valves, ground equipment operation components, flight controls and utilities circuit solenoid shut-off valves, bleed/drain valve, inlet and outlet pressure switches, circuit pressure sensor, oil temperature sensor and switch, oil level microswitches.
The Power Control Module 1 is located on the left side of the upper deck in front of the Main Gearbox, the Power Control Module 2 on the right side of the upper deck in front of the Main Gearbox.
A Tail Rotor Shut-Off Valve, is installed in the circuit 2 only, between the Power Control Module and the tail rotor servo actuator.
The Tail Rotor Shut-Off Valve is located on the right side of the upper deck in front of the Main Gearbox.
The Main Hydraulic Power System can be supplied on ground for maintenance activities through a Ground Equipment connected with quick disconnect couplings located on the Power Control Modules.
Each pump is connected to the PCM1 or PCM2 with three flexible hoses. From each PCM two flexible hoses are routed to the main rotor servo actuators, circuit 1 on the top, circuit 2 on the bottom.
From each PCM two rigid pipelines are routed to the tail servo actuator (in the circuit 2 to the Tail Rotor Shut-Off Valve first) and two rigid pipelines are routed to the utilities system.
Hydraulic pipelines on the structure upper deck are connected with each PCM with flexible hose used to drain the reservoir and pump seal leak and PCM reservoir leak.
The Main Hydraulic Power System is interfaced with:

- Auxiliary Hydraulic Power System
- central fuselage
- tail unit
- flight control system
- utilities system
- electrical system
- Main Gearbox
- Modular Avionics Unit

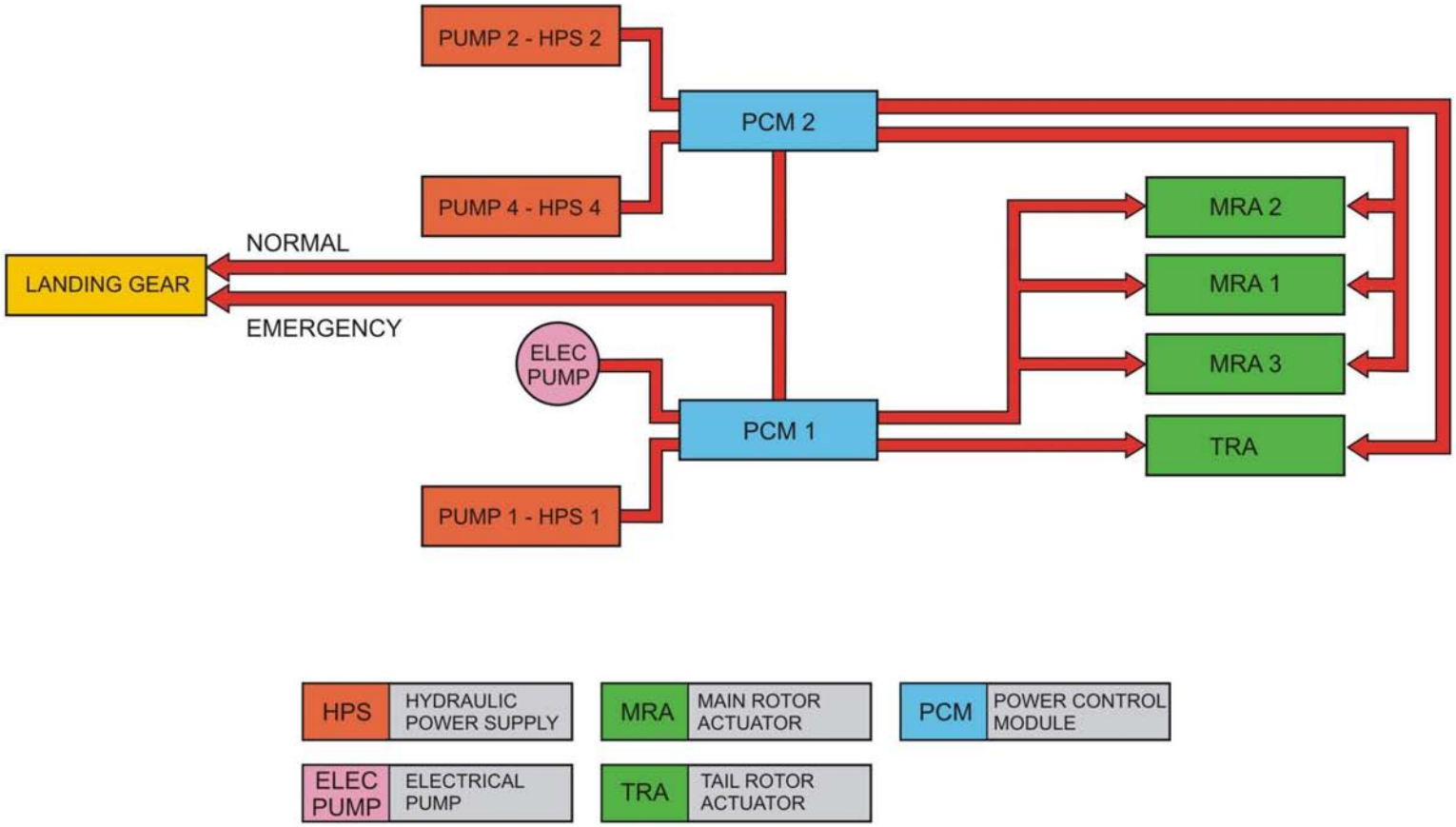
The Main Hydraulic Power System is designed to provide the required safety conditions by:

- A Tail Rotor Shut Off Valve used to isolate the circuit 2 supply to the tail rotor servo actuator in case of excessive leakage in order to provide full flow capabilities to the main rotor servo actuators.

- A set of microswitches in the Power Control Module used to isolate the tail rotor servo actuator and/or the hydraulic utilities as a function of the oil level in the Power Control Module reservoir.
- Interlocks in the Hydraulic Control Panel which prevents depressurization of both circuits at the same time or a circuit in case of failure of the other.
- Equipment and installations positioned on the upper deck will not be adversely affected by fire and will provide their required functions without failures, external leakages or system blockage for a duration of at least 15 minutes to allow a safe landing of the helicopter following the start of a fire.
- Fire resistance is improved by using hydraulic fluid with improved fire resistant properties, titanium skins to cover the Power Control Module reservoir, fire resistant silicon sleeves fitted on connection hoses, rigid tubes made of titanium, seal leakages routed overboard through titanium drain lines and installation of the hydraulic equipment to ensure that hydraulic fluid leaks or sprays will not impinge on high temperature surfaces.
- Hydraulic tubing and equipment installation designed to limit the damage caused by a potential hazard to affect only one circuit obtained by specific routing of the tubing and separation of the two hydraulic circuits.

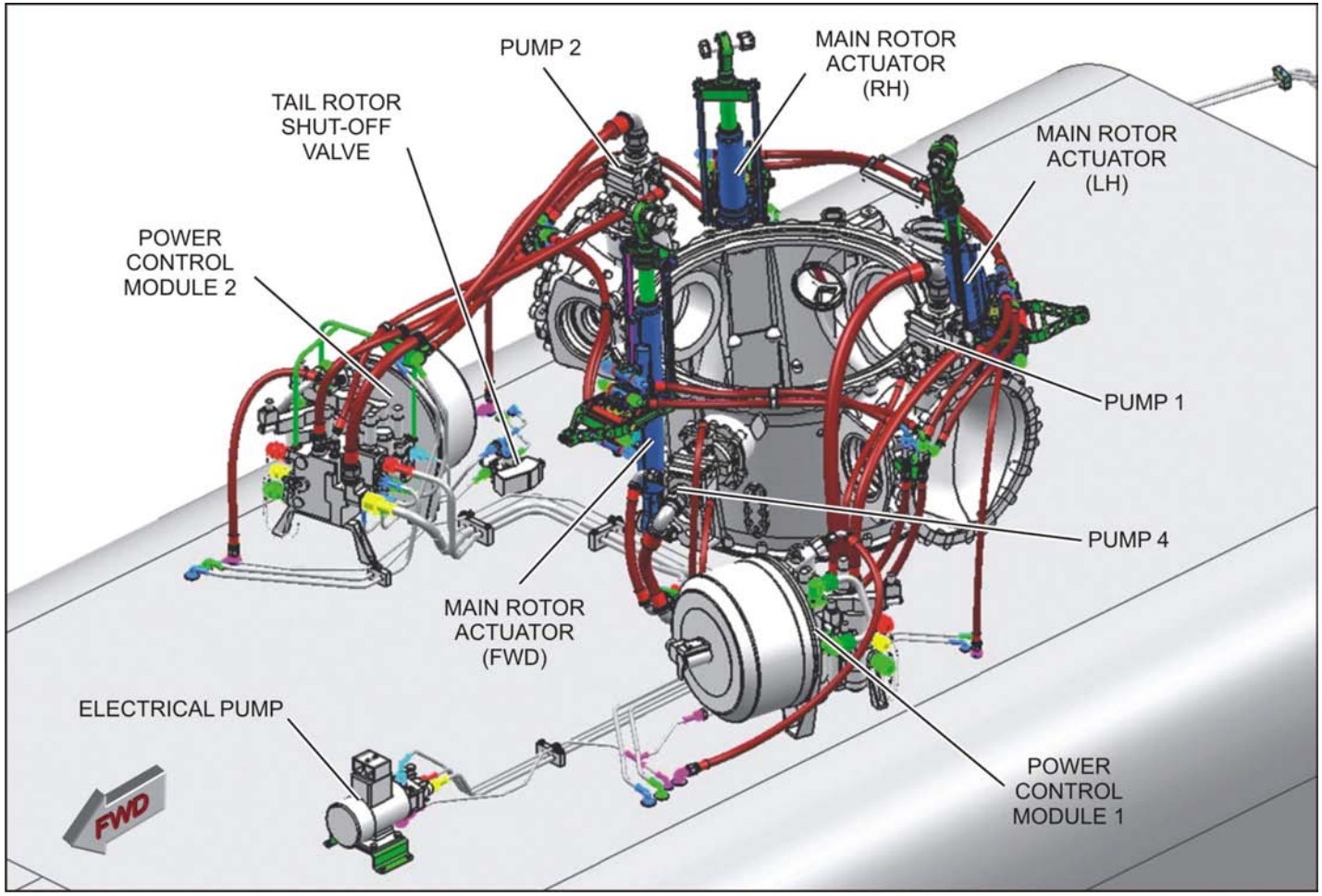
OPERATION

- From the PCM pressurized reservoir, oil is delivered to each pump through a dedicated suction line.
- When a demand is made on the system (pressure drop due to a flight controls or utilities operation) the pump adjusts itself to provide the oil flow to restore the nominal pressure.
- Oil returns pressurized to the PCM where is filtered.
- From the PCM, oil flows to the utilities and/or main and tail rotor servo actuators.
- For circuit 2 only, oil flow to the tail rotor servo actuator is first delivered to the Tail Rotor Shut-Off Valve.
- From the main and tail rotor servo actuators and/or utilities, the oil is then returned to the PCM where it is filtered again.
- When no demand is made on the system (no pressure drop because flight controls are stationary or utilities are not operating), the pump fluid flow to the system is zero and the oil returns to the PCM through the case drain line to the return line.
- In the event of oil fast leakage in the circuit 2, oil level microswitches in the PCM reservoir operate the Tail Rotor Shut-Off Valve to isolate the oil delivery to the tail rotor servo actuator to give priority to the Main Rotor Servo actuators.
- Another shut-off valve installed in the PCM automatically isolates the utilities system in the event of excessive leakage to give priority to main and tail rotor servo actuators.



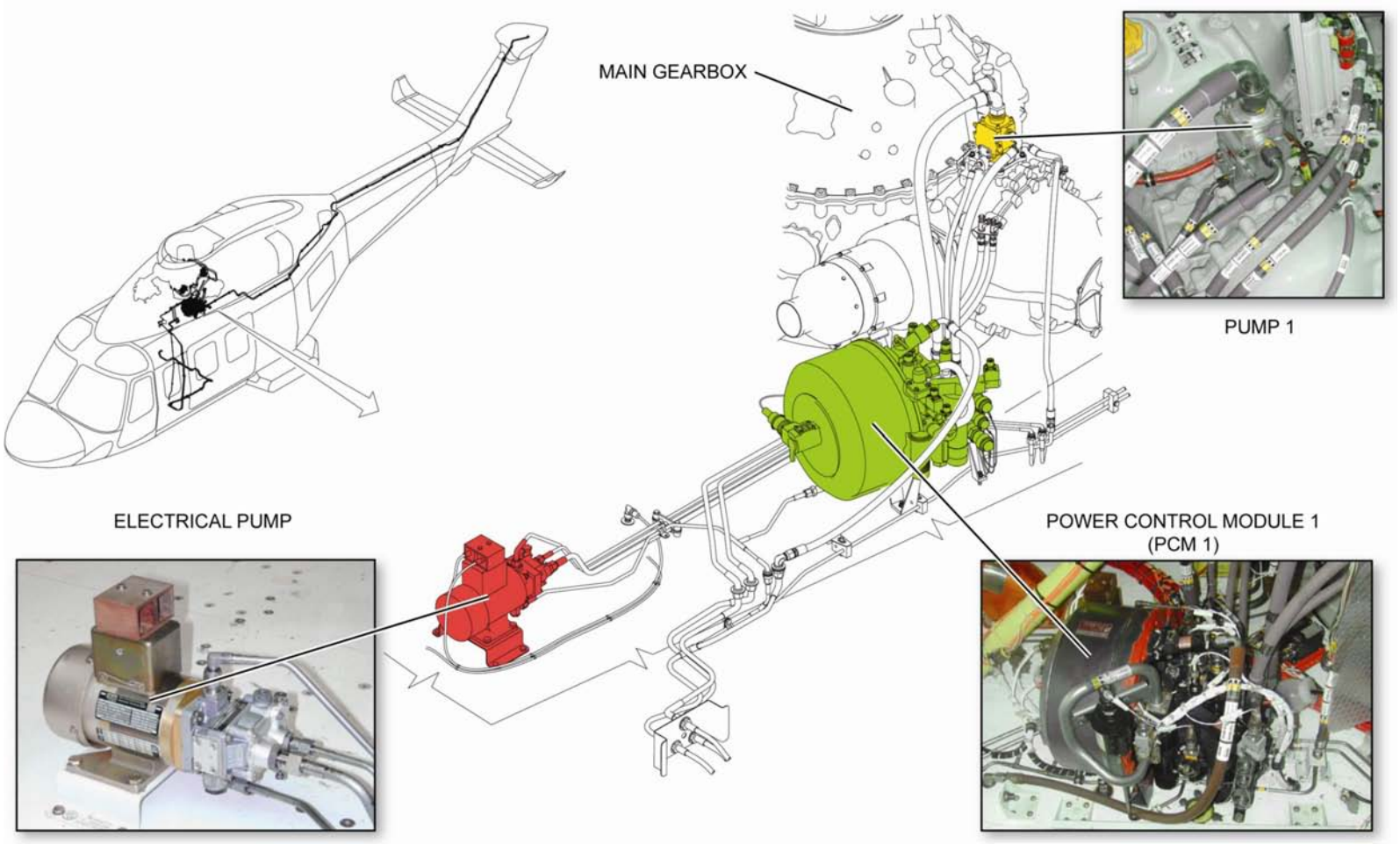
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MAIN HYDRAULIC POWER SYSTEM SCHEMATIC



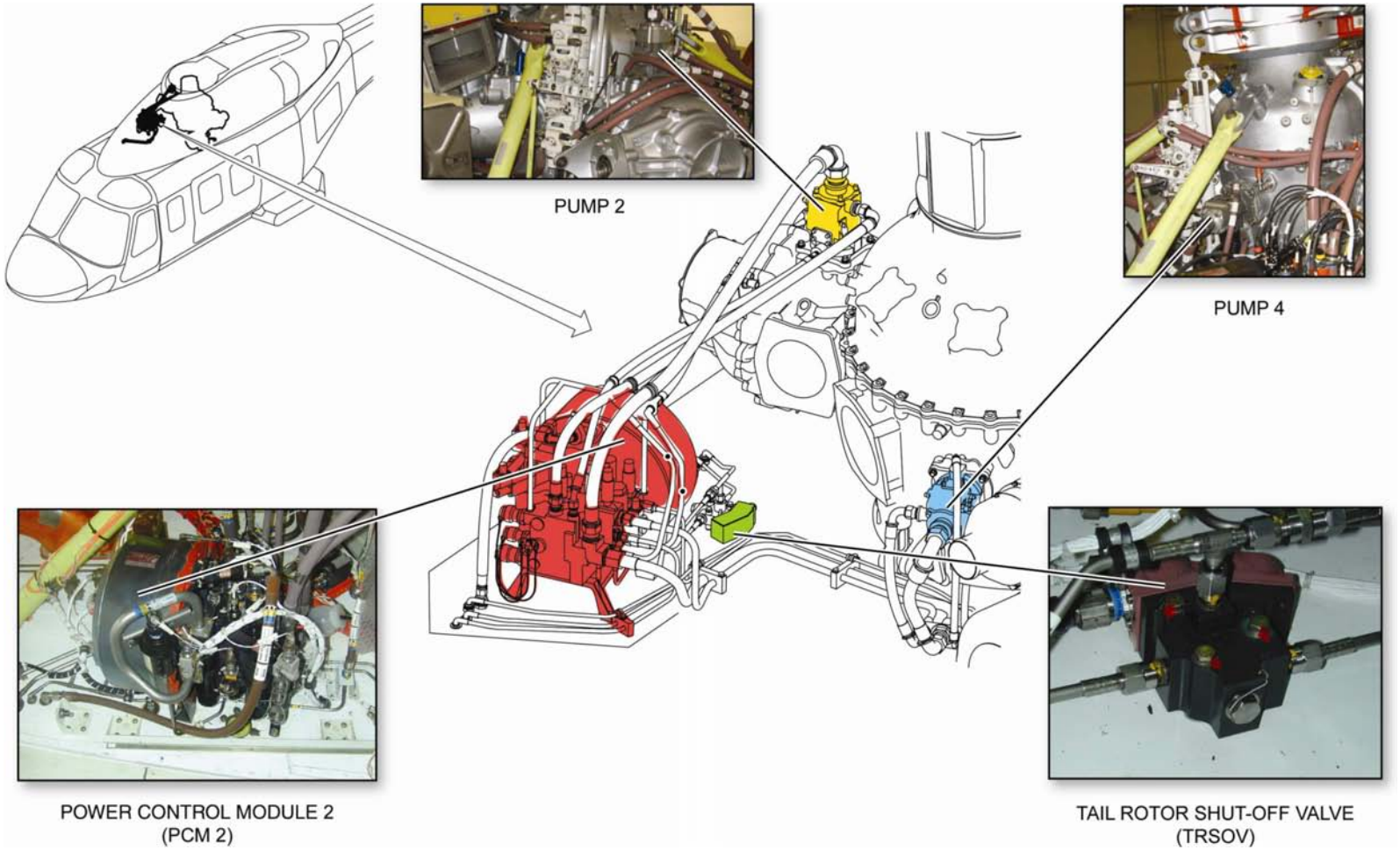
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LOCATION OF MAJOR COMPONENTS



ICN-39-A-291000-G-A0126-00707-A-01-1

HYDRAULIC SYSTEM NO.1 - MAIN COMPONENTS



ICN-39-A-291000-G-A0126-00709-A-01-1

HYDRAULIC SYSTEM NO.2 - MAIN COMPONENTS

CONTROLS AND INDICATORS – HYDRAULIC SYSTEM CONTROL PANEL

1 ELEC PUMP pushbutton

- ON (green) lighted ..... the electrical pump is operating for a maximum time of two minutes (controlled by the time relay)

2 HYD 1 indicator

- PRESS (amber) lighted .... the hydraulic pressure in the system no.1 drops below 163 bar lights on when the SOV switch is set to 1 CLOSE
- TEMP (amber) lighted ..... the temperature of the fluid in the system no.1 is above 134°C

3 SOV guarded switch

- central position ..... the shut-off valves are open and automatically controlled by the system
- CLOSE ..... ON THE GROUND
  allows to select a single system (the no.1 or the no.2) during the ground check
  IN FLIGHT
  allows to isolate a system (the no. 1 or the no.2) in case of anomalous conditions

4 HYD 2 indicator

- PRESS (amber) lighted .... the hydraulic pressure in the system no.2 drops below 163 bar lights on when the SOV switch is set to 2 CLOSE
- TEMP (amber) lighted ..... the temperature of the fluid in the system no.2 is above 134°C



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HYDRAULIC SYSTEM CONTROL PANEL

CONTROLS AND INDICATORS – SYNOPTIC PAGE ON MFD

The synoptic page provides a diagram showing the interface among the PCM pumps, landing gears, servo actuators and TRSOV as follow:

1 Circuit 1 or 2 operating pressure read-out.

2 Circuit 1 or 2 operating temperature read-out.

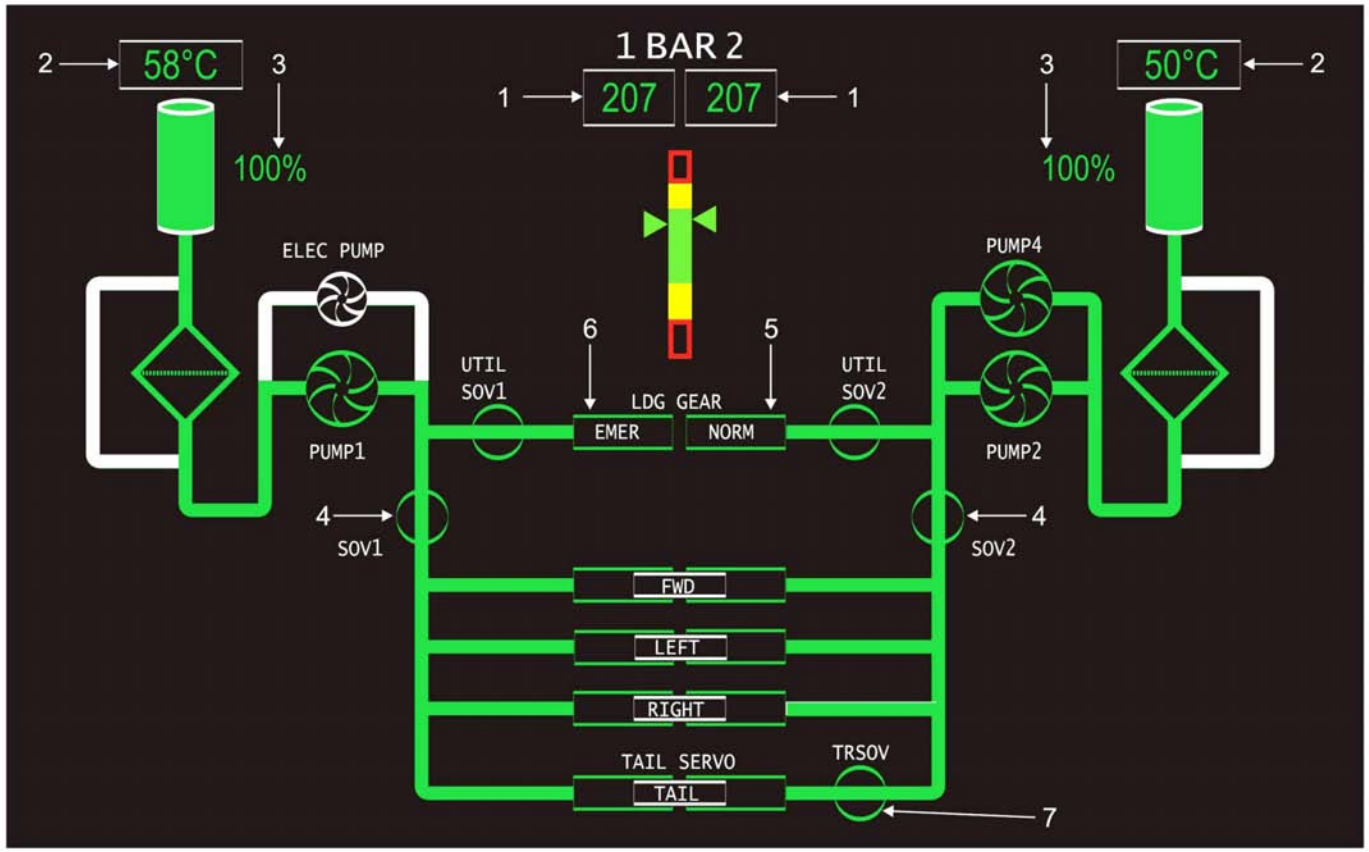
3 Circuit 1 or 2 oil level percentage indicator: ..... indicates the oil level percentage in the PCM 1 (2) reservoir.

4 SOV 1 (2) indicator: ..... indicates the condition of the PCM 1 (2) F/C shut-off valve operated by the 1 (2) CLOSE switch in the Hydraulic Control Panel.
When the valve is open the SOV 1 (2) indicator is vertical. When the valve is closed the SOV 1 (2) indicator is horizontal.
In this condition are also provided the cautions 1 (2) HYD OIL PRESS, 1 (2) SERVO to the MFD and the amber legend PRESS on the HYD 1 (2) indicator in the Hydraulic Control Panel is illuminated.

5 LDG GEAR NORM indicator: ..... indicates the condition of the PCM2 utility shut-off valve operated by the PCM2 oil LOW LEVEL 1, LOW LEVEL 2 and MIN LEVEL microswitches. When the valve is open the LDG GEAR NORM indicator is horizontal.
When the valve is closed the LDG GEAR NORM indicator is vertical. In this condition is also provided the caution HYD UTIL PRESS only or the cautions HYD UTIL PRESS and 2 HYD MIN.

6 LDG GEAR EMER indicator: ..... indicates the condition of the PCM1 utility shut-off valve operated by the PCM1 oil MIN LEVEL microswitch.
When the valve is open the LDG GEAR EMER indicator is horizontal. When the valve is closed the LDG GEAR EMER indicator is vertical. In this condition are also provided the cautions EMERG LDG PRESS and 1 HYD MIN.

7 TAIL SERVO TRSOV indicator: ..... indicates the condition of the Tail Rotor Shut-Off Valve operated by the PCM2 oil LOW LEVEL 2 microswitch.
When the valve is open the TAIL SERVO TRSOV indicator is vertical.
In this condition is also provided the caution 2 SERVO.



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HYDRAULIC SYNOPTIC PAGE

SYMBOL USED IN THE SYNOPTIC

The following table shows the symbols used to represent the major components of the hydraulic system on the hydraulic synoptic page and the relevant states.

PUMP STATE	PUMP GRAPHIC
OFF	
FAILED OFF	
ON	
FAILED ON	
UNDETERMINED	

LEVEL STATE OF THE RESERVOIRE	LEVEL STATE
100%	
50%	
28%	
22%	
UNDETERMINED	

ITEM	NORM	FAILED
LANDING GEAR ACTUATOR GRAPHIC	LDG GEAR EMER NORM	LDG GEAR EMER NORM

ITEM	NORM	FAILED
FLIGHT CONTROL ACTUATOR GRAPHIC	MAIN SERVO FWD LEFT RIGHT TAIL SERVO TAIL	MAIN SERVO FWD LEFT RIGHT TAIL SERVO TAIL

FILTER STATE	FILTER GRAPHIC
NORMAL	
CLOGGED	
UNDETERMINED	

FLOW LINE STATE	FLOW LINE GRAPHIC
NO FLOW	
FLOW	
UNDETERMINED	

VALVE STATE	VALVE GRAPHIC	VALVE STATE	VALVE GRAPHIC
CLOSED		FAIL OPEN	
OPEN		UNDETERMINED	

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SYMBOLS USED IN THE SYNOPTIC DIAGRAM

CAPTIONS

When the amber legend PRESS of the HYD1 indicator is lit on the Hydraulic System Control Panel, other connected cautions are displayed on the CAS message window as follows:

1 1 HYD OIL PRESS ..... when the hydraulic system 1 pressure drops below 163 bar. It extinguishes when above 190 bar.

2 1 SERVO ..... when at least one servo actuator in system 1 is in by-pass.

3 1 HYD PUMP ..... when the 1 main pump outlet pressure drops below 163 bar. It extinguishes when above 190 bar.

4 EMER LDG PRESS ..... when the pressure in the Emergency Landing Gear extension line drops below 163 bar. It extinguishes when above 190 bar.

When the amber legend PRESS of the HYD2 indicator is lit on the Hydraulic system Control Panel, other connected cautions are displayed on the CAS message window as follows:

5 2 HYD OIL PRESS ..... when the hydraulic system 2 pressure drops below 163 bar. It extinguishes when above 190 bar.

6 2 SERVO ..... when at least one servo actuators in system 2 in by-pass.

7 2 – 4 HYD PUMP ..... when the 2 and 4 main pumps outlet pressure drops below 163 bar. It extinguishes when above 190 bar.

8 HYD UTIL PRESS ..... when the pressure in the utility system (Normal Landing Gear extension line) drops below 163 bar. It extinguishes when above 190 bar.

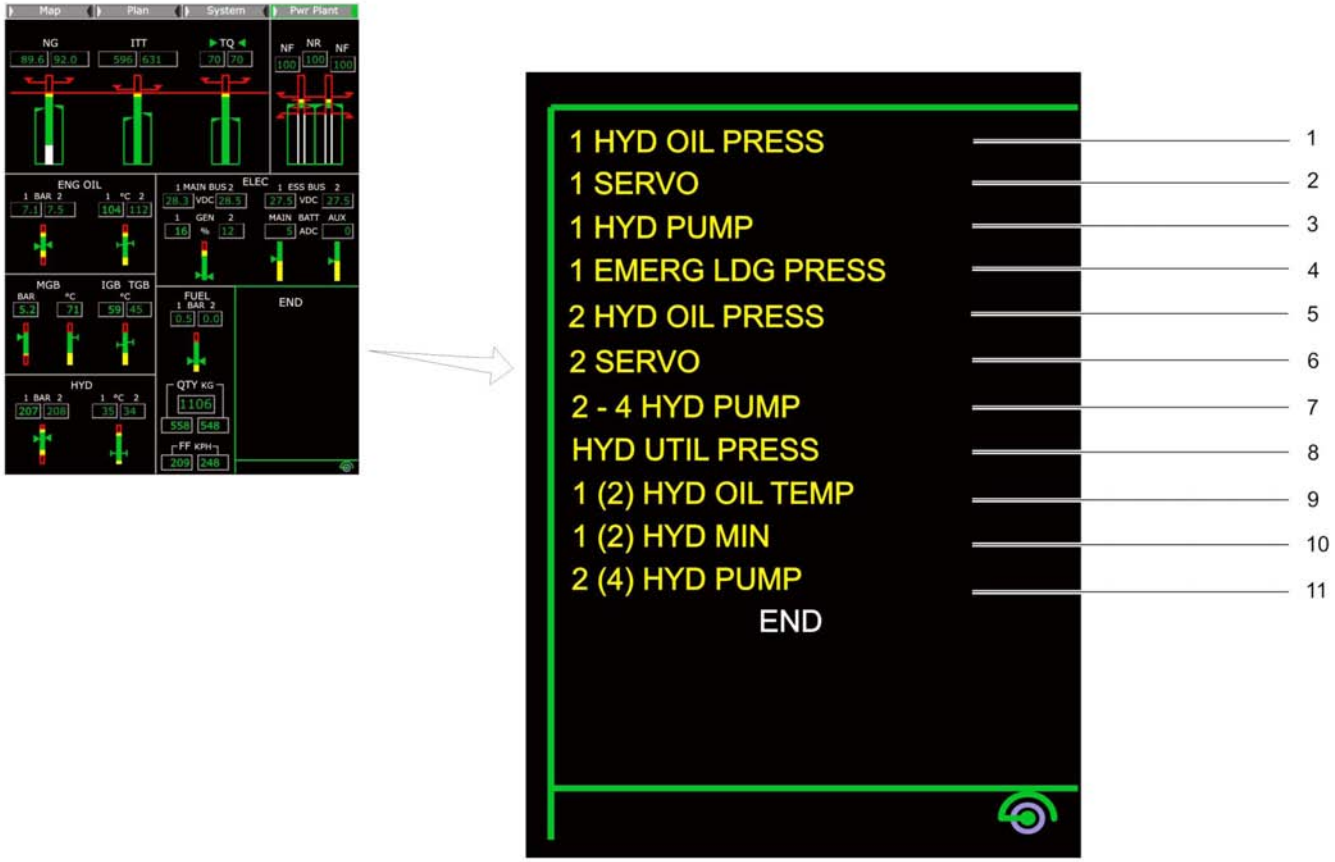
When the amber legend TEMP of the HYD 1 (2) indicator is lit on the Hydraulic System Control Panel, the CAS message window displays as follows:

9 1 (2) HYD OIL TEMP ..... when the hydraulic system 1 oil temperature is above 134°C. It extinguishes when below 127°C.

OTHER CAUTION

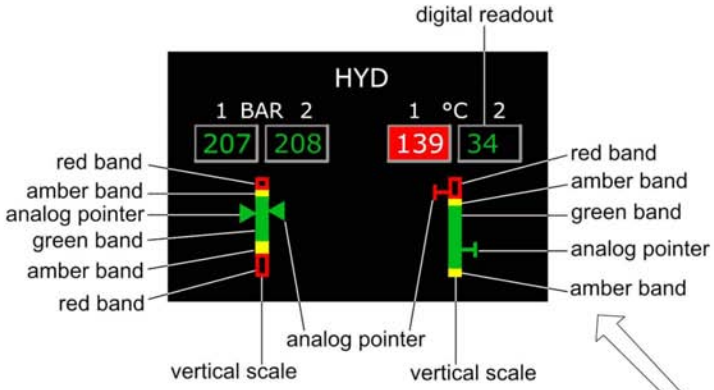
10 1 (2) HYD MIN ..... when fluid level is at minimum in the PCM1 (2) reservoir (0.7 l = 22%).

11 2(4) HYD PUMP ..... when the 2 (4) main pump outlet pressure drops below 163 bar. It extinguishes when above 190 bar.



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MFD - CAPTION

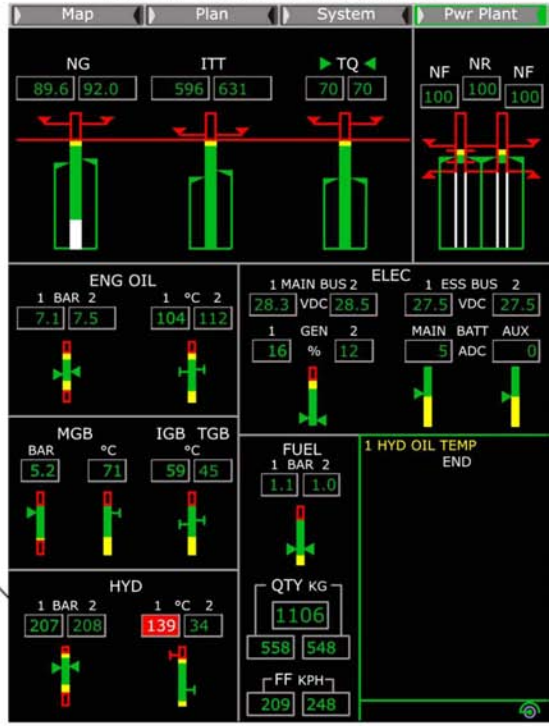


Two vertical scales (PRESS/TEMP), one analog pointer and one digital readout per system

Range:
- for analog scale 0 thru 310 bar
- for analog scale -50 thru 150°C
- for digital readout 0 thru 310 bar
- for digital scale -50 thru 150°C

Resolution:
- for digital readout 1 bar
- for digital readout 0 thru 310 bar

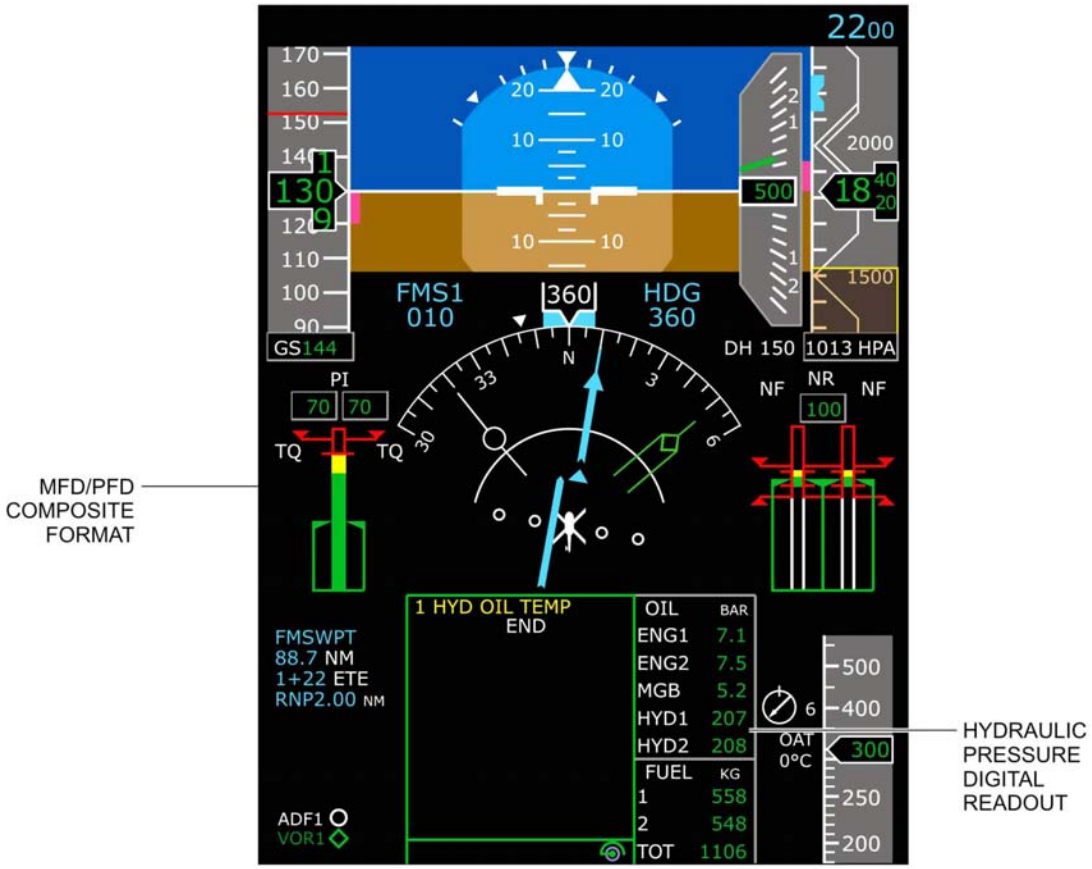
MULTIFUNCTIONAL DISPLAY



NOTE:
When the elctric pump is operative, the digital readout pressure and the relative analog pointer are displayed on the MFD in place of the hydraulic system N.1.

ICN-39-A-291000-G-A0126-00733-A-01-1

MFD - PRESENTATION



ICN-39-A-291000-G-A0126-00734-A-01-1

MFD - COMPOSITE PRESENTATION

SECTION 11 - POWER CONTROL MODULE

POWER CONTROL MODULE

Description

The two Power Control Modules located on the left (PCM 1) and right (PCM 2) side of the upper deck structure in front of the Main Gearbox consist each of the following main items (Refer to the following Figure).

Pressurized Reservoir (1)

A reservoir inside which the oil is separated from the air by using a rubber membrane, energized by a spring which provides pressurization to the oil. In the air side a vent port is provided to allow the reservoir casing to breathe and to drain any water trapped or the hydraulic oil in case of failure of the rubber membrane.

Oil level sensor (2)

An oil level sensor used in case of circuit fluid fast leakage automatically provides a remote indication of reservoir fluid level through three microswitches:

- The LOW 1 LEVEL microswitch (50%) used in PCM2 as a safety threshold closes the utility shut-off valve (utility circuit isolation).
  The caution HYD UTIL PRESS and the hydraulic synoptic page will display on the MFD this condition (valve closed and reservoir oil quantity).
- The LOW 2 LEVEL microswitch (28%) used in PCM2 as a safety threshold closes the Tail Rotor Shut-off valve and re-opens the utility shut-off valve.
  The caution 2 SERVO is provided while the caution HYD UTIL PRESS clears. The hydraulic synoptic page will also display this condition (valve closed and reservoir oil quantity). The microswitch is also used to prevent to set OFF System 1.
- The MIN LEVEL microswitch (22%) used in PCM 2 recloses the utility shut-off valve while in the PCM 1 it closes the utility shut off valve.
  At this stage the quantity of fluid remaining into the reservoirs is sufficient to cover the main rotor actuators volumetric requirements.
  The following cautions 1 (2) HYD MIN, EMER LDG PRESS, (HYD UTIL PRESS) are also displayed on the CAS. The hydraulic synoptic page will also display these conditions (valve closed and reservoir oil quantity).

The microswitches can be manually operated for electrical continuity check through a test lever on the back side of the visual oil level indicator.

Visual oil level indicator (3)

A visual oil level indicator is located on the back side of reservoir. During maintenance operation is possible to check the FULL and ADD oil levels through a sightglass.

Pressure relief / bleeding valve (4)

In the oil side a pressure relief/bleeding valve, located on the top side of the reservoir, is provided to assure that the pressure within the reservoir is not exceeding a safe limit and to provide discharging of the reservoir hydraulic fluid.
The bleed valve is manually open to discharge the fluid by the rotation of a lever positioned on the top side of the valve.

Temperature oil sensor (5)

It is installed on the reservoir suction line.
The temperature sensor provides the continuous read-out to the MFD.

Pressure and return filter assembly (6a – 6b)

Pressure and return filter assembly to provide filtration of the hydraulic fluid.
Return filter only is provided with a by-pass valve to allow oil recirculation in the event the filter is completely clogged. Filter clogged condition is shown on the hydraulic synoptic page.

Filter clogged indicators (7a – 7b)

Two filter clogged indicators for each filter.
The mechanical indicator (red pop-out) once actuated remains extended until reset manually (position hidden from view).
The maintenance messages 1 (2) HYD FILTER are provided to the CAS.

Return filter bypassvalve (8)

Only the return filter have bypass valve.
In case of overpressure on the circuit return line (due to a filter clogged) allows to by-pass the hydraulic fluid to the PCM reservoir.

Flight controls circuit shut-off valve (9a)

Flight controls circuit shut-off valve Flight controls circuit shut-off valve, solenoid operated, open when de-energized.
The valve is used to isolate the flight controls hydraulic circuit and connects the pressure lines to the reservoir (when manually energized with SOV 1 (2) CLOSE switch on the Hydraulic System Control Panel.
When the shut-off valve is set closed, the cautions 1(2) HYD PRESS, 1(2) SERVO are displayed in the CAS and the SOV 1 (SOV 2) indicator condition are provided to the MFD; the amber legend PRESS on the HYD 1 (2) indicator on the Hydraulic System Control Panel is illuminated.

Utility circuit shut-off valve (9b)

Utility circuit shut-off valve, solenoid operated, open when de-energized.
The valve is used to isolate the utilities circuits and connects the pressure lines to the reservoir (when automatically energized by the oil level microswitches. There is no manual control of this valve).
When the shut-off valve is automatically closed, the cautions HYD UTIL PRESS (EMER LDG PRESS) and the LDG GEAR NORM (LDG GEAR EMER) indicator condition are provided to the CAS.

Pressure system relief valve (10)

A pressure system relief valve used to protect the circuit by limiting the pressure in case of failure of the pump compensator.

Pressure sensor (11)

Pressure sensor installed before the shut-off valves provides the pressure continuous read-out to the MFD.

Pressure switches outlet (12a – 12b)

Two pressure switches in PCM1 and PCM2 installed on each pressure circuit outlet (flight controls and utilities).
The F/C outlet pressure switch provides the caution 1 (2) HYD PRESS (together with the caution 1 (2) SERVO) to the MFD, illuminates the amber legend PRESS on the HYD 1 (2) indicator on the Hydraulic System Control Panel.
Is also used as an interlock to prevent to set OFF the other hydraulic circuit.

Pressure switches (12c - 12b)

Two pressure switches in PCM 2 and one in PCM 1 are installed on each pump pressure inlet. In PCM 1 the pressure switch indication for the Electric Pump is installed but not connected.

- The inlet pressure switches in PCM 2 provide the caution2 (4) HYD PUMP to the CAS. In case the 2-4 HYPUMP caution is provided, the cautions 2 HYPRESS, HYD UTIL PRESS and 2 SERVO are shown together with the amber legend PRESS on the HYD 2 indicator on the Hydraulic System Control Panel.
- The inlet pressure switch in PCM 1 provides the caution 1 HYD PUMP together with the cautions 1 HYD PRESS, EMERG LDG PRESS, 1 SERVO to the MFD and illuminates the amber legend PRESS on the HYD 1 indicator on the Hydraulic System Control Panel.

Flow restrictor (13)

A flow restrictor used to discharge on the return line the pressure trapped in the ground equipment pressure lines after maintenance operation.

Return shuttle-valve (14)

A return shuttle-valve used to connect the helicopter hydraulic circuit to the ground equipment reservoir, by passing the PCM reservoir.
The shuttle valve commutation from normal position (return circuit line to PCM reservoir) to maintenance position (return circuit line to ground equipment reservoir) is operated by the ground equipment pressure supply.

Return by-pass valve (15)

A return by-pass valve used as a safety device in the event of a return shuttle valve seizure in maintenance position.

This is a relief valve which in case of overpressure on the circuit return line (due to a shuttle valve seizure) allows to by-pass the hydraulic fluid to the PCM reservoir.

Check valves (16a, 16b, 17a, 17b, 18, 19a, 19b, 20a, 20b)

Nine check valves used to control hydraulic flow.

Case drain differential pressure indicators (21a – 21b)

Two case drain differential pressure indicators used to monitor the flow from the pump to the PCM return line.
When the flow through a pump case drain increases over a limit, it indicates an impending pump failure.
In this case a red mechanical indicator is actuated by the overpressure and remain extended until reset manually (position hidden from view).

Temperature oil sensor and switch (22)

They are both installed on the reservoir suction line.
The temperature sensor provides the continuous read-out to the MFD.
The temperature switch provides the caution 1 (2) HYD TEMP to the MFD and illuminates the amber legend TEMP on the HYD 1 (2) indicator on the Hydraulic System Control Panel.

Overview

- Nine check valves used to control hydraulic flow.
- Pressure and return ground equipment connection ports are located on the right side of the PCM.
- Two pumps suction, pressure and drain lines ports.
- Two flight controls pressure and return circuit lines connection ports.

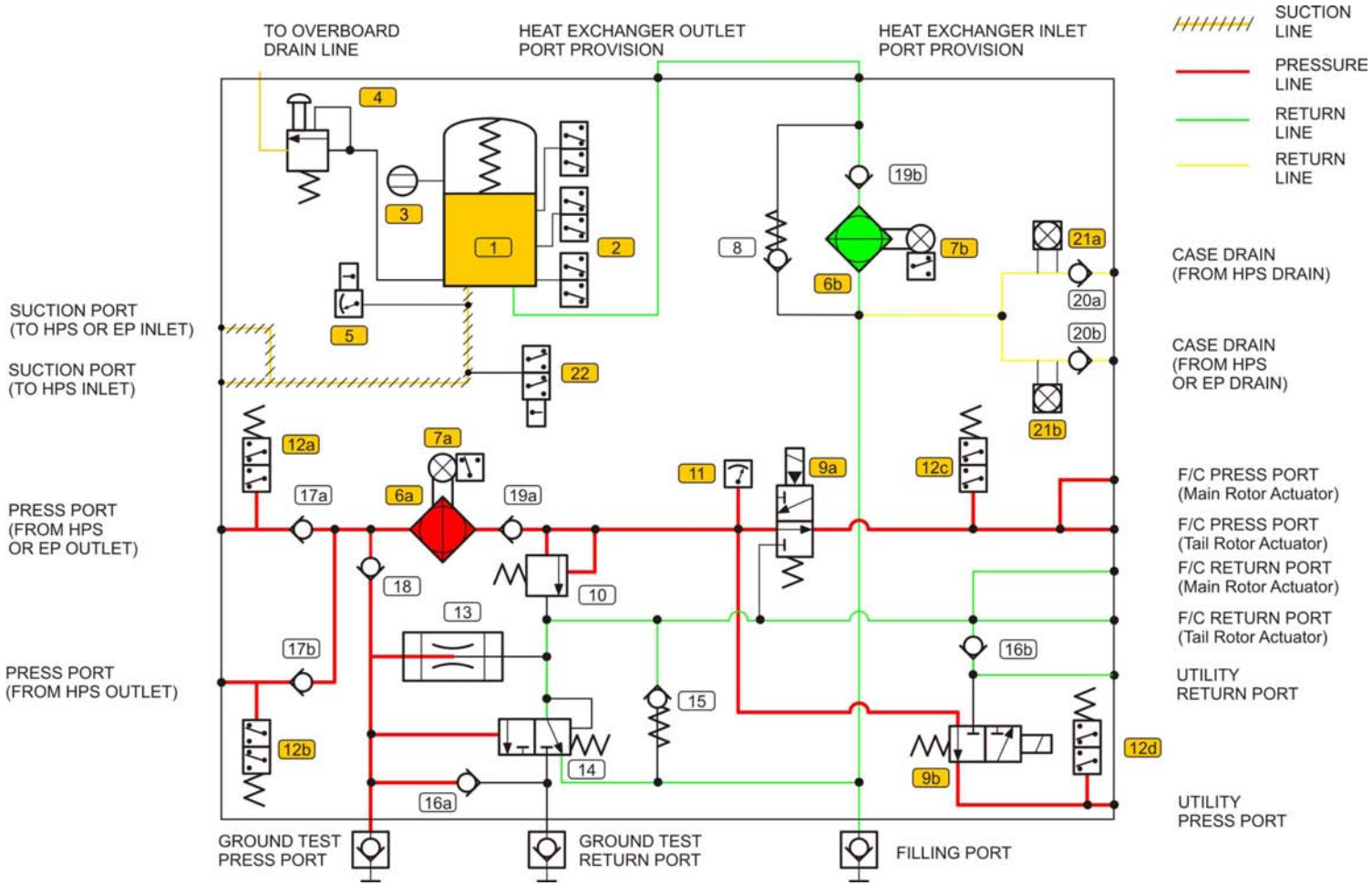
- A utility pressure and return circuit lines connection ports.
- A PCM filling connection port
- A drain overboard line connection port

POWER CONTROL MODULE SCHEMATIC (SHEET 1 OF 2)

(1)	Pressurized reservior
(2)	Oil level microswitches
(3)	Visual oil level indicator (sight glass)
(4)	Pressure relief/bleeding valve
(5)	Temperature oil sensor
(6a)	Pressure filter
(6b)	Return filter
(7a, 7b)	Filters clogged indicators
(8)	Return filter bypass value
(9a)	Flight controls circuit shut-off valve (shown de-energized, System ON)
(9b)	Utility circuit shut-off valve
(10)	Pressure system relief valve
(11)	Pressure sensor
(12a, 12b, 12c, 12d)	Pressure switches
(13)	Flow restrictor
(14)	Return shuttle valve

POWER CONTROL MODULE SCHEMATIC (SHEET 1 OF 2)

(15)	Return bypass valve
(16a, 16b, 17a, 17b, 18, 19a, 19b, 20a, 20b)	Check valves
(21a, 21b)	Case drain differential pressure indicator
(22)	Temperature switch



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POWER CONTROL MODULE SCHEMATIC (SHEET 2 OF 2)

Pre-Flight Check Operations Flight Controls

The system no.1 permits to test the flight controls on ground only.

The pilot pressing the ELEC PUMP push-button on the HYD control panel can carry out cyclic, collective and yaw pedals full and free check.

The full and free check should be carried out with slow displacement of the controls and one control at a time in order not to overload the electric pump.

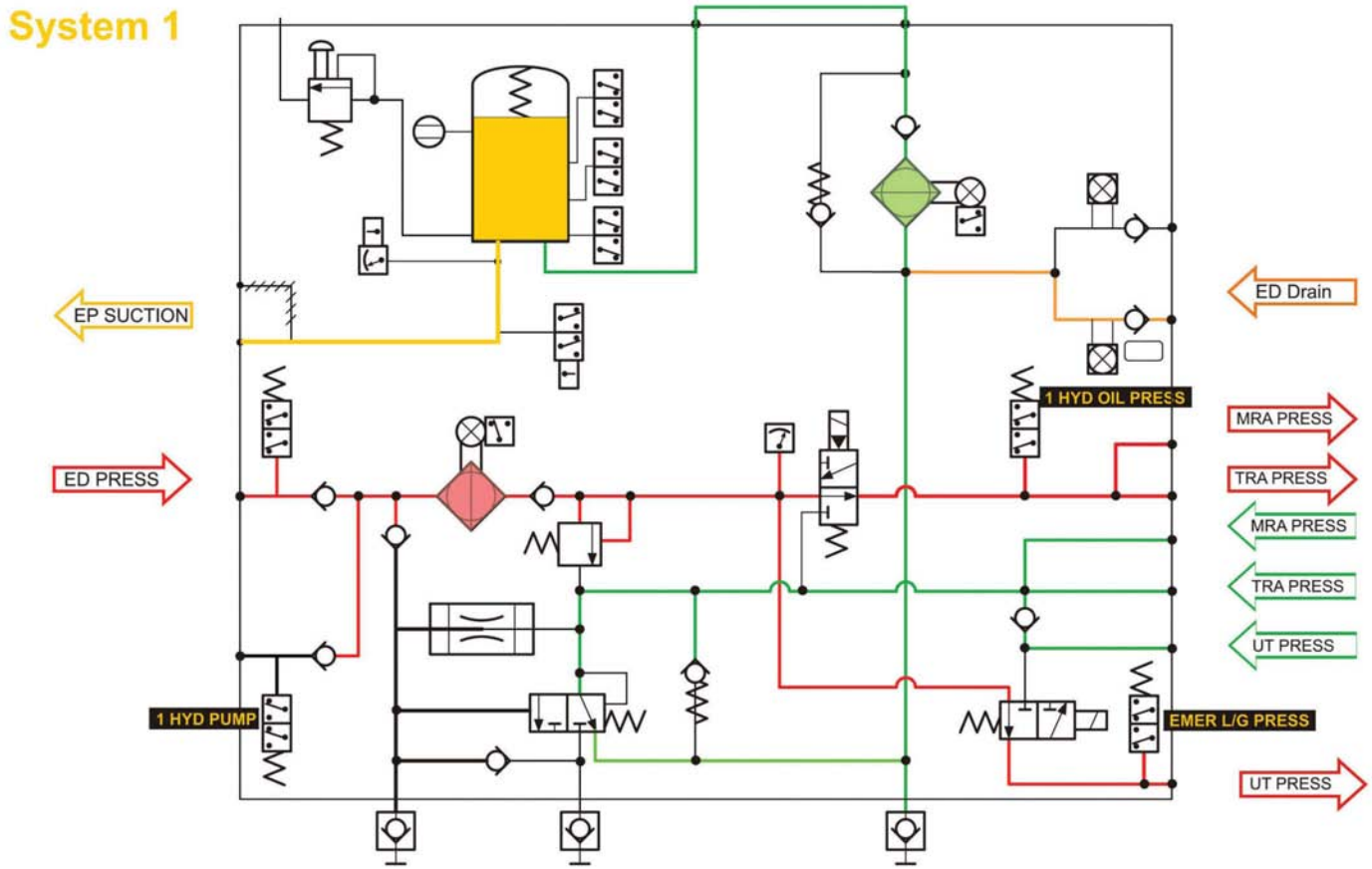
The electrical pump disengages automatically after 2 minutes.

HYDRAULIC synoptic page

- ELEC PUMP pressurises the hydraulic system no.1 at 105 bars. This value of pressure implies a fail condition for the main rotor, tail rotor and emergency landing gear servo-actuators
- SOV1 and SOV2 are displayed in an undetermined status
- cautions are displayed in the CAS window
  - 1-2-4 HYD PUMP
  - 1-2 SERVO
  - 1-2 HYD OIL PRESS
  - HYD UTIL PRESS
  - EMER LDG PRESS

on HYD control panel

- ELEC PUMP: ON lighted (green)
- HYD1 and HYD2: PRESS lighted (amber)



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PRE FLIGHT CHECK OPERATION

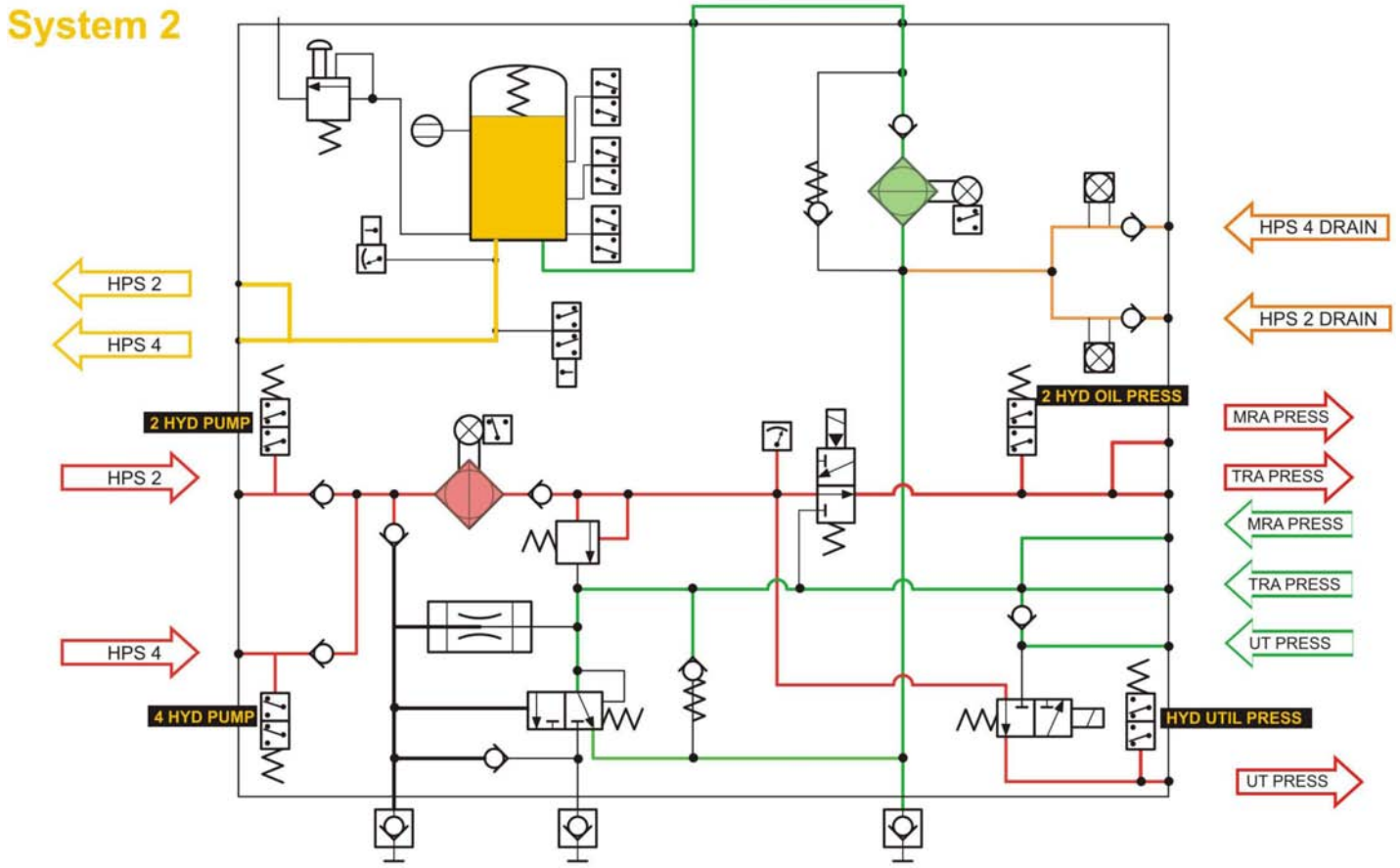
NORMAL OPERATIONS

hydraulic synoptic page

- PUMP1 pressurizes the system no.1
- PUMP2 and PUMP4 pressurize the system no.2
- UTIL SOV1 and UTIL SOV2 are opened and the landing gear is pressurized at 207 bar
- SOV1, SOV2 and TRSOV are opened and flight controls are pressurized at 207 bar by both hydraulic systems

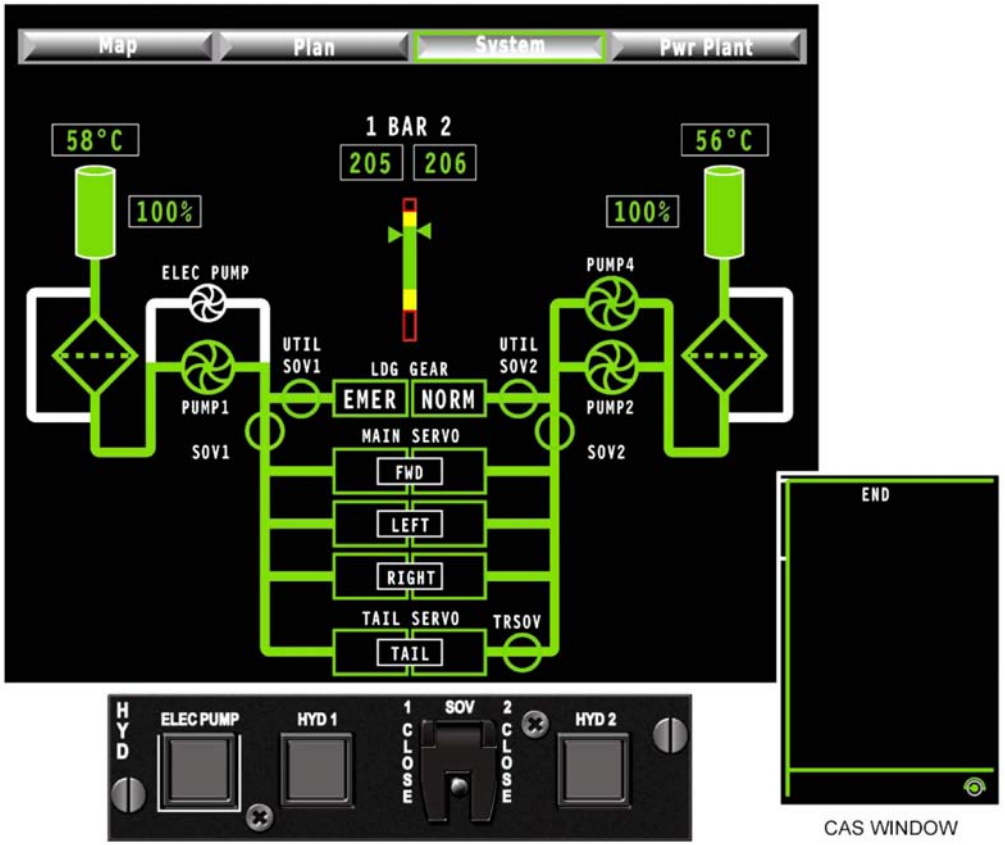
on hyd control panel

- HYD1 and HYD2: blank



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NORMAL OPERATION MODE



ICN-39-A-291100-G-A0126-03301-A-01-1

MFD SYNOPTIC PAGE – NORMAL OPERATION

FAILURE – PUMP NO.2

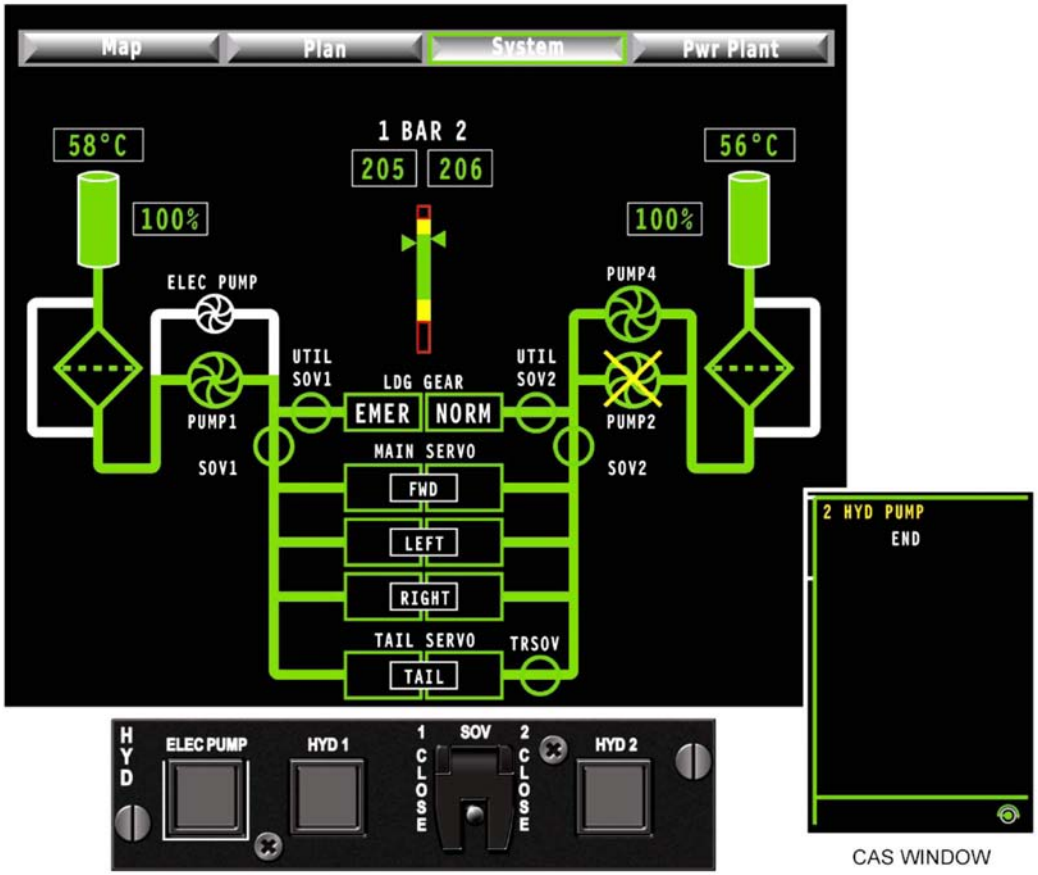
In case of failure of one pump (for example PUMP2), the remaining pump (in this example PUMP4) is able to supply the operating pressure to the system no.2.

hydraulic synoptic page

- PUMP1 pressurizes the system no.1
- PUMP2 fails; PUMP4 pressurize the system no.2
- UTIL SOV1 and UTIL SOV2 are opened and the landing gear is pressurized at 207 bar
- SOV1, SOV2 and TRSOV are opened and flight controls are pressurized at 207 bar by both hydraulic systems
- a caution is displayed in the CAS window and the pilot has to follows the relevant malfunction procedure
- 2 HYD PUMP

on hyd control panel

- HYD1 and HYD2: blank



ICN-39-A-291100-G-A0126-03302-A-01-1

MFD SYNOPTIC PAGE – FAILURE: PUMP NO.2

FAILURE – PUMP NO.1

PUMP1 failure implies the total loss of system no.1 operations. In this case hydraulic power is supplied by the system no.2 only.

hydraulic synoptic page

- PUMP1 fails
- PUMP2 and PUMP4 pressurize the system no.2
- SOV1 and UTIL SOV1 are in an undetermined status. The flight controls are considered failed for the system no.1. The landing gear emergency operation is not available
- UTIL SOV2 is opened and the landing gear normal operations is pressurized at 207 bar
- SOV2 and TRSOV are opened and flight controls are pressurized at 207 bar only by the system no.2
- cautions are displayed in the CAS window and the pilot has to follows the relevant malfunction procedure

  1 HYD PUMP

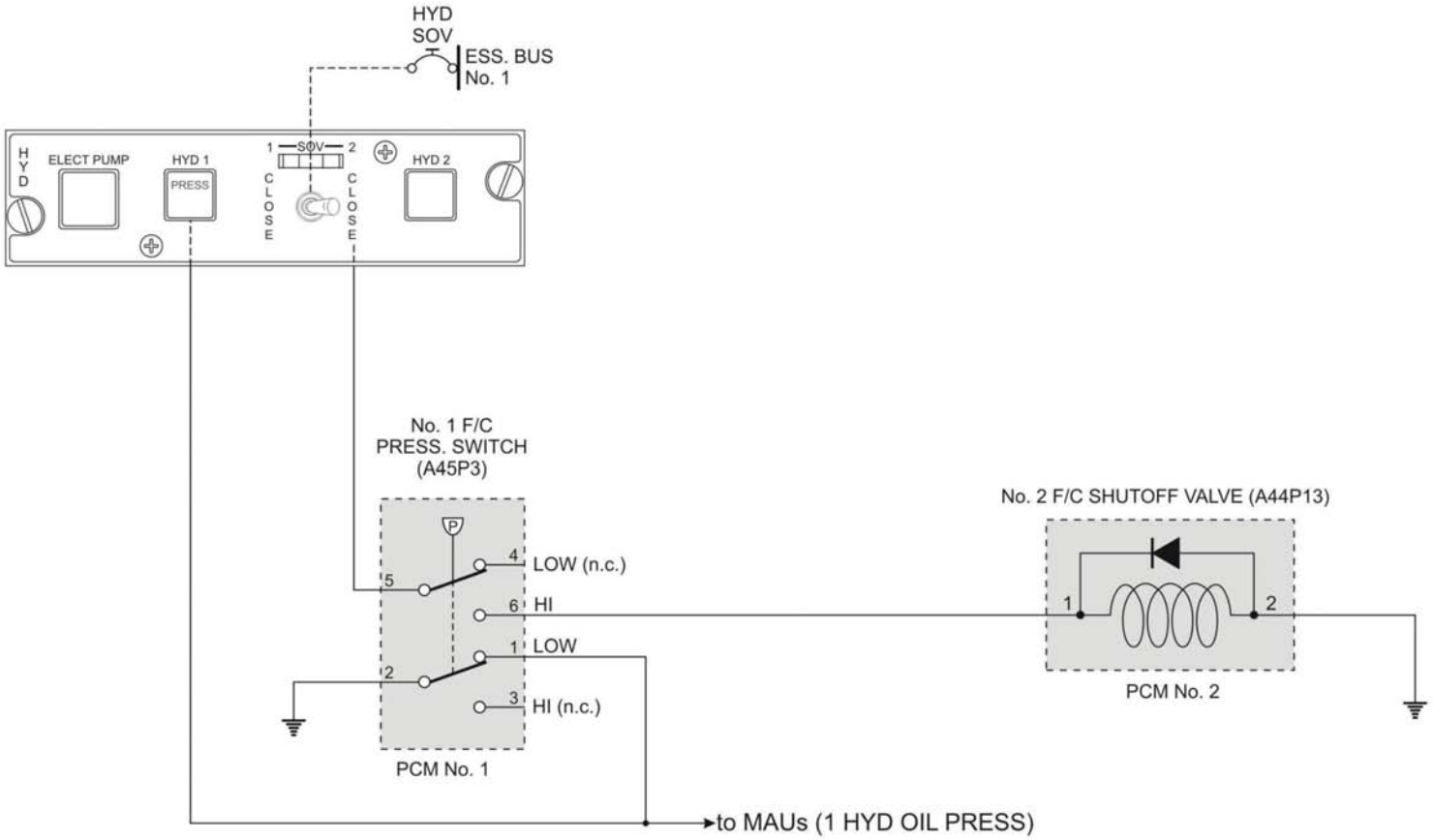
  1 HYD OIL PRESS

  1 SERVO

  EMER LDG PRESS

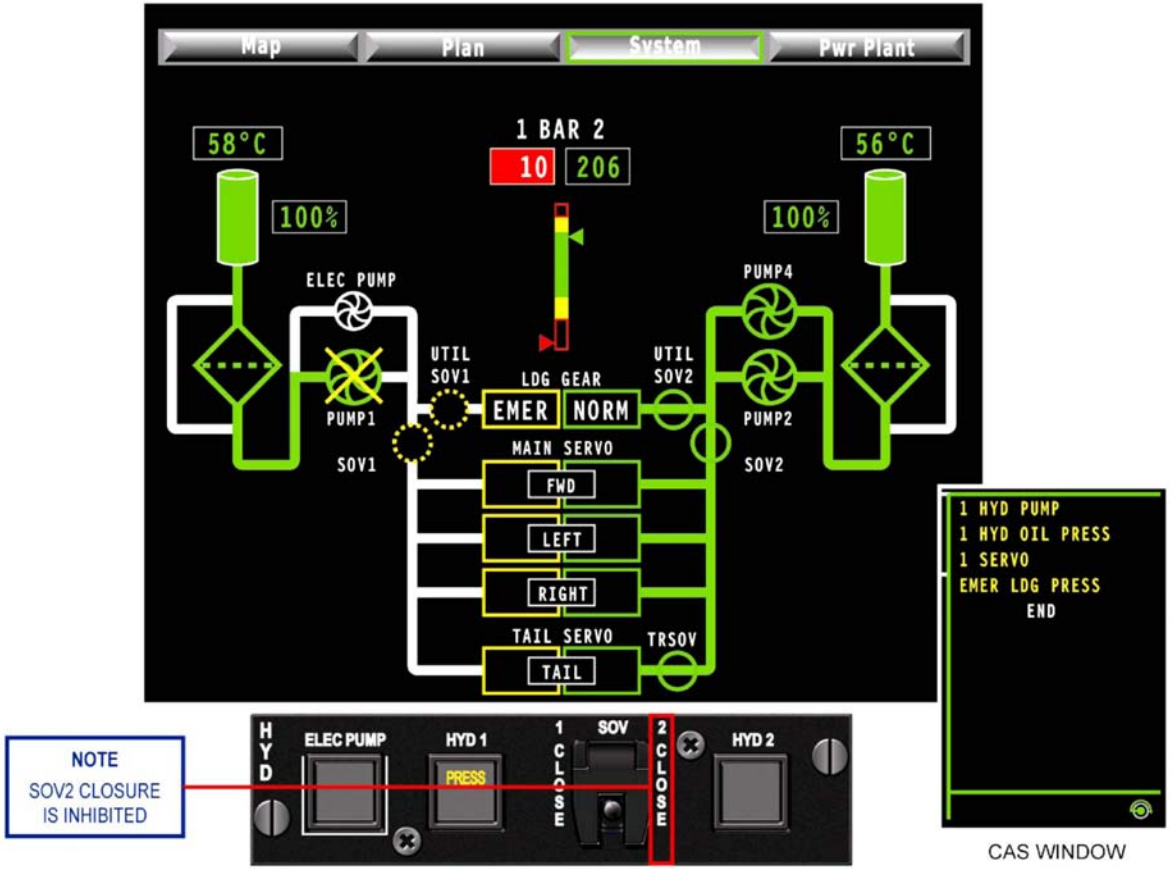
on hyd control panel

- HYD 1: PRESS lighted (amber)
- HYD 2: blank



ICN-39-A-291100-G-A0126-00824-A-01-1

MFD SYNOPTIC PAGE – FAILURE: PUMP NO.2



ICN-39-A-291100-G-A0126-03303-A-01-1

MFD SYNOPTIC PAGE – FAILURE: PUMP NO.1

FAILURE – HYD 2 FLUID LEVEL AT 50%

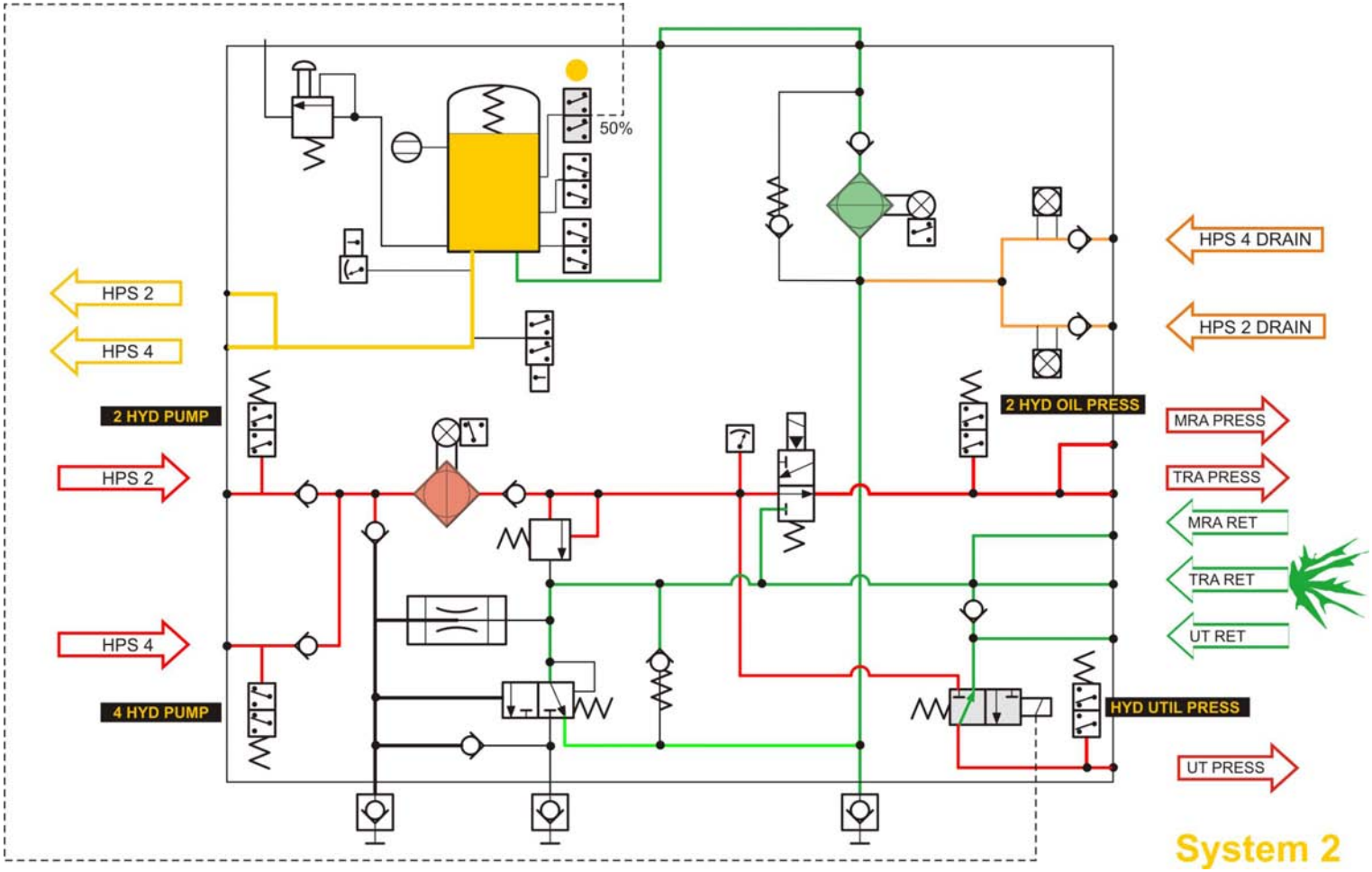
Hereafter are described some failure relevant different conditions of leakage in the hydraulic system no.2. In this case, the three level microswitches installed inside the reservoir, control the UTIL SOV and the TRSOV to contain the leakage.

hydraulic synoptic page

- PUMP1 pressurizes the system no.1
- PUMP2 and PUMP4 pressurize the system no.2
- UTIL SOV1 is opened and the landing gear EMER circuit is pressurized at 207 bar
- UTIL SOV2 is automatically closed by the level microswitch and the landing gear normal operation is not available. The landing gear free falls due to lack of pressure
- SOV1, SOV2 and TRSOV are opened and the flight controls are pressurized at 207 bar by the system no.1 and no.2
- a caution is displayed in the CAS window and the pilot has to follows the relevant malfunction procedure
- HYD UTIL PRESS

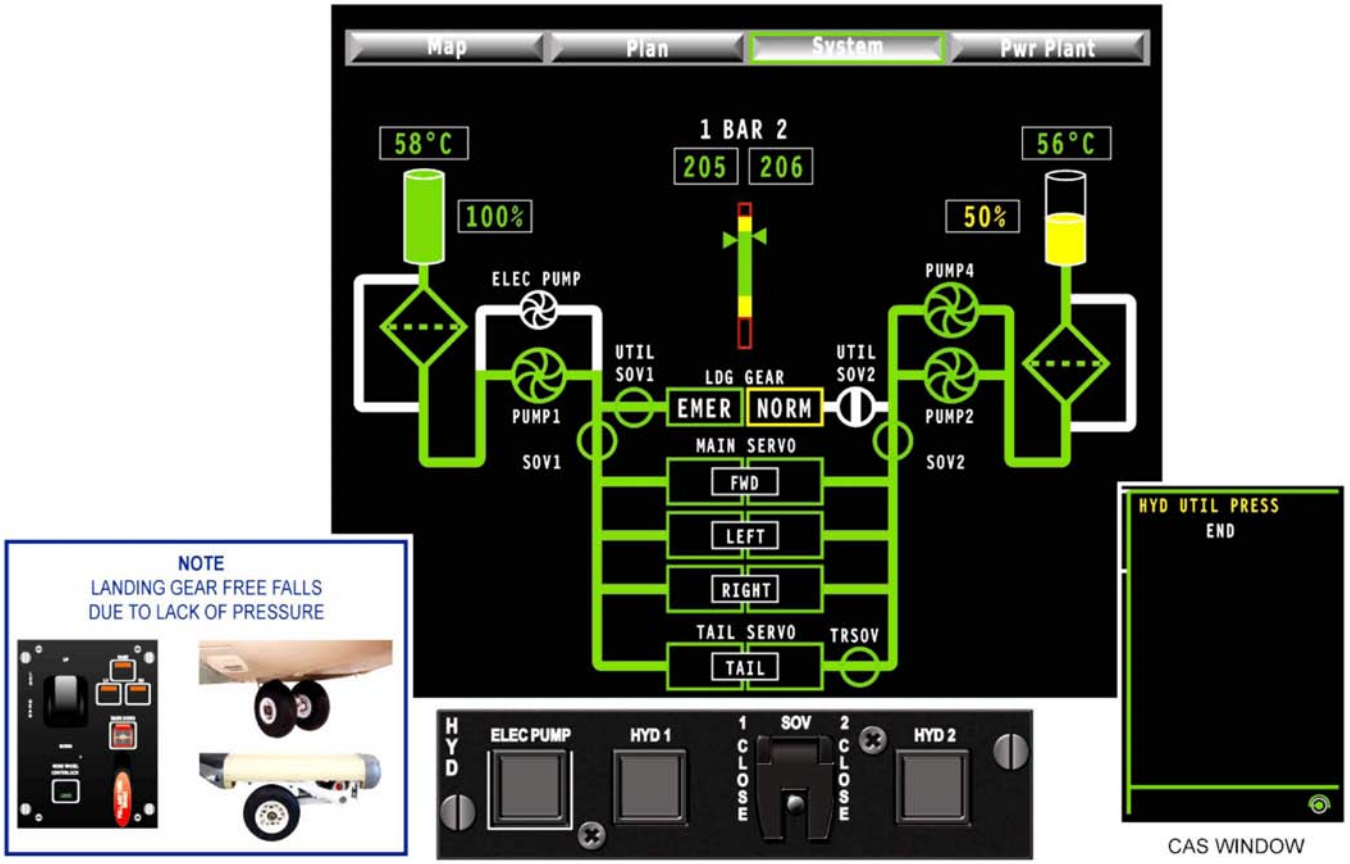
on hyd control panel

- HYD1 and HYD 2: blank



ICN-39-A-291100-G-A0126-00826-A-01-1

LOW LEVEL 1 - 50% - OPERATION MODE PCM2



ICN-39-A-291100-G-A0126-03304-A-01-1

FAILURE: HYD 2 FLUID LEVEL AT 50%

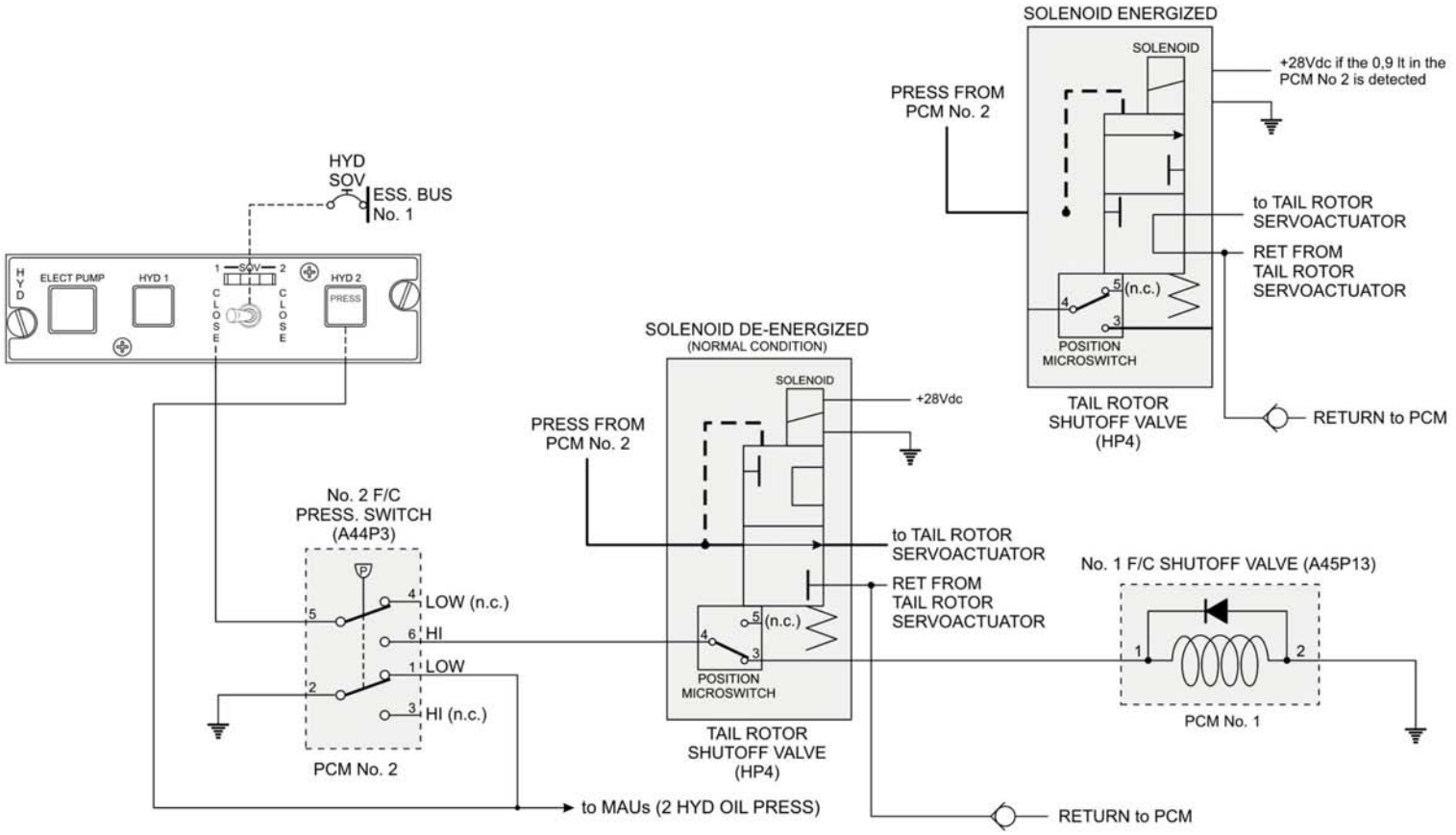
FAILURE – HYD 2 FLUID LEVEL AT 28%

hydraulic synoptic page

- PUMP1 pressurizes the system no.1
- PUMP2 and PUMP4 pressurize the system no.2
- UTIL SOV1 and UTIL SOV2 are opened. UTIL SOV2 is automatically re-opened by the 28% level microswitch
- SOV1 and SOV2 are opened and flight controls are pressurized at 207 bars by the system no.1
- TRSOV is automatically closed by the level microswitch
- a caution is displayed in the CAS window and the PLT has to follows the relevant malfunction procedure 2 SERVO

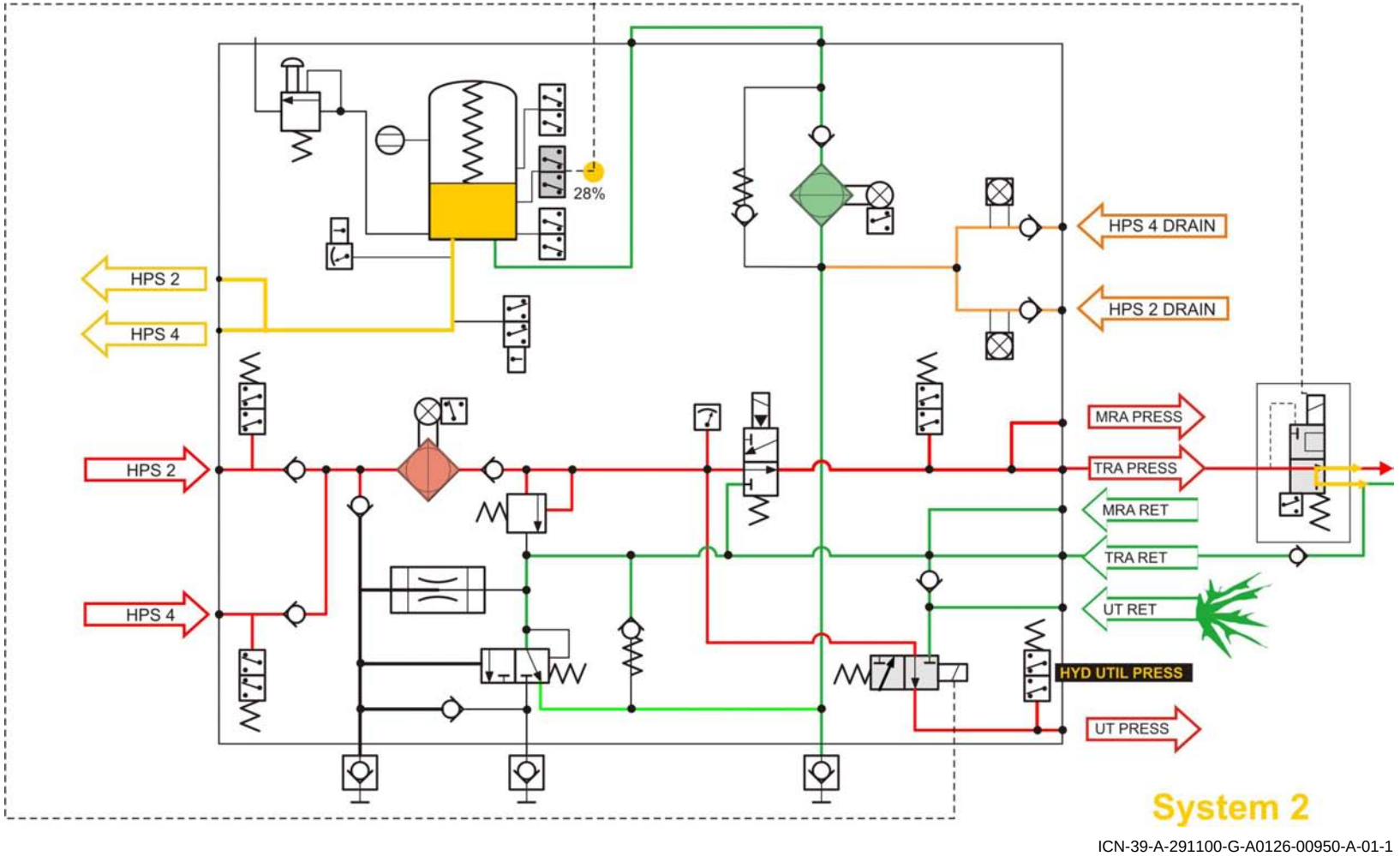
on hyd control panel

- HYD1 and HYD2: blank



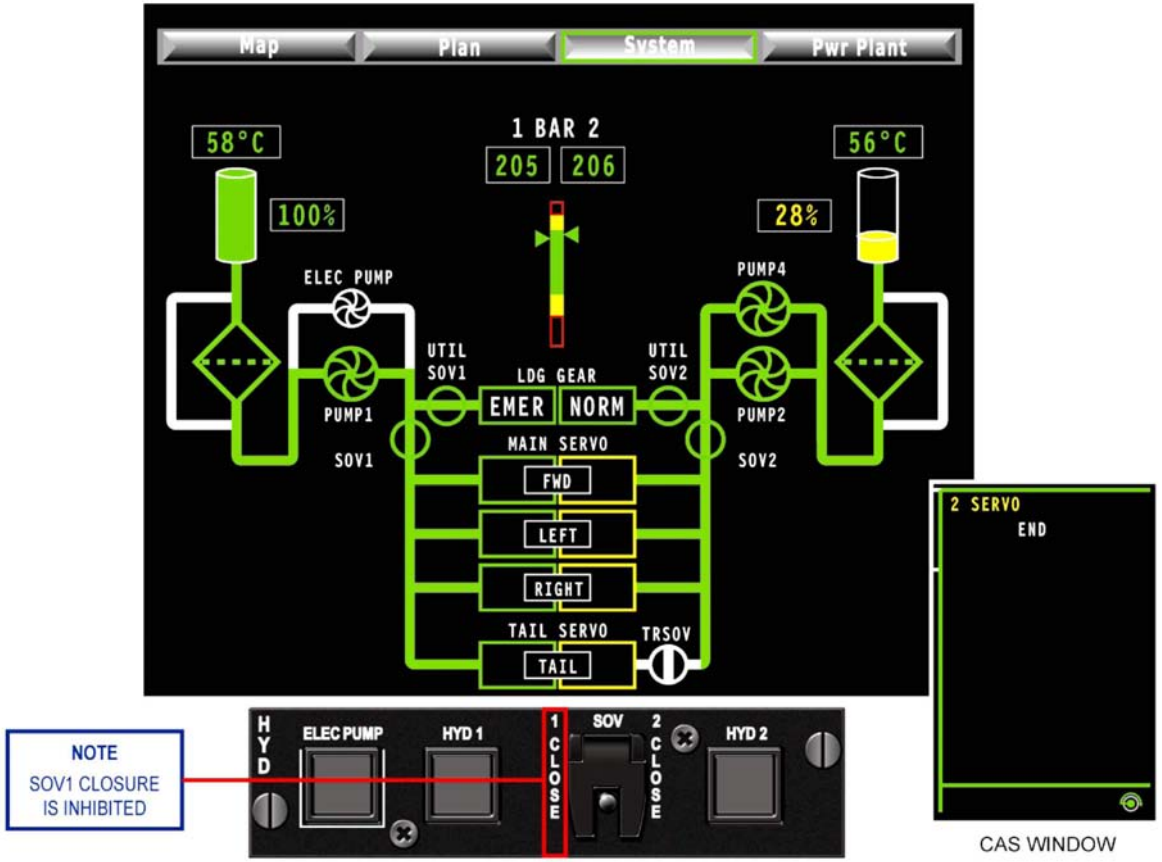
ICN-39-A-291100-G-A0126-00828-A-01-1

HYD CONTROL PANEL ELECTRICAL SCHEMATIC



ICN-39-A-291100-G-A0126-00950-A-01-1

LOW LEVEL 2 - 28% - OPERATION MODE PCM 2



ICN-39-A-291100-G-A0126-03305-A-01-1

MFD SYNOPTIC PAGE – FAILURE: HYD 2 FLUID LEVEL AT 28%

FAILURE – HYD 2 FLUID LEVEL AT 22% (MIN)

hydraulic synoptic page

- PUMP1 pressurizes the system no.1
- PUMP2 and PUMP4 pressurize the system no.2
- UTIL SOV 1 is opened and the landing gear emergency circuit is pressurized at 207 bar
- UTIL SOV2 and TRSOV are automatically closed by the minimum level microswitch (22%). The landing gear normal operation is not available and the landing gear free falls due to a lack of pressure
- cautions are displayed in the CAS window and the pilot has to follows the relevant malfunction procedure

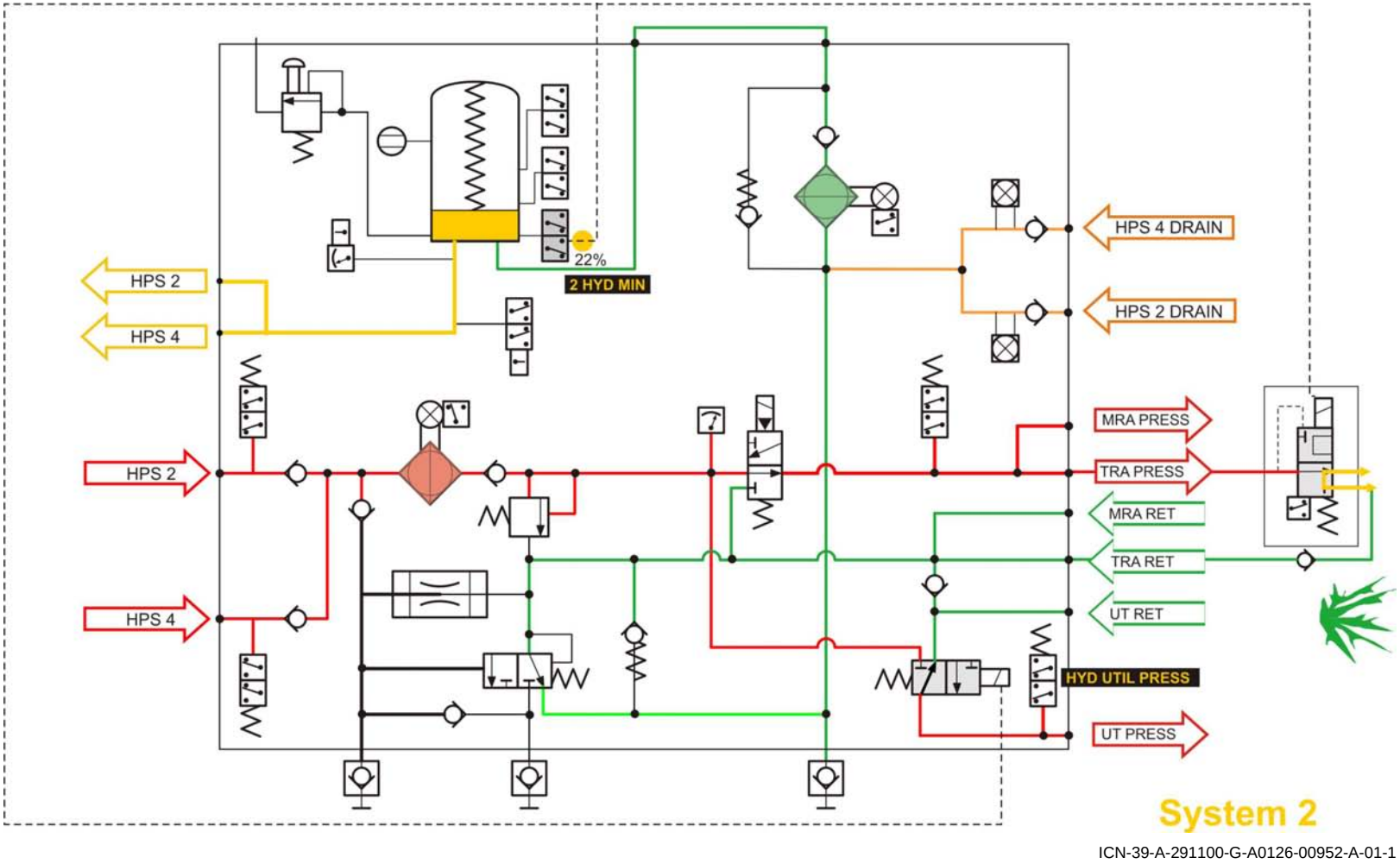
  2 SERVO

  HYD UTIL PRESS

  2 HYD MIN

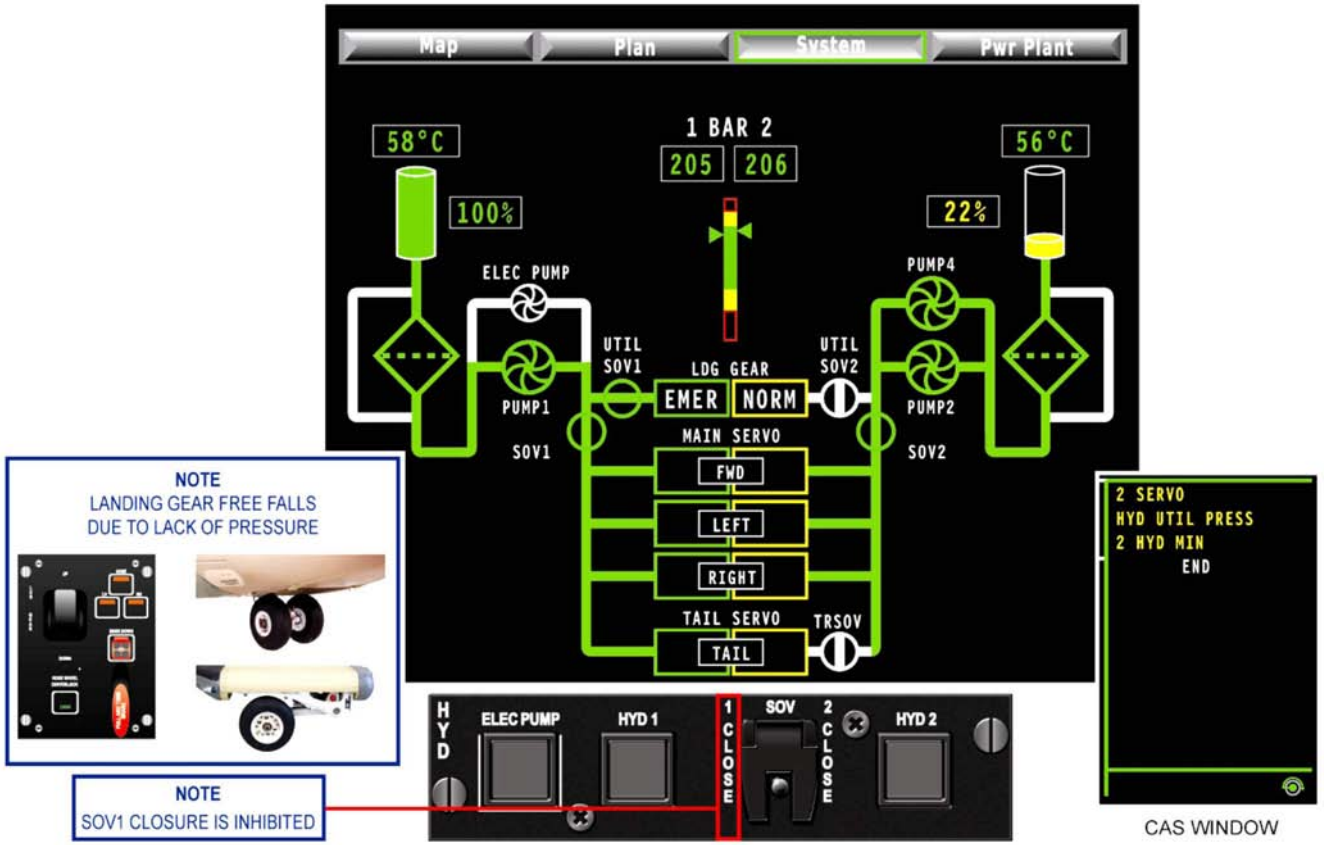
on hyd control panel

- HYD1 and HYD 2: blank



ICN-39-A-291100-G-A0126-00952-A-01-1

MIN LEVEL 22% - OPERATION MODE PCM 2



ICN-39-A-291100-G-A0126-03306-A-01-1

MFD SYNOPTIC PAGE – FAILURE: HYD 2 FLUID LEVEL AT 22% (MIN)

FAILURE – HYD 1 FLUID LEVEL AT 50%

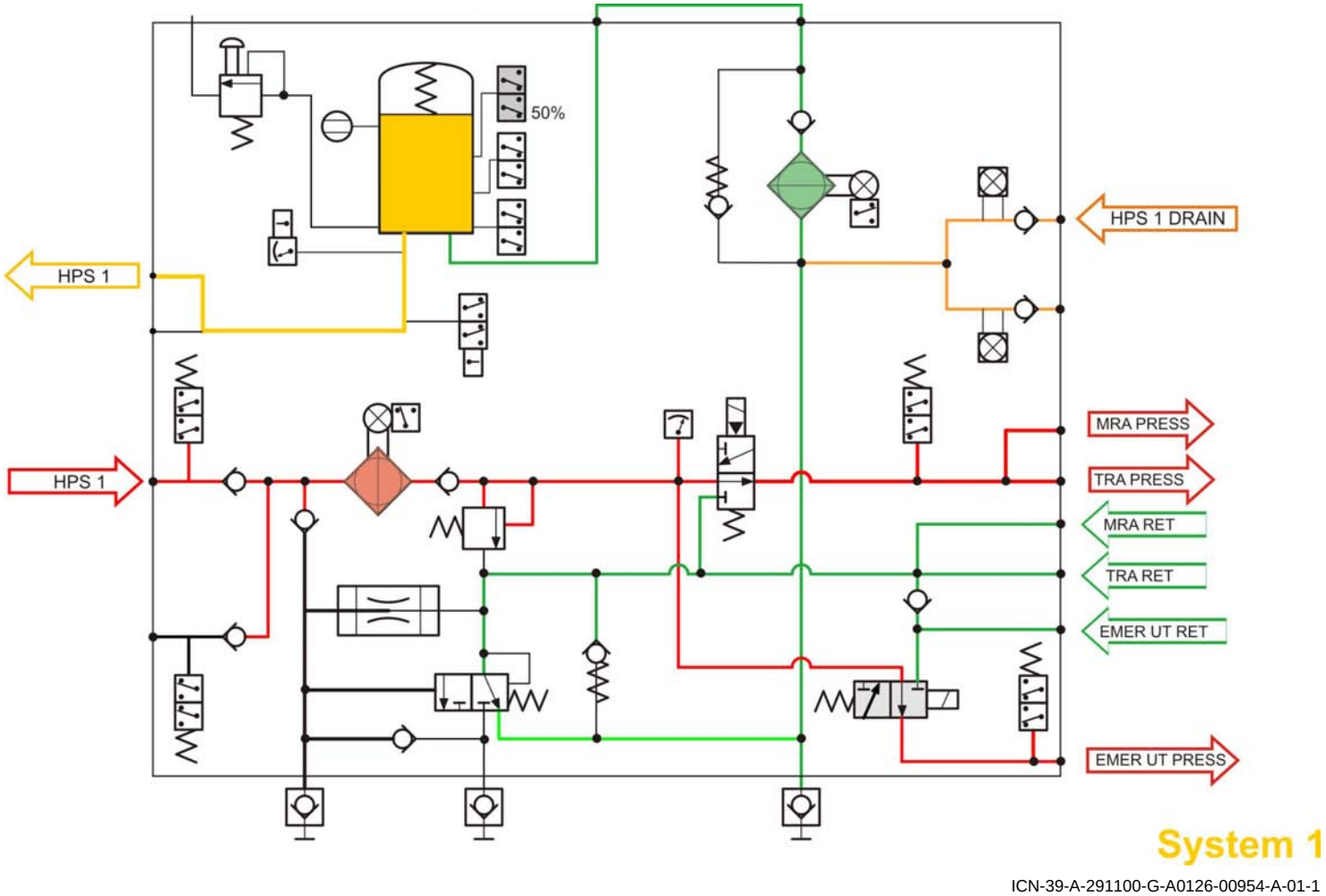
When the level of the fluid inside the PCM1 reservoir reaches the 50%, there will be no effects on the system operations. Only the indications in the synoptic page (reservoir and read-out amber) are provided.

hydraulic synoptic page

- PUMP1 pressurizes the system no.1
- PUMP2 and PUMP4 pressurize the system no.2
- UTIL SOV1 and UTIL SOV2 are opened and the landing gear circuit is pressurized at 207 bar
- SOV1, SOV2 and TRSOV are opened and the flight control circuit is pressurized at 207 bar

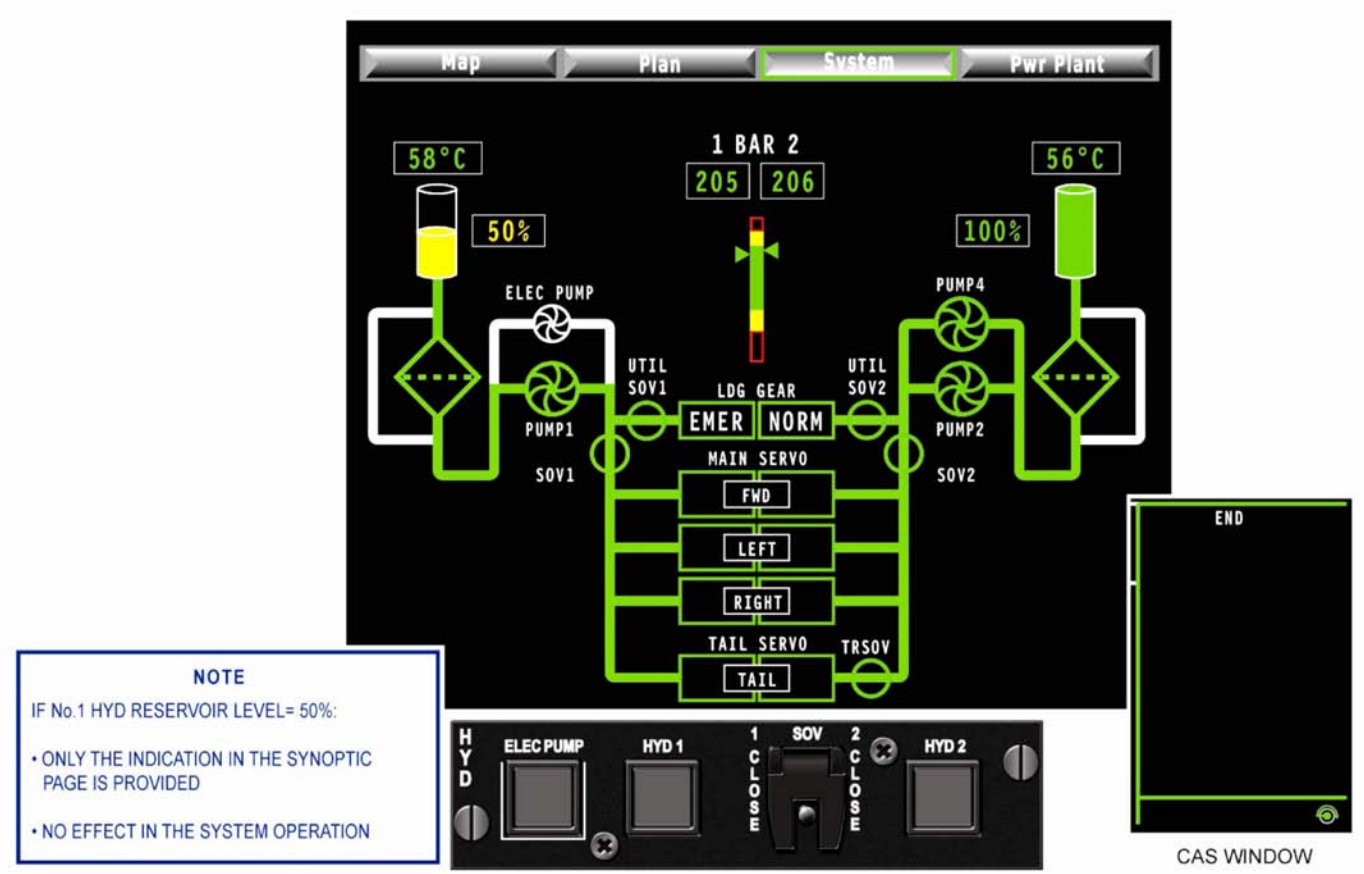
on hyd control panel

- HYD1 and HYD 2: blank



ICN-39-A-291100-G-A0126-00954-A-01-1

LOW LEVEL 1 - 50% - OPERATION MODE PCM 1



ICN-39-A-291100-G-A0126-03307-A-01-1

MFD SYNOPTIC PAGE – FAILURE: HYD 1 FLUID LEVEL AT 50%

FAILURE – HYD 1 FLUID LEVEL AT 28%

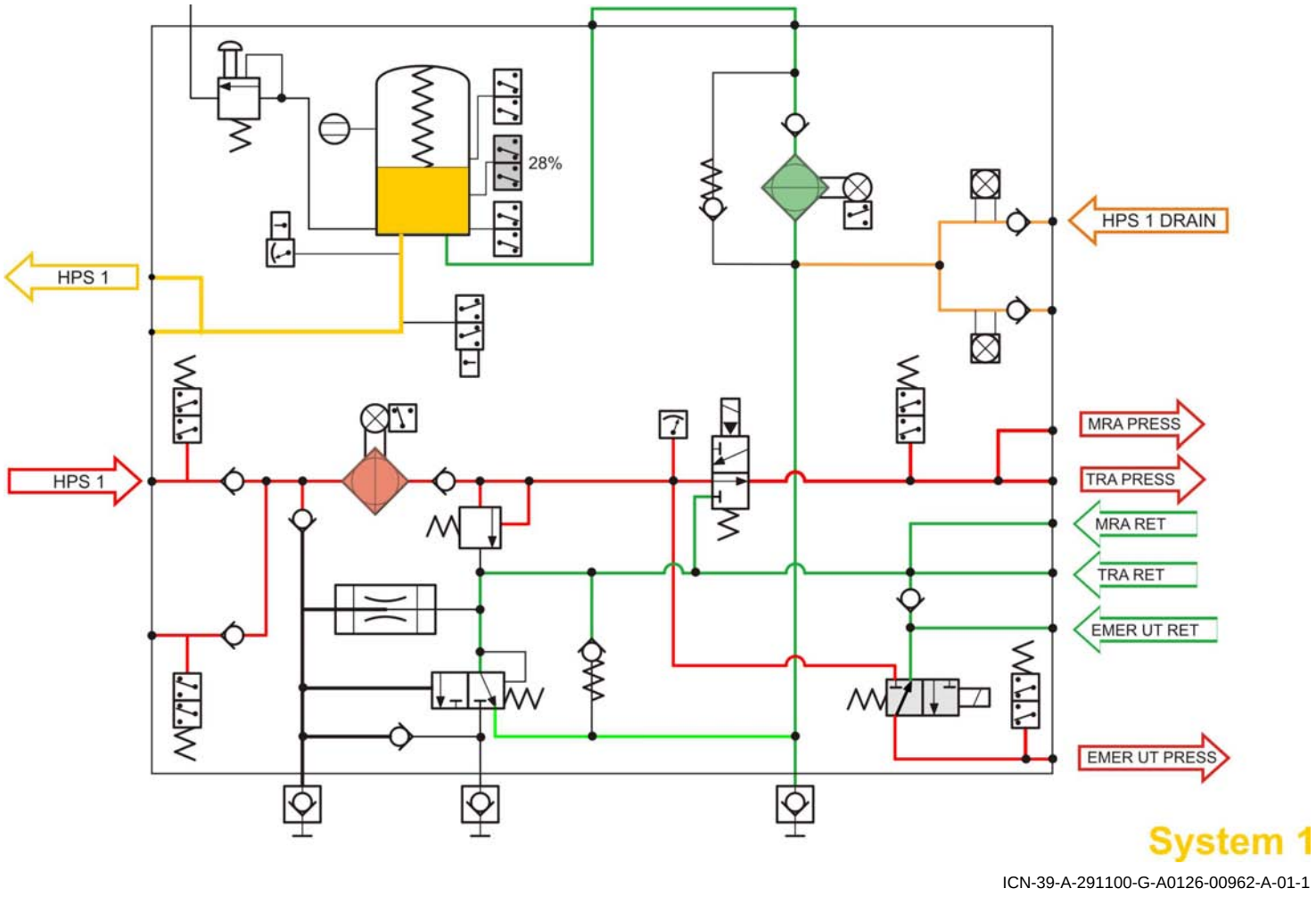
When the level of the fluid inside the PCM1 reservoir reaches the 28%, there will be no effects on the system operations. Only the indications in the synoptic page (reservoir and read-out amber) are provided.

hydraulic synoptic page

- PUMP1 pressurizes the system no.1
- PUMP2 and PUMP4 pressurize the system no.2
- UTIL SOV1 and UTIL SOV2 are opened and the landing gear circuit is pressurized at 207 bar
- SOV1, SOV2 and TRSOV are opened and the flight control circuit is pressurized at 207 bar

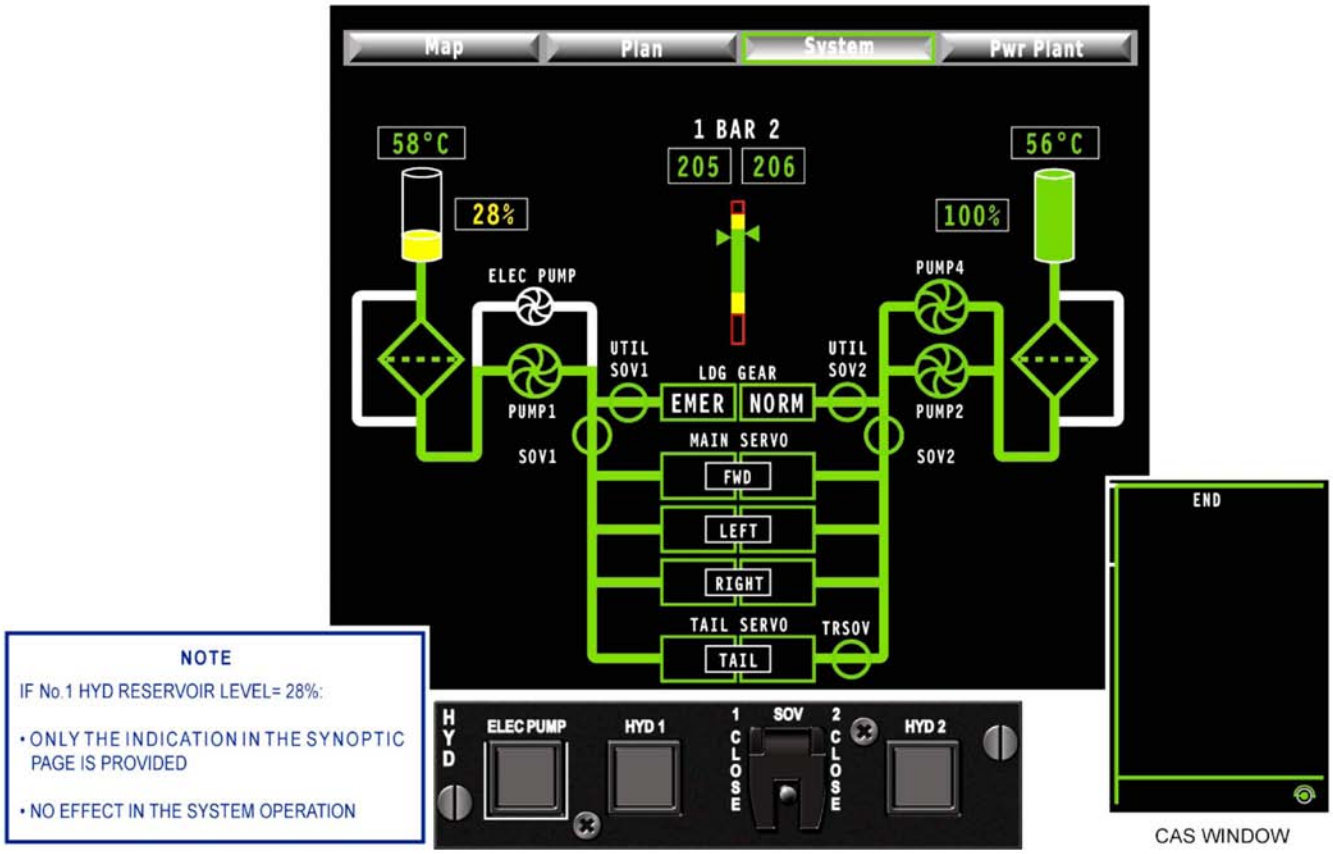
on hyd control panel

- HYD1 and HYD 2: blank



ICN-39-A-291100-G-A0126-00962-A-01-1

LOW LEVEL 2 28% - OPERATION MODE PCM1



ICN-39-A-291100-G-A0126-03308-A-01-1

MFD SYNOPTIC PAGE – FAILURE: HYD 1 FLUID LEVEL AT 28%

FAILURE – HYD 1 FLUID LEVEL AT 22% (MIN)

When the level of the fluid inside the PCM1 reservoir reaches the 22%, there will be no effects on the system operations.

hydraulic synoptic page

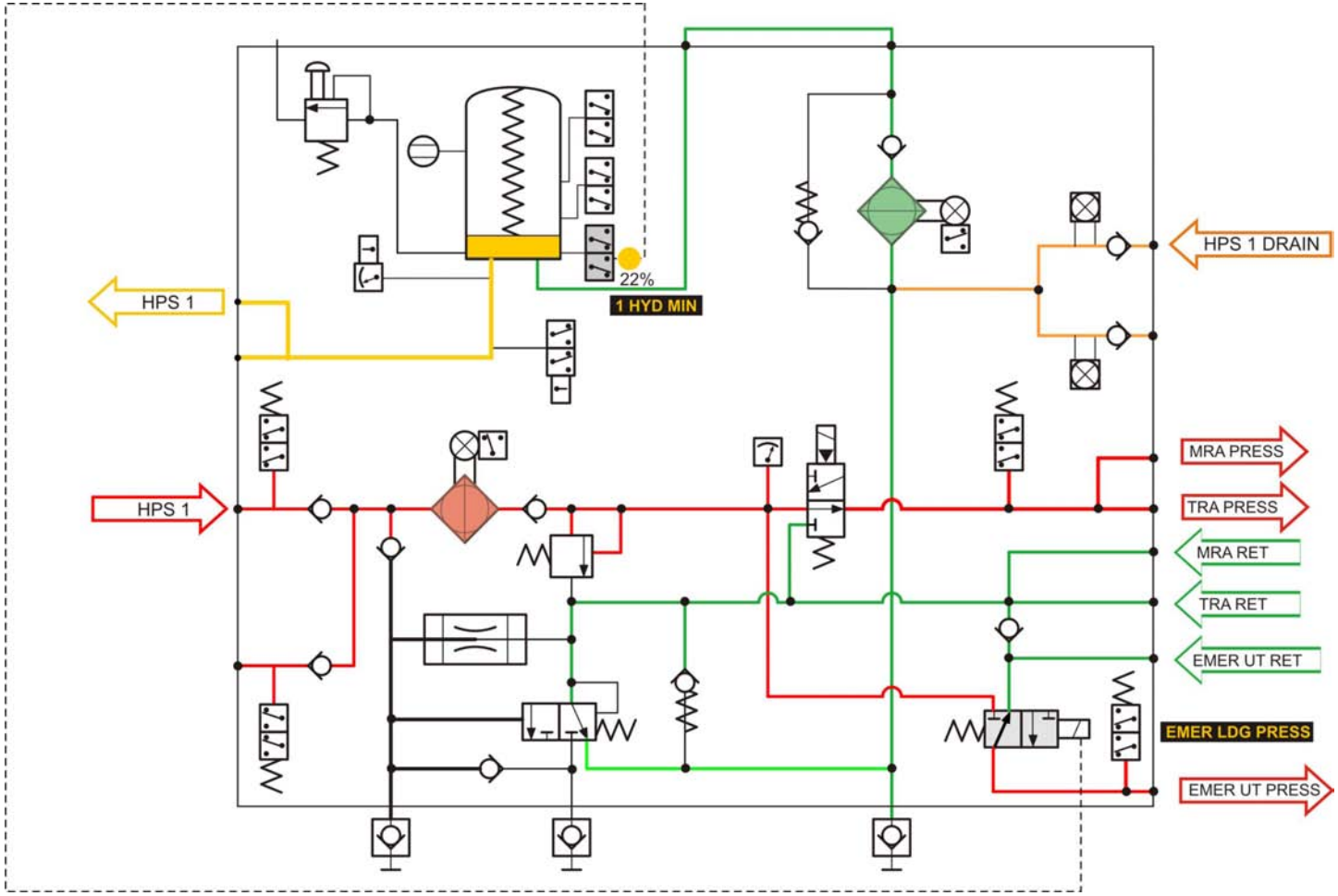
- PUMP1 pressurizes the system no.1
- PUMP2 and PUMP4 pressurize the system no.2
- UTIL SOV1 automatically closed by the minimum level microswitch (22%) and the landing gear emergency operation is not available
- UTIL SOV2 and TRSOV are opened.
- SOV1, SOV2 and TRSOV are opened and the flight control circuit is pressurized at 207 bar
- cautions are displayed in the CAS window and the pilot has to follows the relevant malfunction procedure

  1 HYD MIN

  EMER LDG PRESS

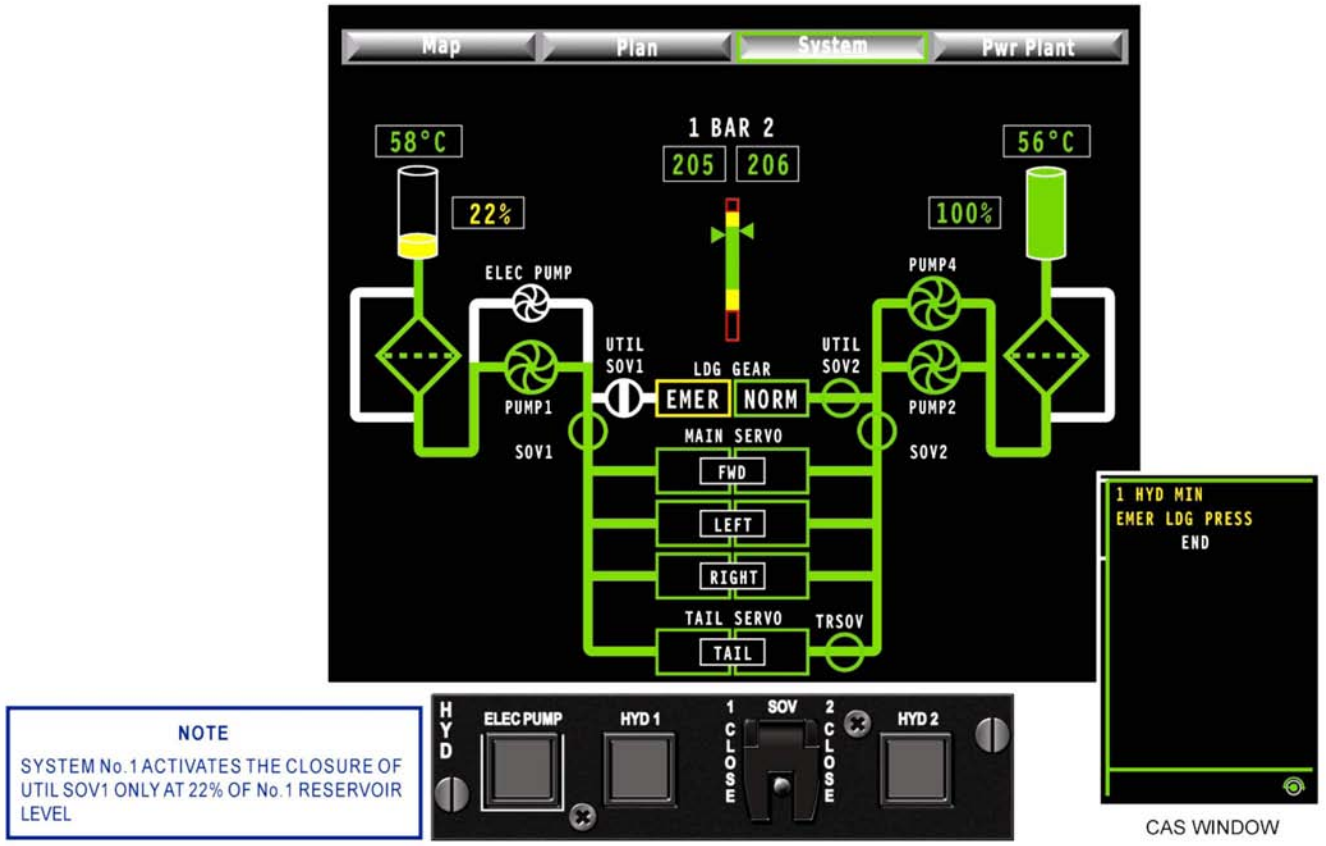
on hyd control panel

- HYD1 and HYD 2: blank



ICN-39-A-291100-G-A0126-00964-A-01-1

LOW LEVEL 22% - OPERATION MODE PCM 1



CAS WINDOW

ICN-39-A-291100-G-A0126-03309-A-01-1

MFD SYNOPTIC PAGE – FAILURE: HYD 1 FLUID LEVEL AT 22% (MIN)

FAILURE – SYSTEM 2 OVERHEATING

When the hydraulic fluid reaches an overheating condition (the temperature is greater than 134°C), the system (for example the no.2) must be isolated.

hydraulic synoptic page

- check to confirm the system no.2 over temperature
- lower the LDG GEAR following the normal procedure
- lift the cover of the SOV switch on the HYD control panel and move the switch to 2 CLOSE
- cautions are displayed in the CAS window and the pilot has to follows the relevant malfunction procedure

  2 HYD OIL TEMP

  2 HYD OIL PRESS

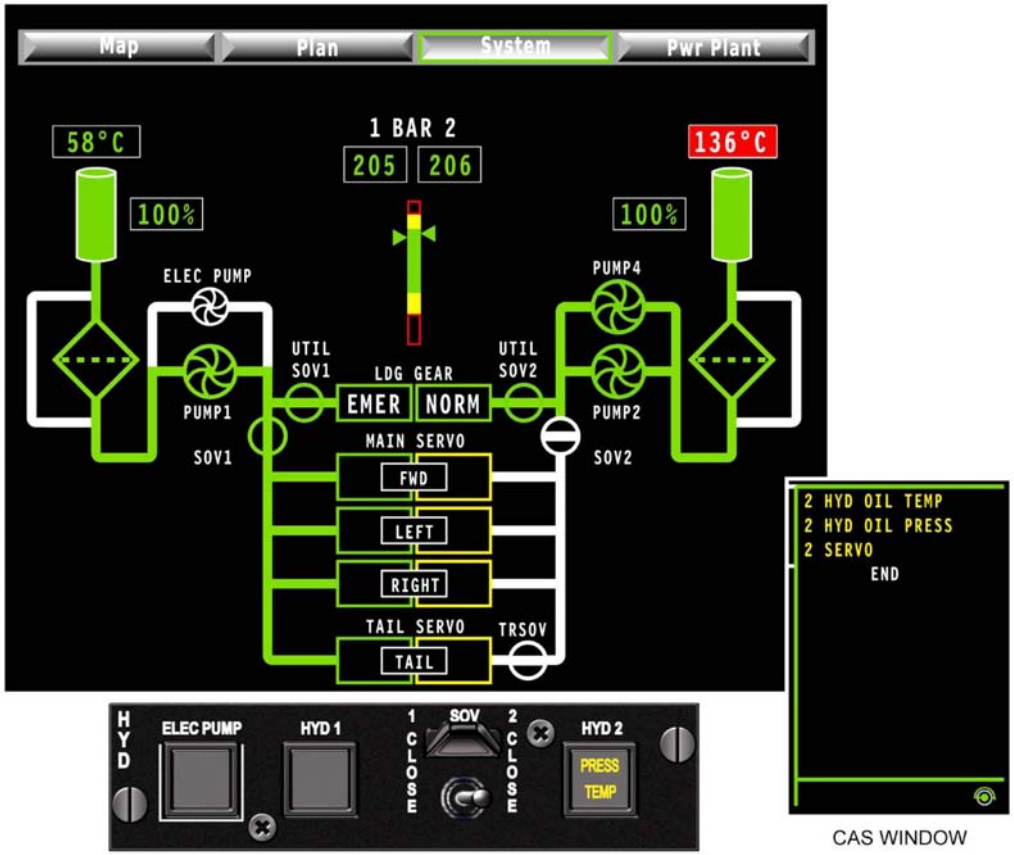
  2 SERVO

NOTE A

With one hydraulic system SOV shut off, a subsequent drop of pressure in the other system will override the SOV selection and reinstate pressure to the servo's. In these conditions the SOV switch will not be automatically reset.

on hyd control panel

- HYD 1: blank
- HYD 2: TEMP and PRESS lighted (amber)



ICN-39-A-291100-G-A0126-03310-A-01-1

MFD SYNOPTIC PAGE – FAILURE: SYSTEM 2 OVERHEATING

FAILURE – 2 SERVO AND SYSTEM 1 OVERHEATING

When 2 SERVO and 1 HYD OIL TEMP cautions are both illuminated, the SOV switch on the HYD control panel is not inhibited.

Moving to 1 CLOSE will cause loss of control in no.1 servo-jack.

NOTE B

The SOV switch is ineffective when 1 (2) HYD OIL PRESS caution is active.

hydraulic synoptic page

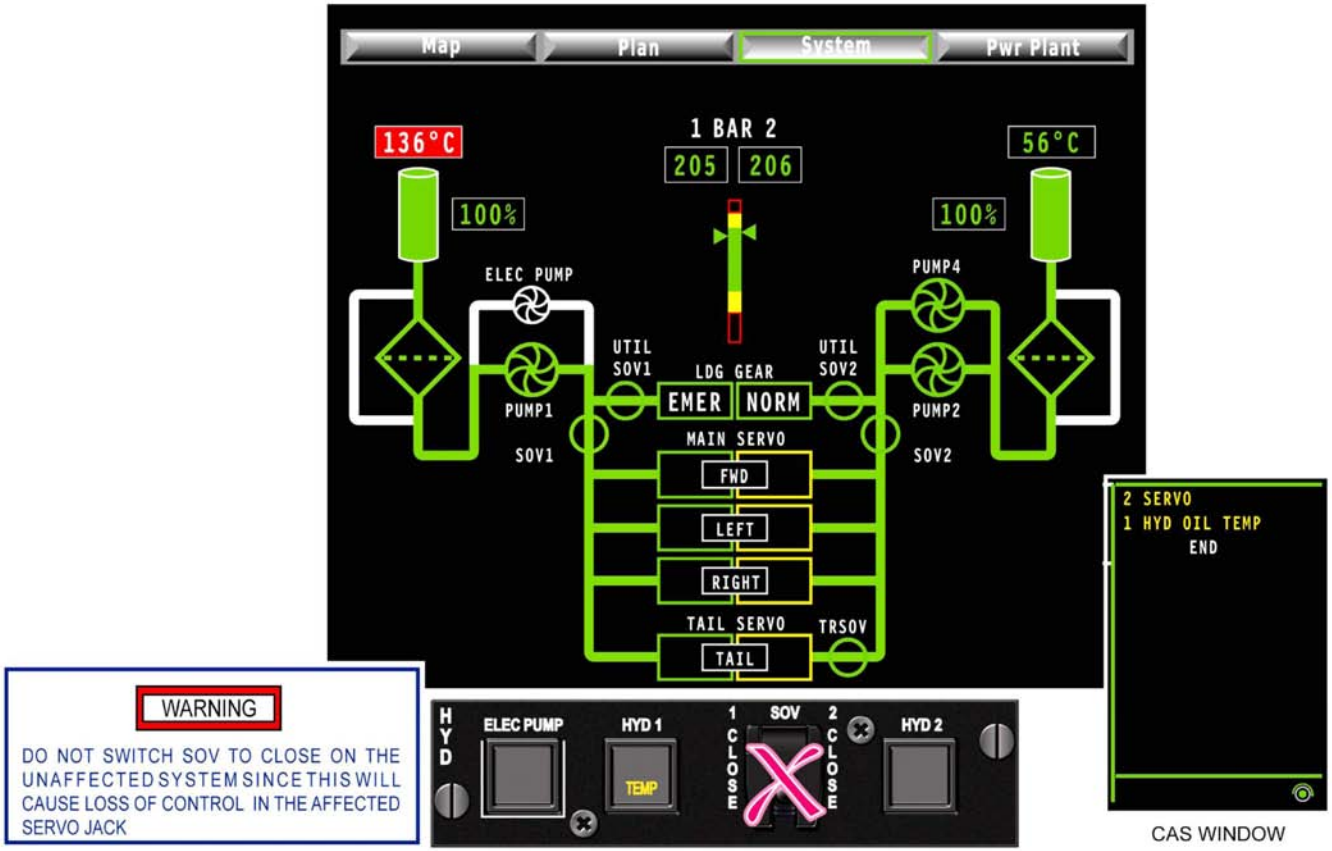
- check to confirm the system no.1 over temperature
- cautions are displayed in the CAS windows and the pilot has to follows the relevant malfunction procedure

  2 SERVO

  1 HYD OIL TEMP

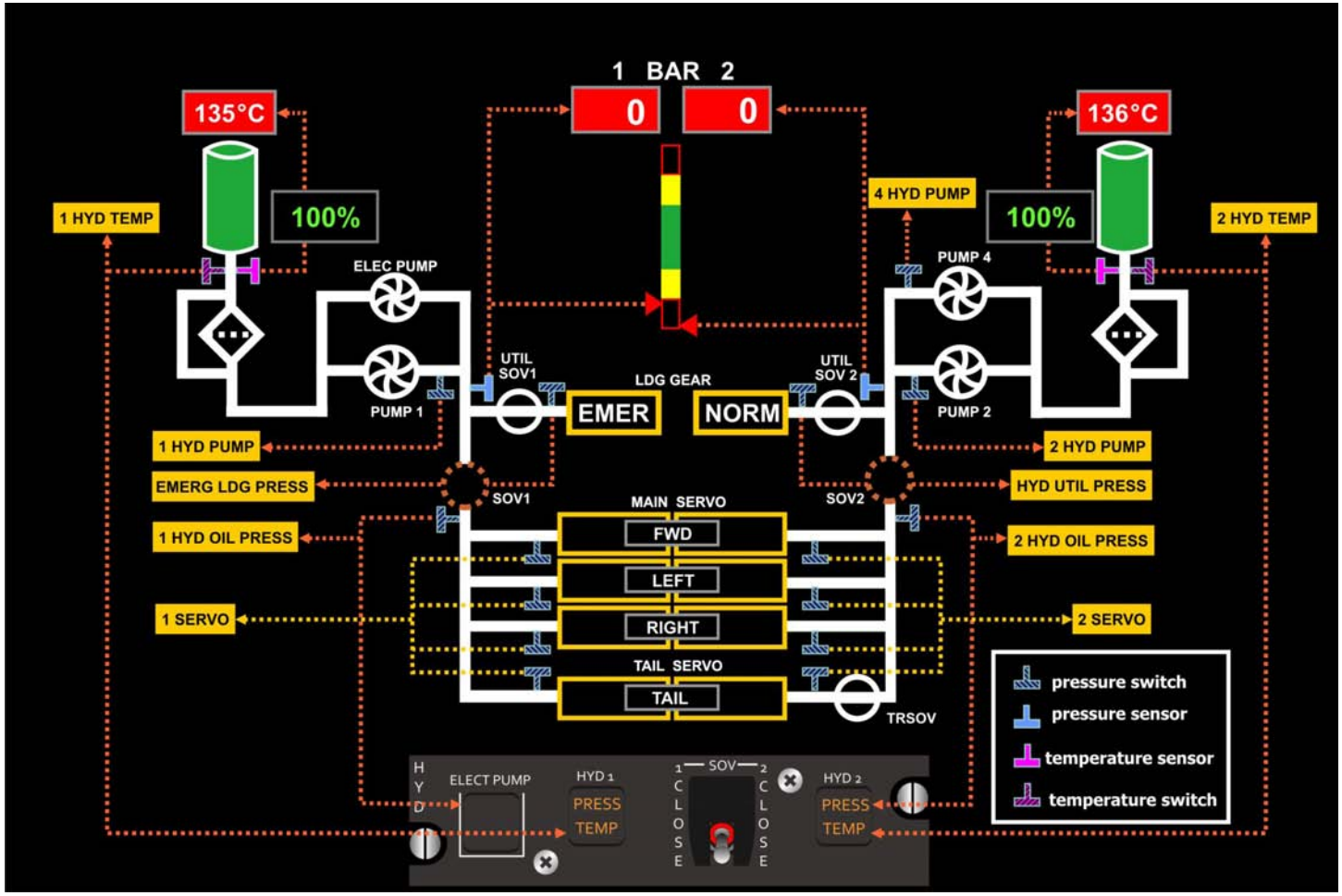
on hyd control panel

- HYD 1: TEMP lighted (amber)
- HYD 2: blank



ICN-39-A-291100-G-A0126-03311-A-01-1

MFD SYNOPTIC PAGE – FAILURE: 2 SERVO AND SYSTEM 1 OVERHEATING



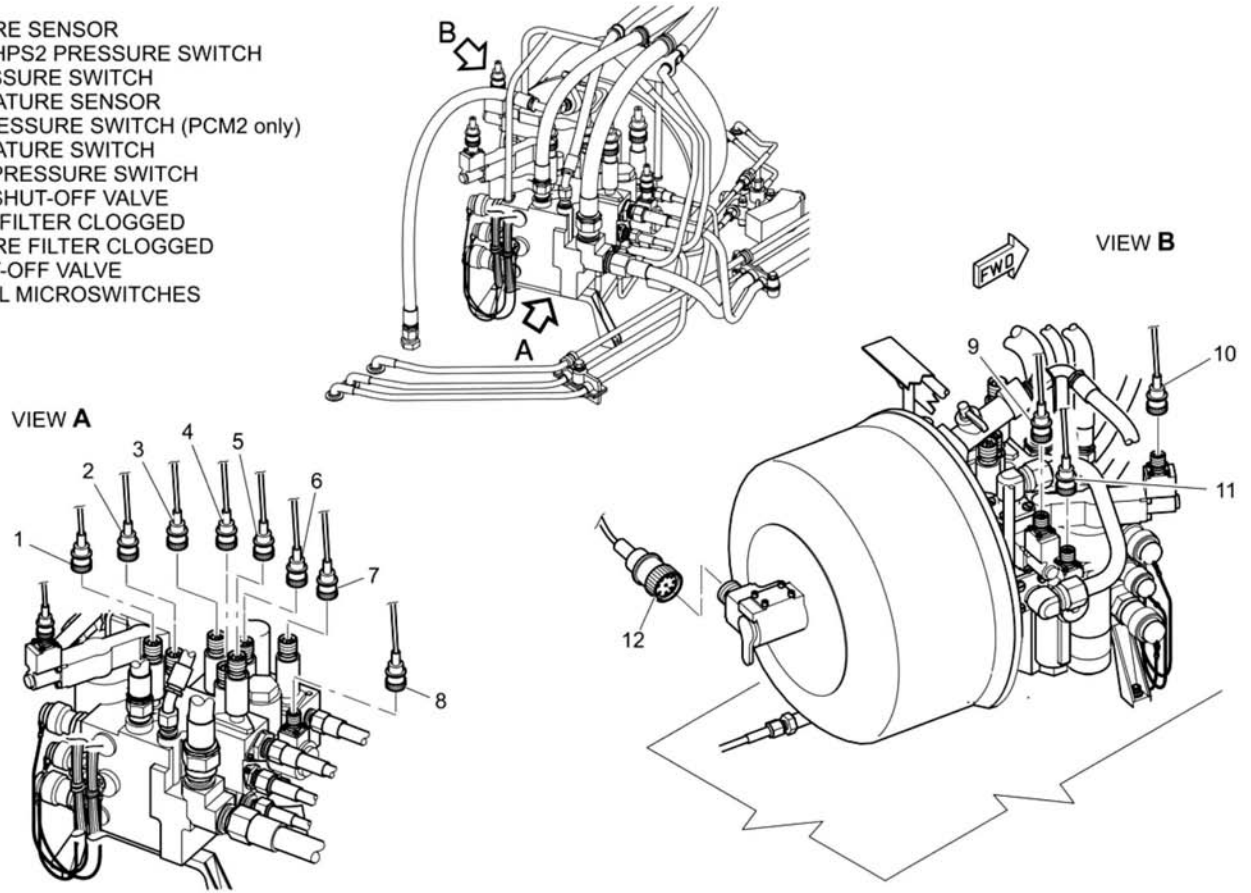
ICN-39-A-291100-G-A0126-03312-A-01-1

SENSORS AND CAUTIONS

CAS CAUTION MESSAGES	
CAS CAPTION	**FAILURE DESCRIPTION**
1 HYD OIL PRESS 1 HYD PUMP 1 SERVO EMER LDG PRESS	Loss of pressure in associated hydraulic system (less than 163 bar)
2 HYD OIL PRESS 2-4 HYD PUMP 2 SERVO HYD UTIL PRESS	Loss of pressure in associated hydraulic system (less than 163 bar)
HYD UTIL PRESS	Low pressure in landing gear NORM hydraulic system

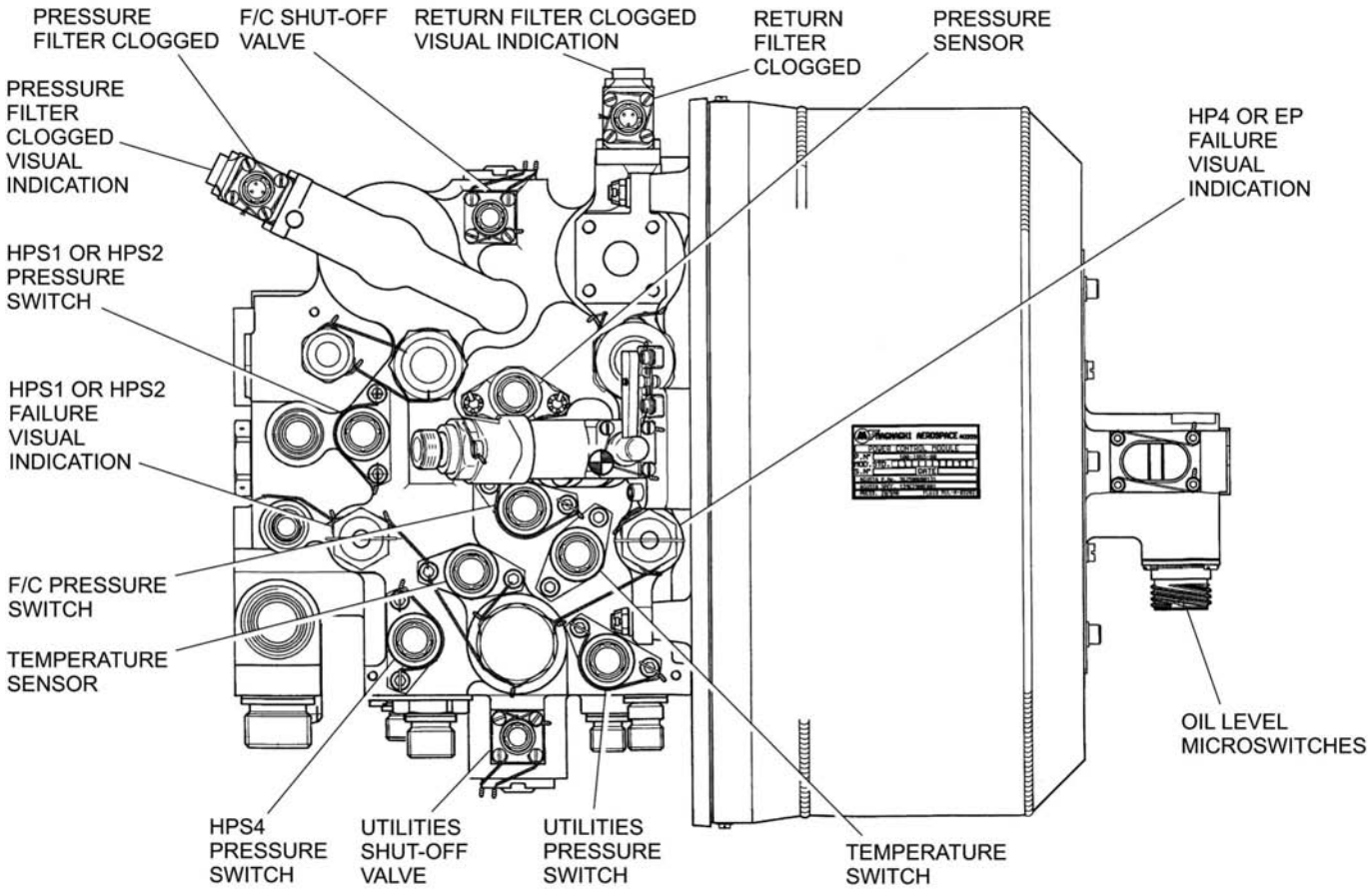
CAS CAUTION MESSAGES	
CAS CAPTION	**FAILURE DESCRIPTION**
EMER LDG PRESS	Low pressure in emergency landing gear hydraulic system
1(2) HYD OIL TEMP	Associated hydraulic system overheat (greater than 134°C)
1(2) HYD MIN	Associated system low hydraulic fluid level
1(2)(4) HYD PUMP	Low pressure at pump outlet

1. PRESSURE SENSOR
2. HPS1 or HPS2 PRESSURE SWITCH
3. F/C PRESSURE SWITCH
4. TEMPERATURE SENSOR
5. HPS4 PRESSURE SWITCH (PCM2 only)
6. TEMPERATURE SWITCH
7. UTILITY PRESSURE SWITCH
8. UTILITY SHUT-OFF VALVE
9. RETURN FILTER CLOGGED
10. PRESSURE FILTER CLOGGED
11. F/C SHUT-OFF VALVE
12. OIL LEVEL MICROSWITCHES



ICN-39-A-291100-G-A0126-03313-A-01-1

POWER CONTROL MODULE ELECTRICAL CONNECTORS (SHEET 1 OF 2)



ICN-39-A-291100-G-A0126-03314-A-01-1

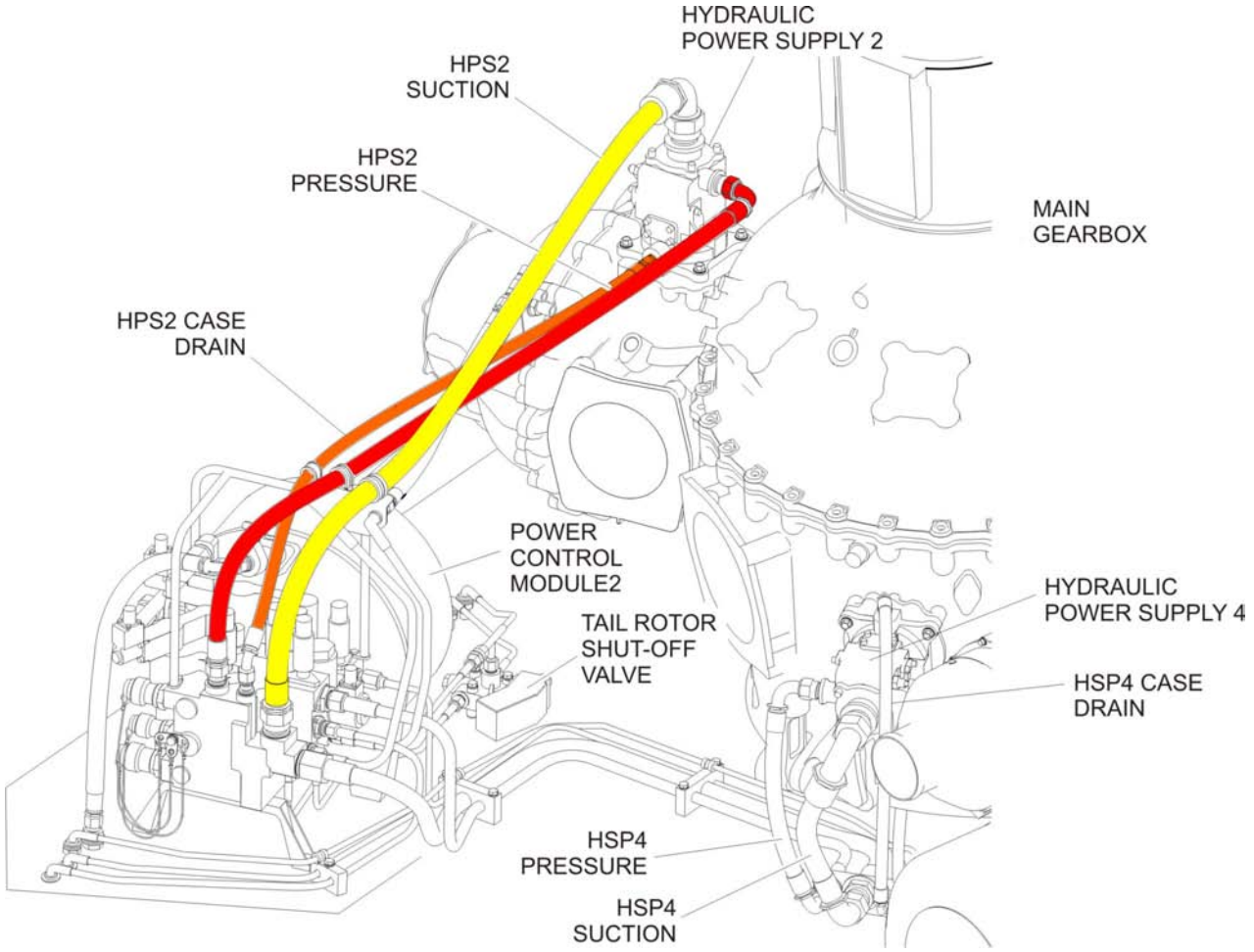
POWER CONTROL MODULE ELECTRICAL CONNECTORS (SHEET 2 OF 2)

SECTION 12 - HYDRAULIC POWER SUPPLY

DESCRIPTION

The Hydraulic Power Supply 1 is located on the top of the left engine input stage of the Main Gearbox.
The Hydraulic Power Supply 2 is located on the top of the right engine input stage of the Main Gearbox.
The Hydraulic Power Supply 4 is located on the left front side of the Main Gearbox.
The pump units are variable delivery, constant pressure, self regulating axial piston type.
Variable flow demand is achieved by altering the angle of the swashplate which controls the stroke of the nine pistons.
The pistons displacement (swashplate angle) is controlled by an internal pressure compensator valve.
The pump incorporates a centrifugal boost impeller to reduce risk of cavitation.
A cooling and lubrication flow (Case Drain Flow) passes through the pump casing and is taken from a drain port and piped to the relative port on the associated PCM.
The three pumps are identical; the lower flow requirement on the HPS 4 is achieved by driving it at slower speed.
The mechanical pumps have a shear shaft designed to break if the driving torque exceeds a preset value; the shear coupling is hold in place by a positive retainer.
This feature prevents damage to the gearbox in the event of a pump seizure.
The hydraulic pumps are provided with four ports for suction, pressure, case drain and shaft seal leak drain.
The mechanical pumps are driven by different gearbox shafts to provide maximum separation of power sources for safety purposes.
The mechanical pumps are interfaced with:

- Power Control Module 1 and 2.
- Main Gearbox.



ICN-39-A-291200-G-A0126-01017-A-01-1

HYDRAULIC POWER SUPPLY 2 AND 4 HYDRAULIC LINES

OPERATION

- The pump cylinder housing containing the nine pistons is driven by a shaft.
- As the cylinder housing rotates, the pistons reciprocate within their bores intaking (from PCM suction line) and discharging fluid (to PCM pressure line)
- Varying flow demands are satisfied by charging the angle of the swashplate which controls the nine pistons stroke.
- A compensating valve senses the pump output pressure and a stroking piston, together with a spring, controls the swashplate.
- As the pressure drops, the swashplate is tilted producing increased flow to the system.
- As soon pressure reaches the preset value, swashplate angle is reduced thus reducing fluid flow.
- When no demands are made on the system, fluid flow to the system will be zero and fluid supplied by the pump is delivered completely to the case drain line (to PCM case drain line).
- The case drain line is required to provide lubrication and heat dissipation to the pump internal rotating components.
- Fluid is provided to the case drain line during any pump operating condition.

MAINTENANCE INFORMATION

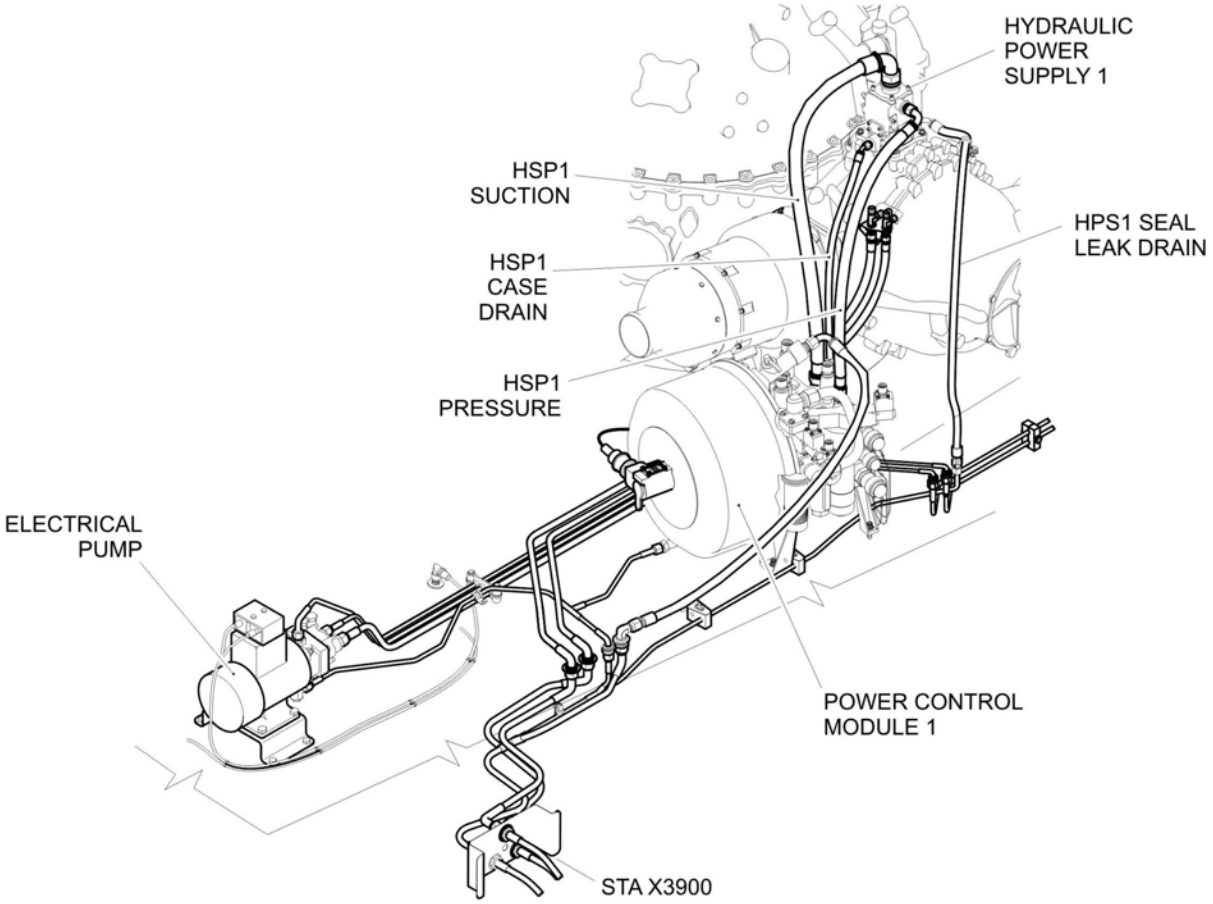
- Before HPS removal, the associated PCM must be drained.
- The splines on the HPS shaft must be lubricated with the same oil of the main gearbox before installation.
- An HPS operational check must be carried out after replacement.

LEADING PARTICULARS

- HPS1/HPS2 flow rate..... 35 l/min (9 US gall/min)
- HPS4 flow rate..... 30 l/min (7.9 US gall/min)
- HPS1/2/4 rated pressure..... 207 bar (3000 psi)

SAFETY PRECAUTIONS

- The grease is a dangerous material.
- Make sure that you know all the safety precautions and first aid instructions for grease.



ICN-39-A-291200-G-A0126-01018-A-01-1

HYDRAULIC POWER SUPPLY 1 HYDRAULIC LINES

SECTION 13 - TAIL ROTOR SHUT-OFF VALVE

DESCRIPTION

The Tail Rotor Shut-Off Valve is a three ways, two position solenoid operated valve located on the right side of the upper deck in front of the Main Gearbox. It is composed of a spool operated by a solenoid which is normally de-energized (electrical power removed).
The spool is also controlled by a spring which keeps it in the open position.
A microswitch is provided to give a signal to the MFD.
The Tail Rotor Shut-Off Valve is interfaced with:

- Central Fuselage
- Power Control Module 2
- Modular Avionics Unit
- Electrical System

OPERATION

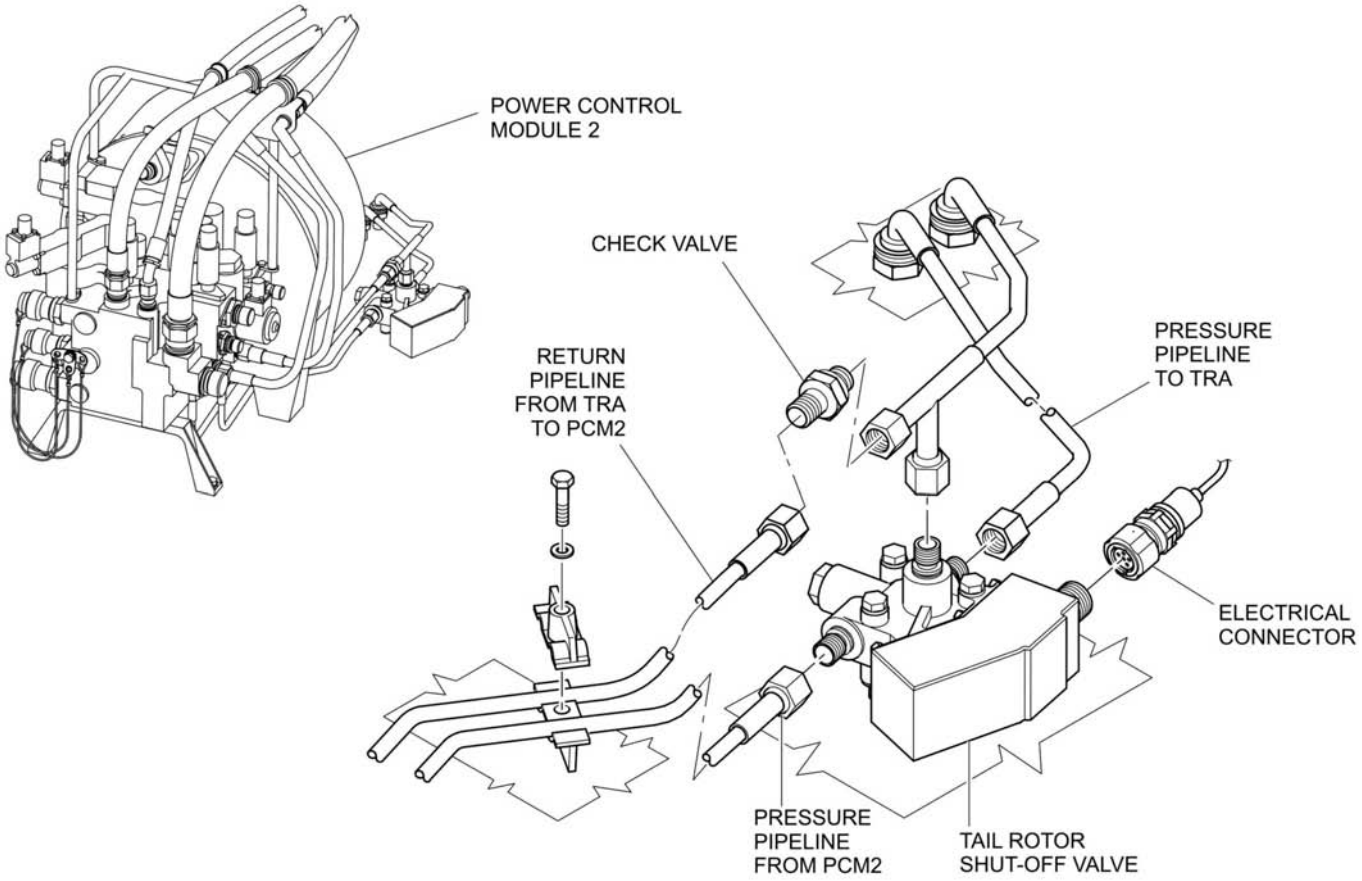
- During normal operation solenoid is de-energized, the spool is positioned by the spring to keep the valve open thus providing pressure flow to the tail servo actuator pressure line.
- When the LOW LEVEL 2 microswitch in the Power Control Module 2 detects a low oil level condition (28%), an electrical signal is provided to the valve to energize the solenoid to close the pressure supply to the tail rotor.
  Tail rotor servo actuator pressure line is then connected to the return.
  The caution 2 SERVO and TAIL SERVO TRSOV indicator condition are provided to the MFD.

MAINTENANCE INFORMATION

An operational test of the TRSOV must be carried out after TRSOV and check valve replacement.

LEADING PARTICULARS

- Rated flow..... 3.2 l/min (3.4 US quarts/min)
- Electrical supply..... 28 VDC ESS BUS 2



ICN-39-A-291300-G-A0126-01053-A-01-1

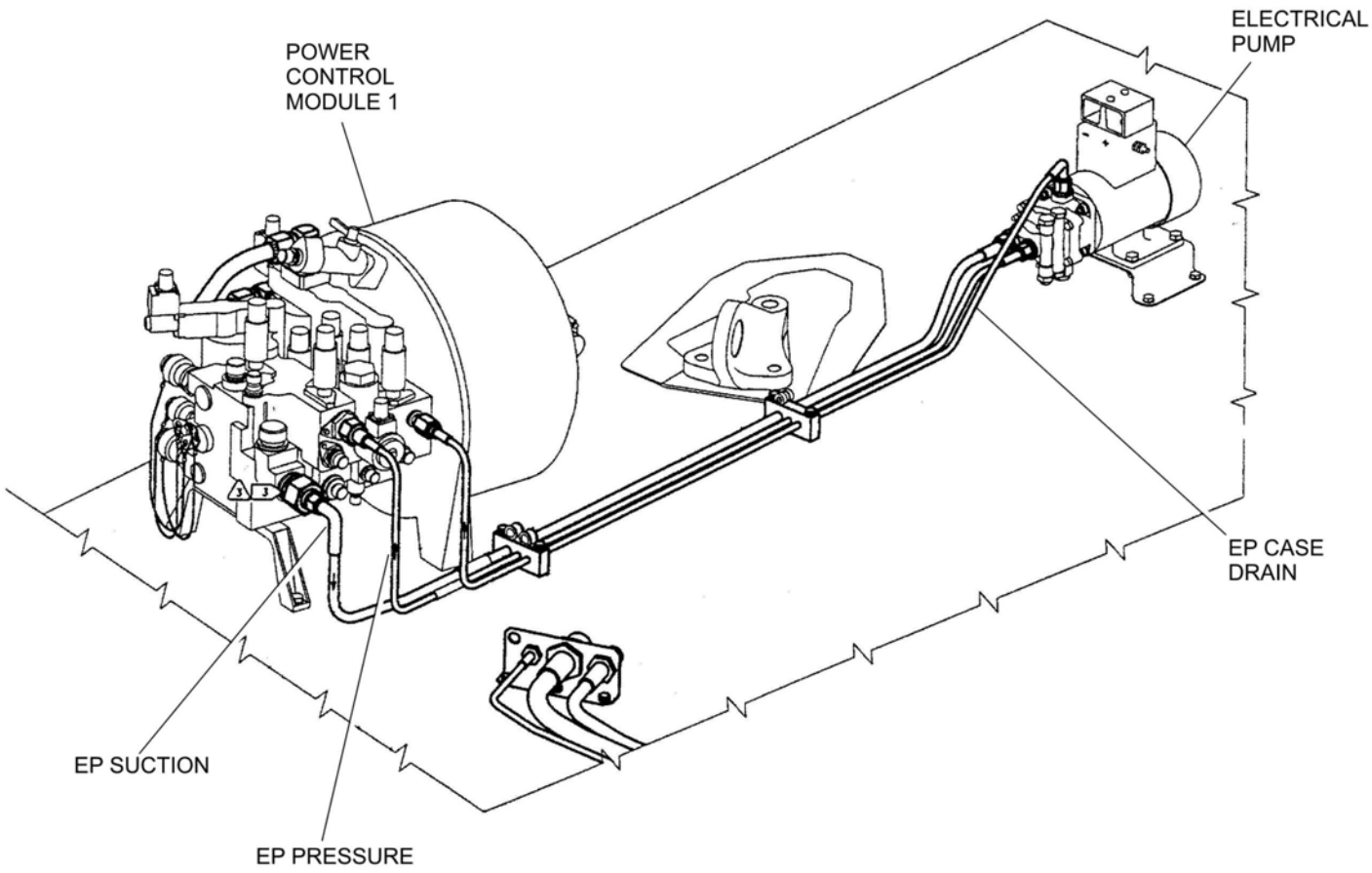
TAIL ROTOR SHUT-OFF VALVE

SECTION 20 - AUXILIARY

DESCRIPTION

The purpose of the Auxiliary Hydraulic Power System is to provide a limited hydraulic flow to operate the main and tail rotor servo actuators AT A LIMITED PRESSURE OF 90/105 bar for pre-flight checks only.
The Auxiliary Hydraulic Power System is composed by an Electrical Pump (EP) which powers with reduced pressure the Flight Controls hydraulic circuit 1.
This power takes place of the main hydraulic system, when engines are shut-down, for a limited period of time for pre-flight checks only.
The Electrical Pump is located on the upper deck in front of the Main Gearbox and is connected with the Power Control Module 1.
The pump unit is a variable delivery, constant pressure, self-regulating axial piston type.
Variable flow is achieved by altering the angle of the swashplate which controls the stroke of the nine pistons.
The pistons displacement (swashplate angle) is controlled by an internal pressure compensator valve.
The pump incorporates a centrifugal boost impeller to reduce risk of cavitation.
A cooling and lubrication flow (case drain flow) passes through the pump casing and is taken from a drain port and piped to the relative port on the PCM1.
The Electrical Pump is driven by a fan cooled DC electrical motor. It is explosion proofed, equipped with a radio noise filter and overheating switch.
A timer relay limits motor operation to two minutes maximum to avoid battery discharge and prevent motor overheating.
The Electrical Pump is connected to the PCM1 with three pipelines.
The Electrical Pump is interfaced with:

- Central Fuselage
- Hydraulic Control Panel
- Main Hydraulic Power System (Power Control Module 1)
- Electrical system
- Modular Avionics Unit



ICN-39-A-292000-G-A0126-01101-A-01-1

ELECTRICAL PUMP HYDRAULIC PIPELINES GENERAL LAYOUT

CONTROLS AND INDICATORS

The Auxiliary Hydraulic Power System control is housed on the Hydraulic System Control Panel and is installed on the console.
The operative function is provided on the same control panel.
The Hydraulic System Control Panel provides the following function and indication:

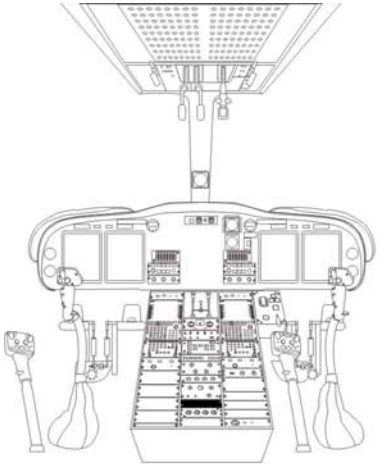
1 ELEC PUMP push-button:

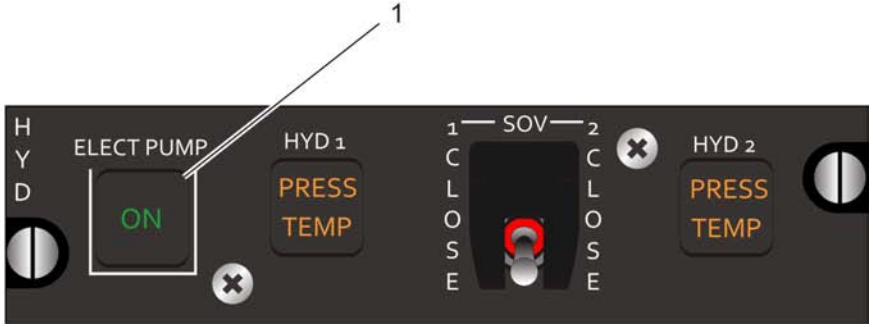
- Blank: ..... when not pressed, no legend in view, the electric pump is not operative.
- ON: ..... when pressed allows the electric pump operational condition. The ON green legend is then lit. This operation is inhibited if a flight condition is detected.

NOTE A

Once engaged, the electric pump has an operative time of 2 MIN max to prevent battery discharge. After this time, the electric pump will be automatically switched OFF.

When the electric pump is operative at 105 bar (zero flow) or at 90 bar (maximum full flow) the rated pressure is not sufficient to extinguish the cautions 1 HYD OIL PRESS, 1 SERVO and EMER LDG PRESS on the MFD and the legend PRESS of the HYD 1 TEMP/PRESS indicator on the Hydraulic System Control Panel.
However, this pressure allows the pre-flight full and free movement of the flight controls to be verified.





ICN-39-A-292000-G-A0126-01103-A-01-1

AUXILIARY POWER SYSTEM CONTROLS AND INDICATING

MAINTENANCE INFORMATION

- Before EP removal, the PCM1 must be drained.
- A bleed of the main hydraulic system and an EP operation test must be carried out after replacement.
- In case of EP replacement, a dressing procedure must be carried out.

LEADING PARTICULARS

- Electric pump flow rate..... 3 l/min (32 US quarts/min)
- Electric pump rated speed..... 10000 rpm
- Electric pump pressure:
  - full flow..... 90 bar (1305 psi)
  - zero flow..... 105 bar (1522 psi)
- Electric pump inlet operating pressure:
  - absolute..... 0.4 bar (5.8 psi)
- Electric pump step response..... 0.05 sec max
- Electric pump weight..... 3.1 Kg (6.8 lb)
- Electric pump electrical supply..... 28 VDC MAIN BUS 1

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CHAPTER
30
ICE AND RAIN PROTECTION

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SECTION 30 - PITOT AND STATIC

DESCRIPTION

To prevent ice formation the Pitot tube is provided with integral electrical heating elements controlled by the PITOT toggle-type circuit breaker powered by the 28 VDC bus bars.

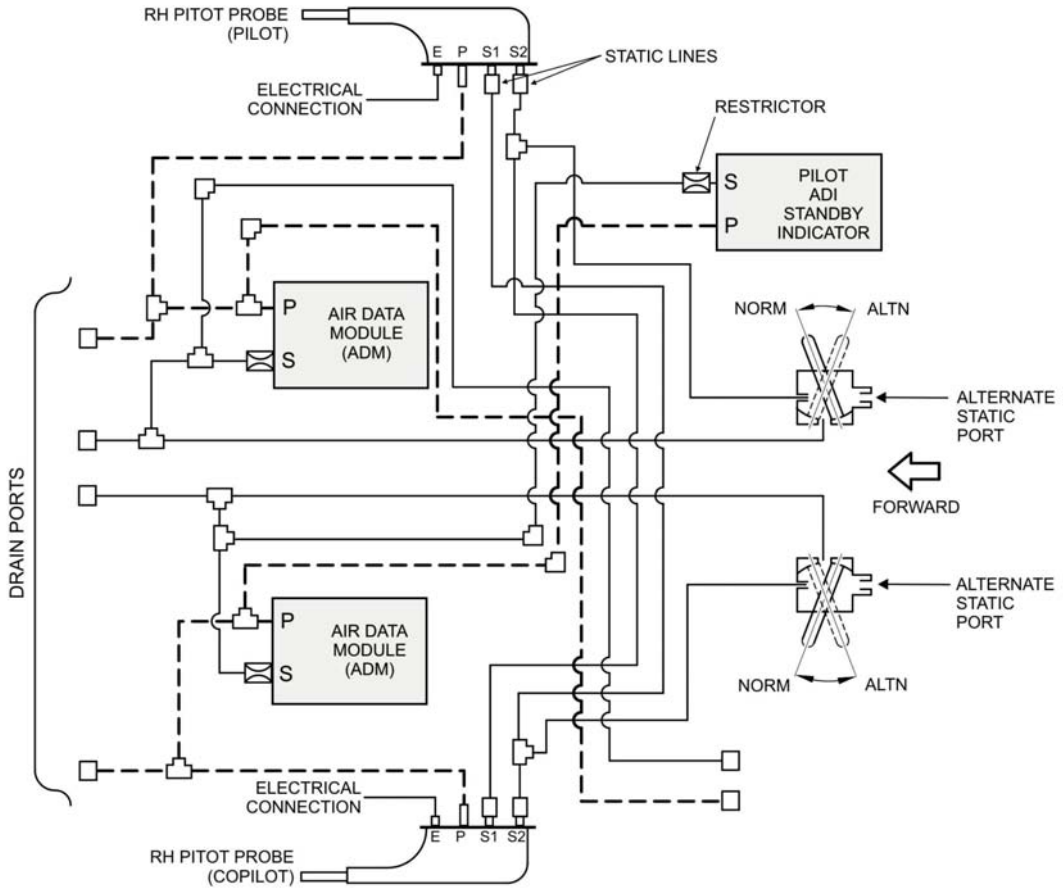
Each Pitot tube has a heating element (an internal resistor), that is installed in the rear part of the Pitot tube. A cover is its protection. Two electrical wires connect the resistor to the related current monitor and to ground.

Pilot Circuit Breaker

The circuit breaker PITOT HTR PLT (CB138) is of toggle type and is installed in the overhead console. The circuit breaker PITOT HTR PLT energizes the PITOT HEATHER switch (S72). This switch is a two-position (ON and OFF) toggle switch.

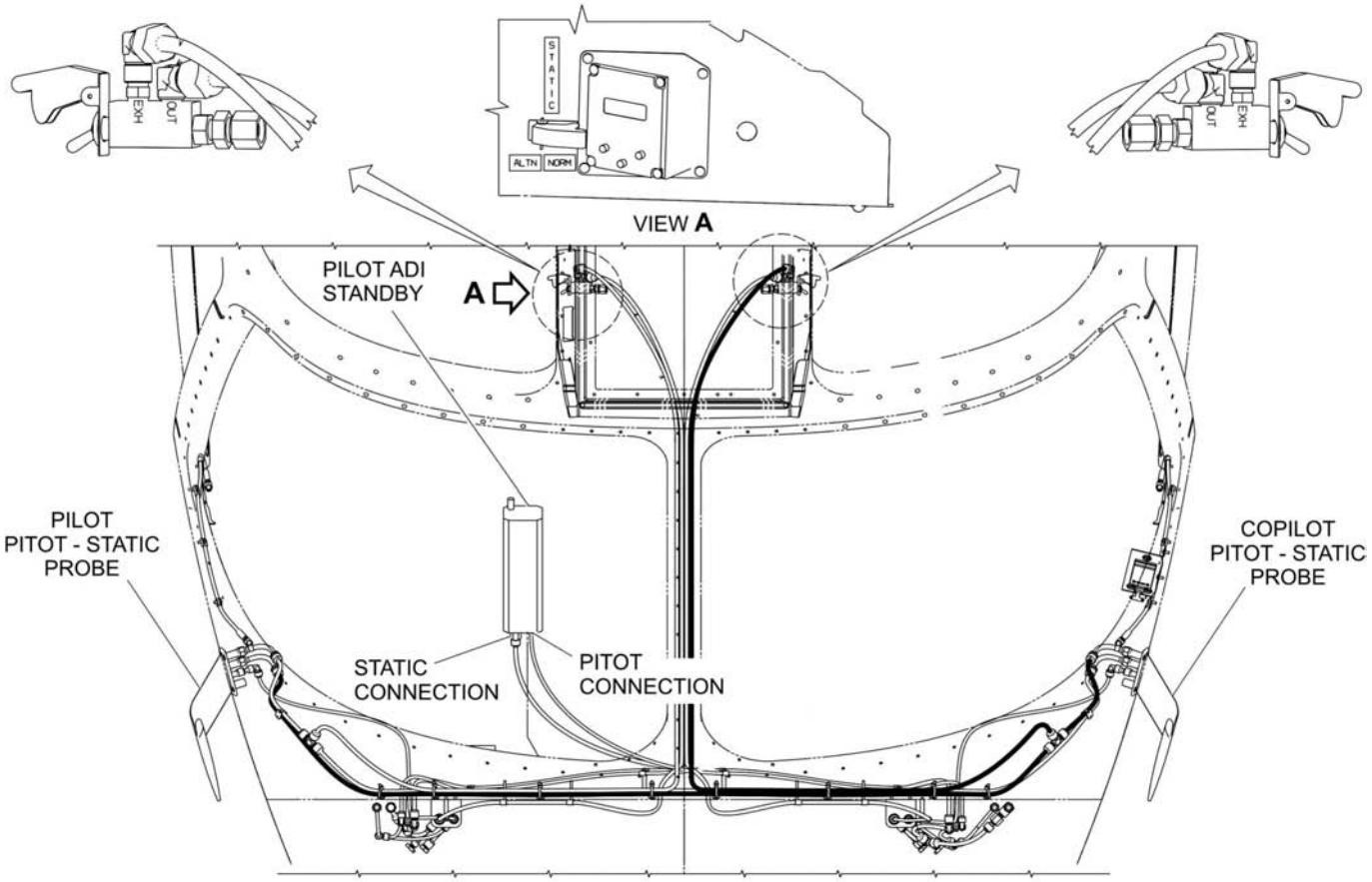
Copilot Circuit Breaker

The circuit breaker PITOT HTR CPLT (CB139) is of toggle type and is installed in the overhead console. The circuit breaker PITOT HTR CPLT energizes the PITOT HEATHER switch (S73). This switch is a two-position (ON and OFF) toggle switch.



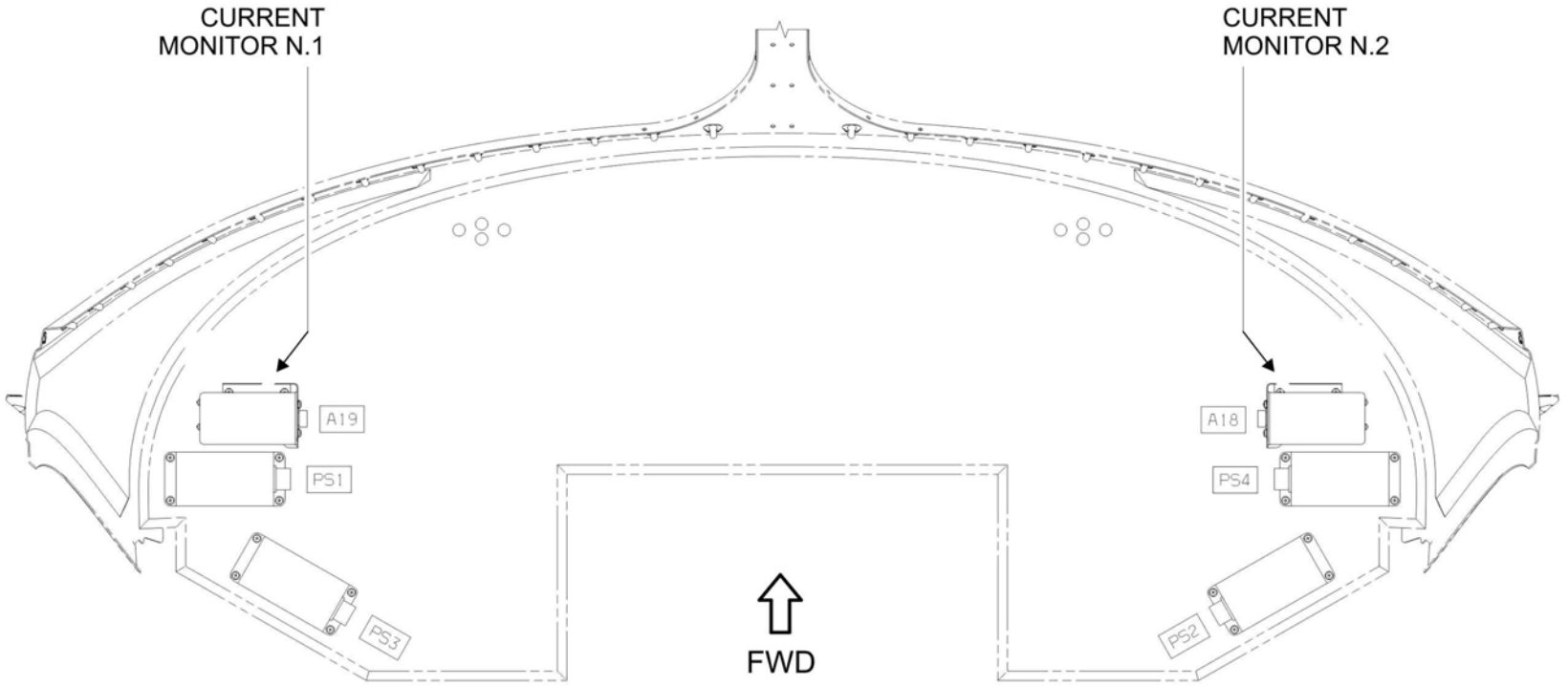
ICN-39-A-303000-G-A0126-01131-A-01-1

AIR PIPE PITOT – STATIC



ICN-39-A-303000-G-A0126-01132-A-01-1

PITOT AND STATIC GENERAL LAYOUT (SHEET 1 OF 2)



ICN-39-A-303000-G-A0126-01133-A-01-1

PITOT AND STATIC GENERAL LAYOUT (SHEET 2 OF 2)

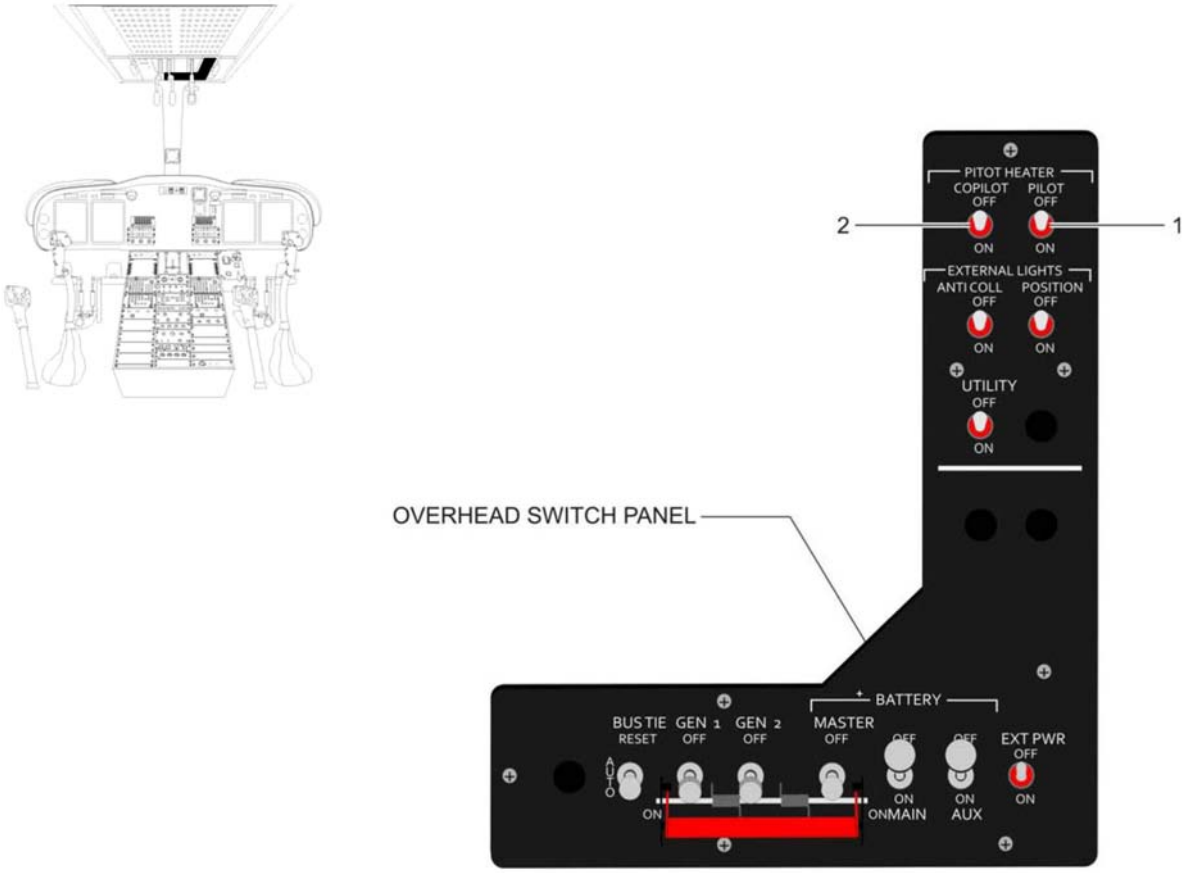
OPERATION

The Pitot tube heating system prevents the accretion of ice on the Pitot tubes.
The PITOT FAIL PLT and PITOT FAIL CPLT circuit breakers on the overhead console control the Pitot tube heating. They are also a protection of the Pitot-tube resistors from overvoltage.
The two circuit breakers can operate independently. When a circuit breaker is in the ON position, the related relay is energized with 28 V DC electrical power.
The relay K55 receives the electrical power from the Number 1 28 V DC bus, the relay K56 from the Number 2 28 V DC bus. Also, the related resistor is energized to increase the temperature of the Pitot tube, thus there is no ice accretion on the Pitot tube.
The PITOT 1 HEAT ON advisory light shows on the MFD1 when the circuit breaker CB139 is set at ON.
The PITOT 2 HEAT ON advisory light shows on the MFD2 when the circuit breaker CB138 is set at ON.
The caution message PITOT FAIL PLT or PITOT FAIL CPLT comes in view on the MAU1 and on the MAU2 when the system does not operate correctly.
The correct operation and the malfunction of the system are displayed on the MFDs.

PITOT AND STATIC CONTROLS

The pitots heater system has the control housed on the circuit breakers panel through the PITOT HEATER switches.
The operation and the malfunction are displayed on the MFD.
The circuit breaker panel provides the following function:
1(2) PITOT HEATER switch:

- OFF/ON..... allows the pilot/copilot pitot heater operational condition.



ICN-39-A-303000-G-A0126-02940-A-01-1

PITOT HEATER CONTROLS

PITOT AND STATIC INDICATING

The caution provided in the CAS message is as follows:

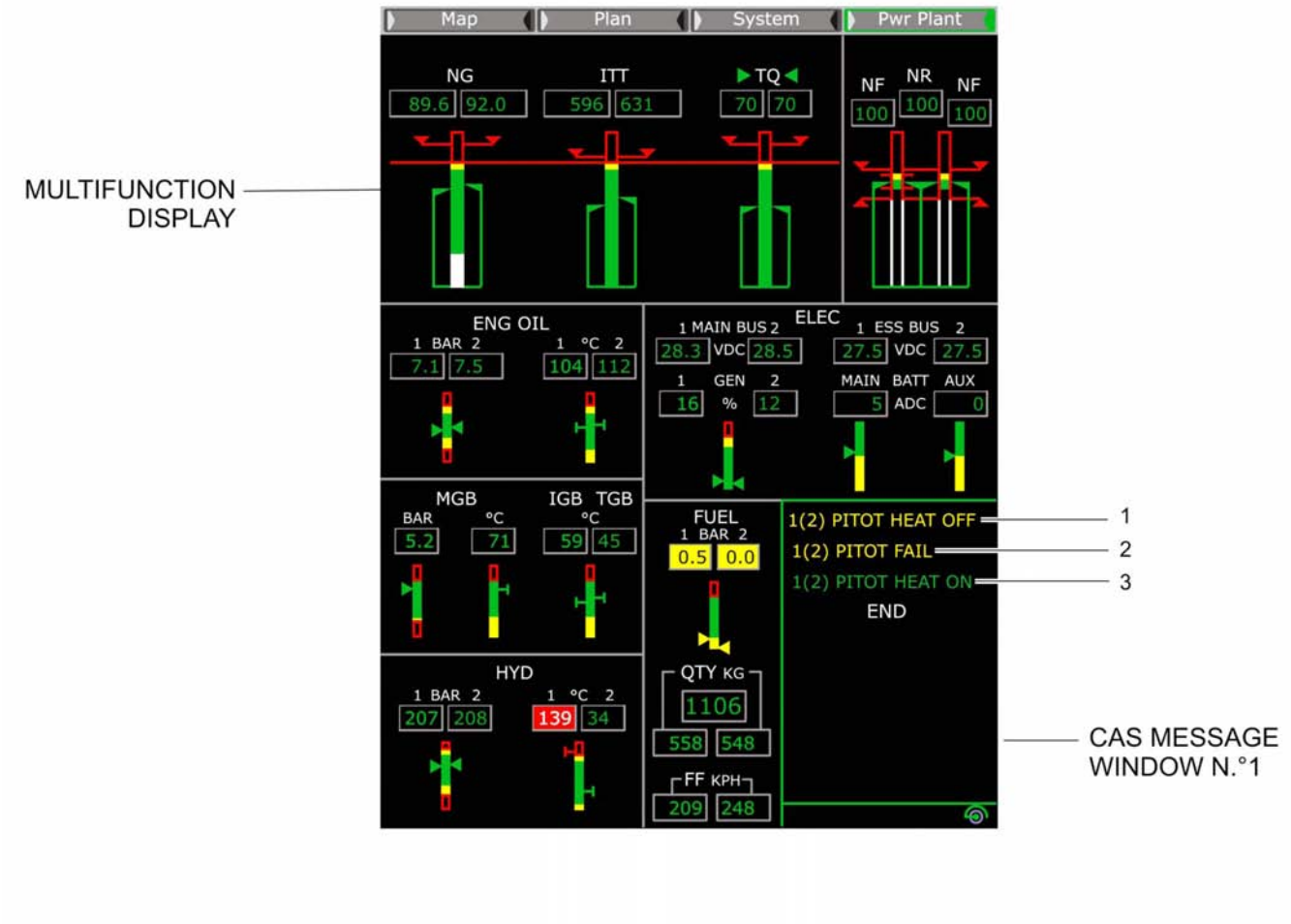
1	1(2) PITOT HEAT OFF	when the PITOT No.1 (2) is not activated with the OAT < 4°C.
2	1(2) PITOT FAIL	when the PITOT No.1 (2) is activated and the current monitor detects the pitot heat current below 0.6 A.

The advisory provided in the CAS message window is as follows:

3	1(2) PITOT HEAT ON	when the PITOT No. 1 (2) is set to ON.

NOTE A

The 1-2 PITOT HEAT ON shall be embedded in the same line message without any need to have separated messages.

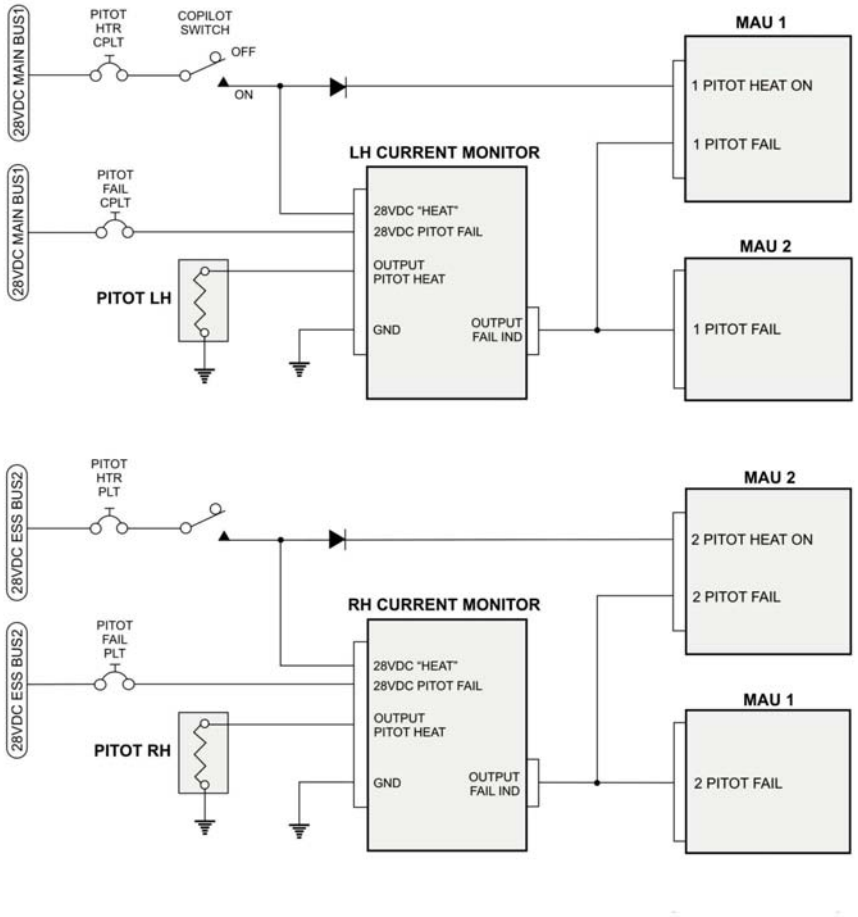


ICN-39-A-303000-G-A0126-01135-A-01-1

PITOT AND STATICS – CAPTIONS

LEADING PARTICULARS

- PILOT CURRENT MONITOR:
  - Nominal supply voltage..... 28 VDC (ESS BUS 2 + ESS BUS 2)
  - Trip current point..... < 0.6 A
- COPILOT CURRENT MONITOR:
  - Nominal supply voltage..... 28 VDC (MAIN BUS 1 + MAIN BUS 1)
  - Trip current point..... < 0.6 A



ICN-39-A-303000-G-A0126-01136-A-01-1

PITOT HEATERS ELECTRICAL SCHEMATIC

SECTION 40 - WINDOWS, WINDSHIELDS, CANOPIES AND DOORS

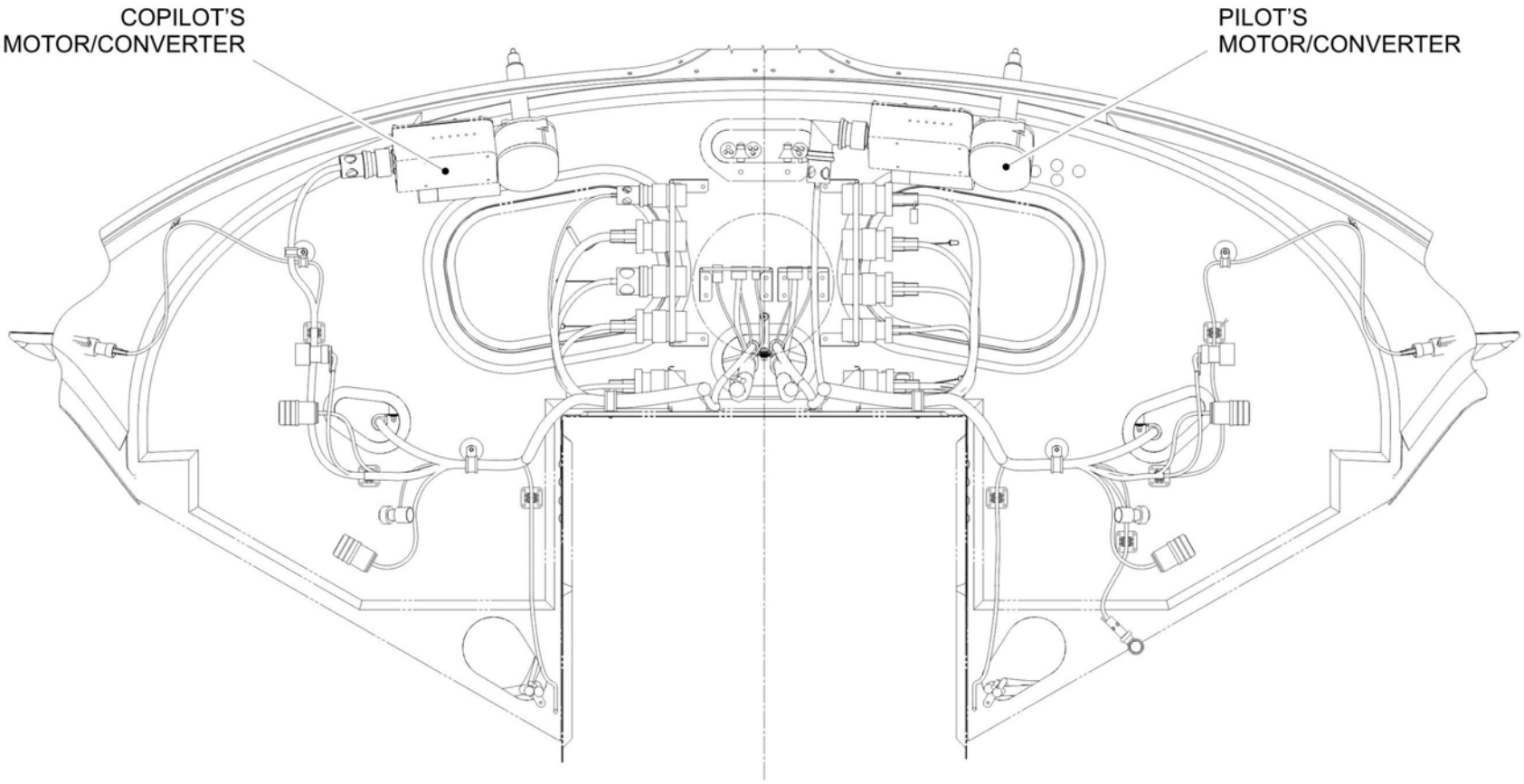
DESCRIPTION

The aircraft includes a twin windscreen Wiper/Washer system with parallel action and allows the operation of the two Pilot and Copilot Wiper blades separated or together.
The Wiper/Washer system is capable of cleaning from the windscreen surface any accumulation of water, insects, dirt, sand, dust or salt spray and a thin coat of soft snow.
The windshield wiping system consists of two identical installations, one for the pilot windshield and the other for the copilot windshield and allows the operation of the two pilot and copilot wiper blades separated or together.
The function of the windshield wiping system is to keep cleaning from the windshield surface any accumulation of water, insects, dirt, sand, dust or salt spray and a thin coat of soft snow. The system is able to function with no harmful effects derived by the flying in snow and icing conditions, and in normal condition over a helicopter speed range from –50 knots to 140 knots, and sideways flight velocity up to 50 knots.

PRIMARY COMPONENTS

The primary components of the windshield wiping system are:

- The windshield wiping control panel
- The right wiper blade
- The left wiper blade
- The right wiper arm
- The left wiper arm
- The right wiper motor converter
- The left wiper motor converter

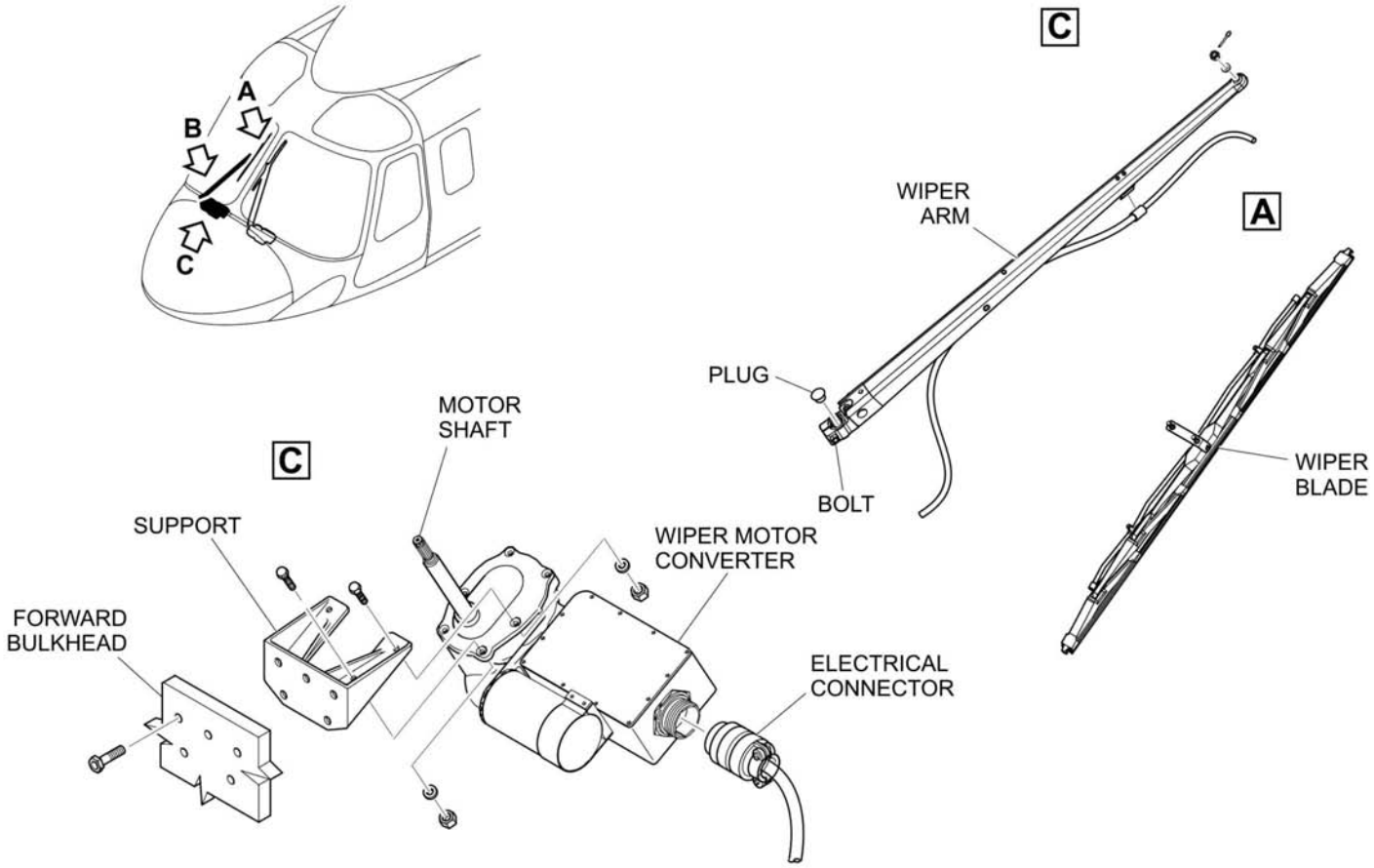


ICN-39-A-304000-G-A0126-01164-A-01-1

WINDSCREEN WIPER/WASHER SYSTEM GENERAL LAYOUT

OPERATION

The Windscreen Wiper/Washer system operational mode selection is made from a three positions rotary switch (OFF, LOW SPEED, HIGH SPEED) and by a two positions selector switch (DUAL/SINGLE).
The Windscreen Wiper sequence is controlled via a WIPER button switch located on the Cyclic stick, for both Pilot/Copilot positions.
The Windscreen Washer sequence is controlled via a push button switch located in the cockpit.
The Washing system is capable of supplying a sufficient quantity of cleaning (de-icing) liquid to both the two wiper blades dedicated spray facilities and includes a low level sensor to detect a residual liquid quantity.
The MAU1 monitors the signal and sends the maintenance message “W/S WASH LOW” to the MFDs.



ICN-39-A-304000-G-A0126-01169-A-01-1

WINDSHIELD WIPING SYSTEM – PRIMARY COMPONENTS

WINDSHIELDS WIPER CONTROLS

The windscreen wiper/washer system control panel is housed on the console.
The cyclic sticks are also involved with the operation.
The MFD displays the generic “MAINTENANCE” status message in white in the CAS area, when a residual liquid quantity in the reservoir is detected.
The windscreen wiper/washer control panel provides the following functions:

1 WINDSHIELD WASHER push-button: ..... when pressed the windscreen washer is activated. If pressed less than 1 sec, the light washing cycle is actuated: a complete low speed cycle with washing and two complete low speed cycles to clean up the windscreen faces.
If pressed more than 1 sec, the heavy washing cycle is activated: a continuous low speed cycle with washing up to command disengage and two complete low speed cycles to clean up the windscreen faces.

NOTE A

The washing mode has precedence against all the other operational modes, unless for the OFF position.
It will override either the SLOW/FAST rotary switch, the WIPER momentary switches and the SINGLE/DUAL rotary switch.
The washing cycle will engage always either the pilot and copilot systems, according to the light/heavy washing cycle.

2 SINGLE/DUAL rotary switch:

- SINGLE ..... when selected and the pilot or copilot WIPER momentary switch pressed, the corresponding system starts working.
- DUAL ..... when selected and the pilot or copilot WIPER momentary switch pressed, the systems star working.

3 OFF/SLOW/FAST rotary switch:

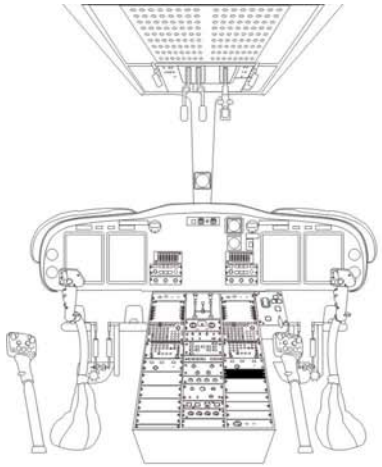
- OFF ..... when selected, the wipers are off once they reach the parking position.
- SLOW ..... the wiping sequence operates with a frequency of 45 cycles per minute.
- FAST ..... the wiping sequence operates with a frequency of 90 cycles per minute.

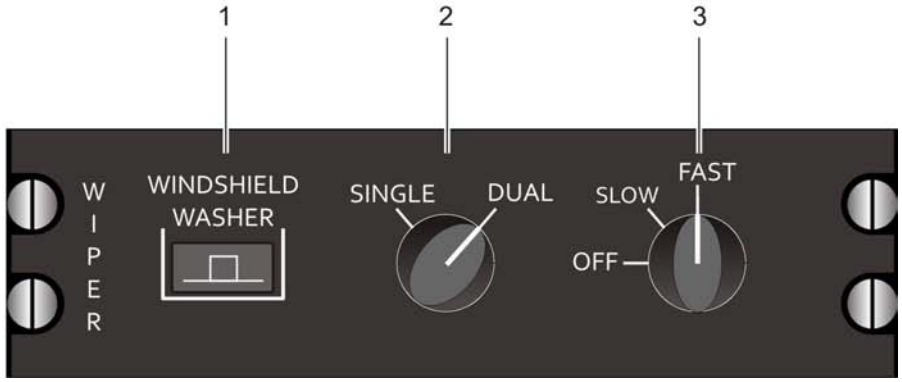
The pilot and copilot cyclic stick provide the following function:

4 WIPER momentary switch: ..... when pressed, the corresponding wiping sequence starts. To stop the wiping sequence a further action is necessary.

NOTE B

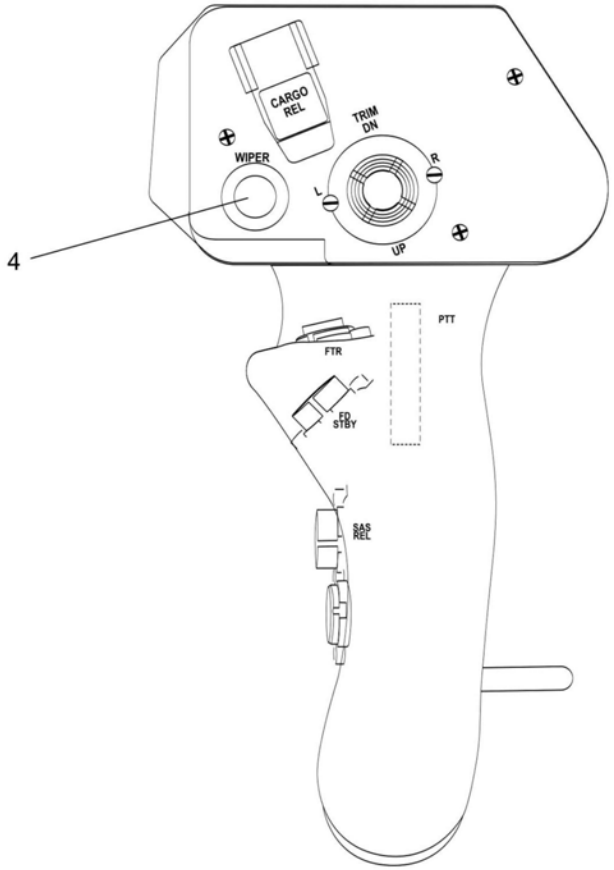
When the DUAL mode is selected, a further action on whichever momentary switch (pilot or copilot) will stop both the systems.





ICN-39-A-304000-G-A0126-01171-A-01-1

WINDSCREEN WIPER / WASHER CONTROL PANEL



ICN-39-A-304000-G-A0126-01172-A-01-1

PILOT AND COPILOT STICK GRIP

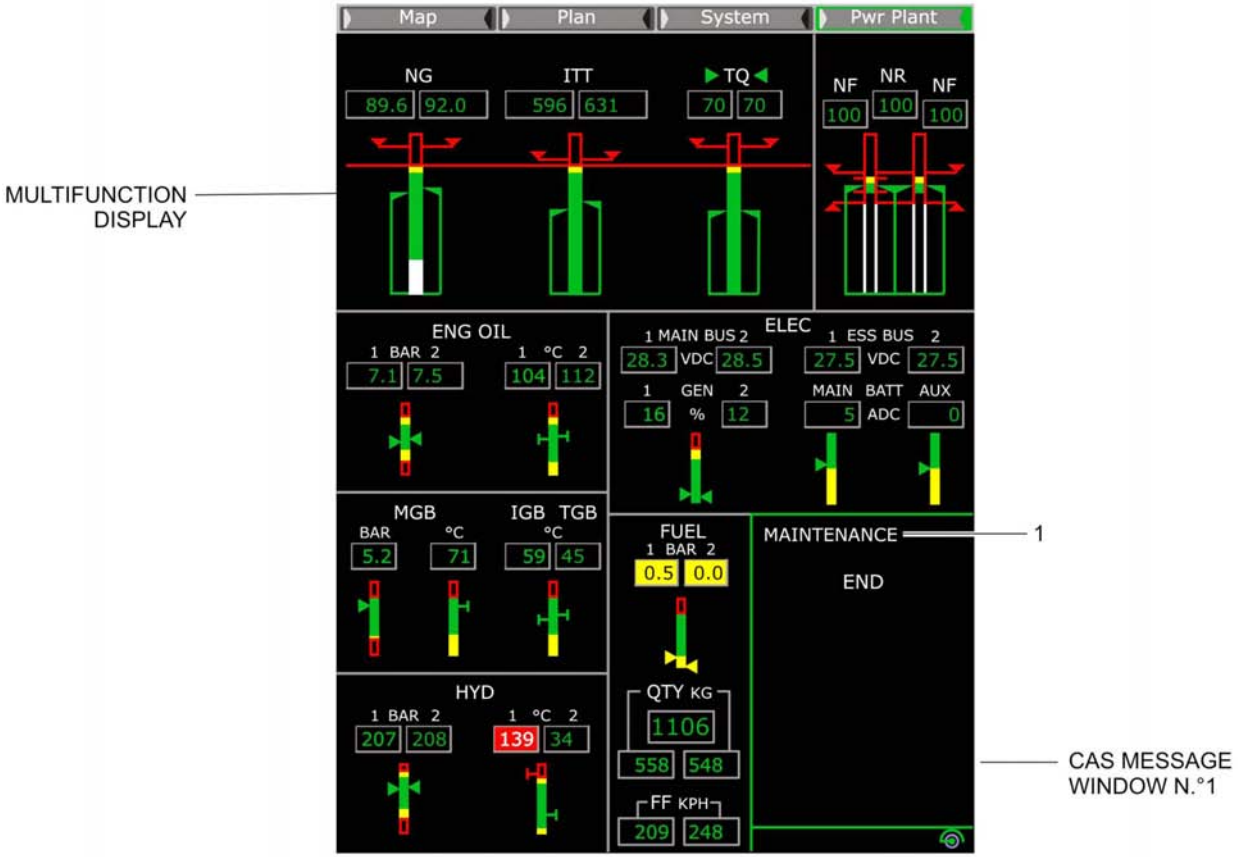
WINDSHIELDS WIPER INDICATING

The malfunctions provided in the CAS message window is as follows:

1 MAINTENANCE ..... is displayed when the liquid low level sensor inside the reservoir detects a residual quantity of 1 l nominal.

NOTE C

This MAINTENANCE caption will be displayed automatically when the A/C is on ground and both the engines off.
The dedicated message is W/S WASH LOW and is stored in the Fault Log of the Central Maintenance Computer (CMC) module visible on the MFD through the Cursor Control Device (CCD) use.



ICN-39-A-304000-G-A0126-01173-A-01-1

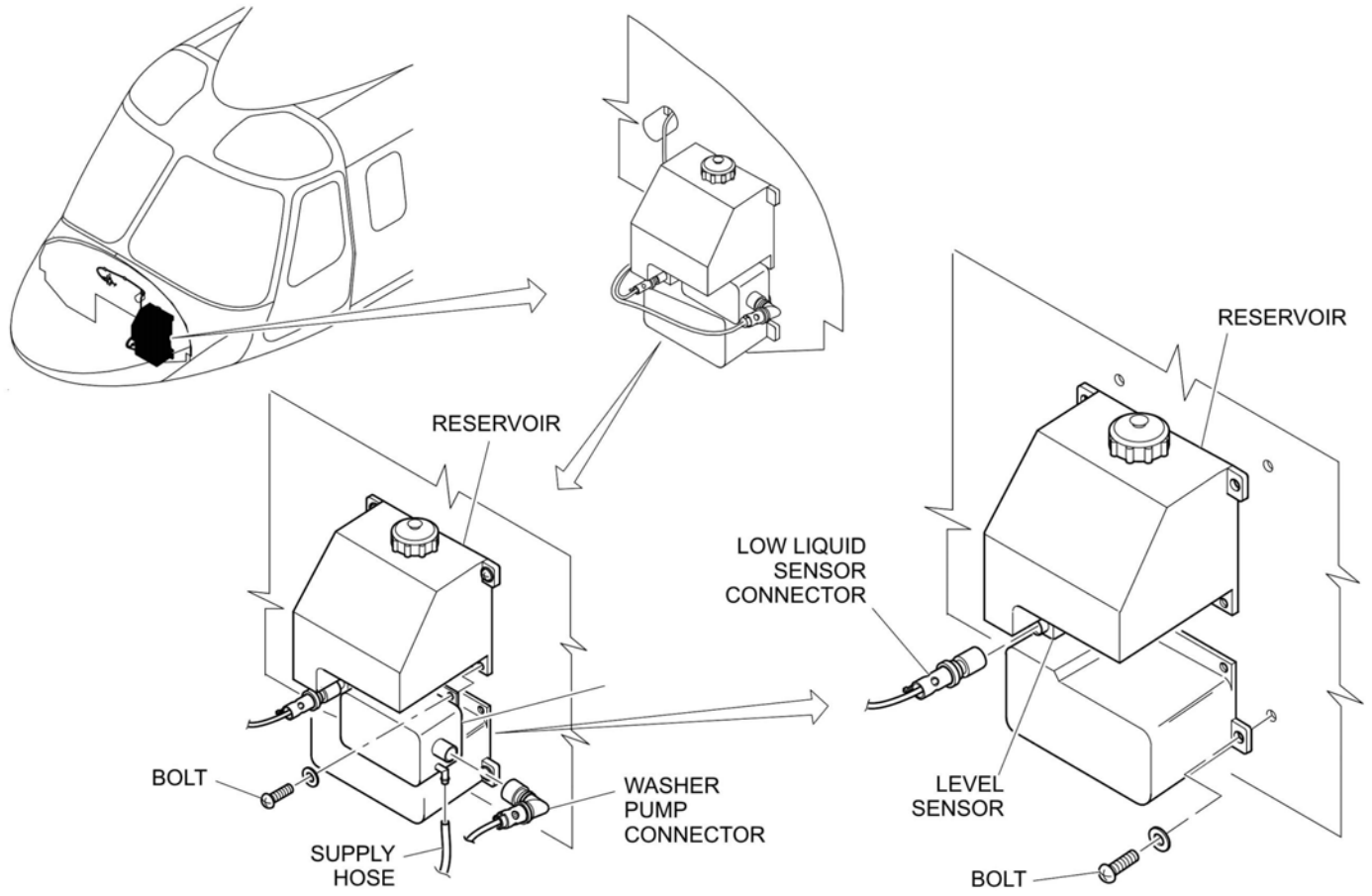
WINDSHIELDS WIPER INDICATING

MAINTENANCE INFORMATION

- Record the position of the wiper arm on the motor shaft before removal.
- The wiper motor must be moved linearly when the shaft is inside the bracket.

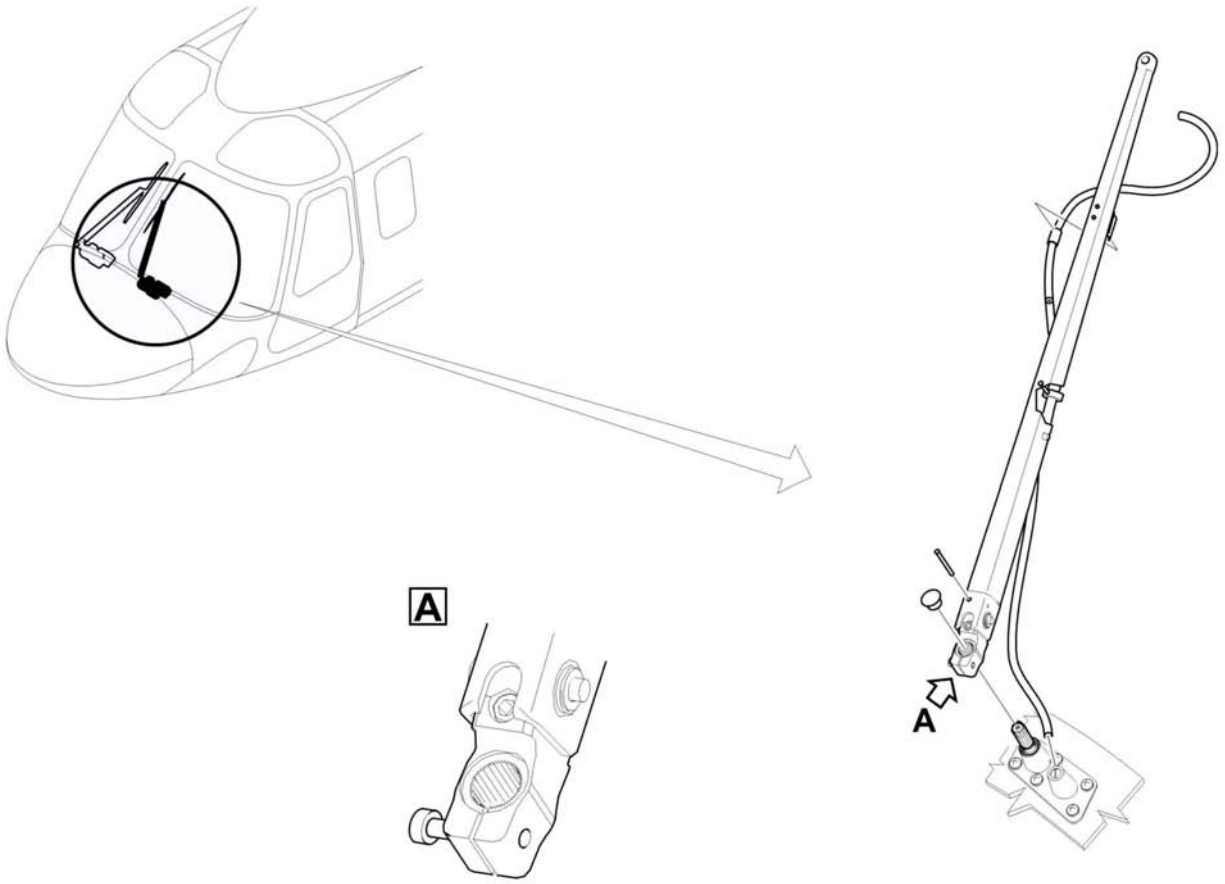
LEADING PARTICULARS

- PILOT CONVERTER:
  - Nominal supply voltage ..... 28VDC (MAIN BUS 2)
  - Converter speed - H. Speed ..... 3500 rpm
  - Converter speed - L. Speed ..... 2100 rpm
- COPILOT CONVERTER:
  - Nominal supply voltage ..... 28VDC (NON ESS BUS 1)
  - Converter speed - H. Speed ..... 3500 rpm
  - Converter speed - L. Speed ..... 2100 rpm



ICN-39-A-304000-G-A0126-01174-A-01-1

WINDSHIELD WASHING RESERVOIR AND PUMP – REMOVAL/INSTALLATION



ICN-39-A-304000-G-A0126-01175-A-01-1

LEFT WIPER ARM – REMOVE PROCEDURE

CHAPTER 31

INDICATING/RECORDING SYSTEMS

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SECTION 00 - INDICATING/RECORDING SYSTEMS

INDICATING/RECORDING SYSTEMS

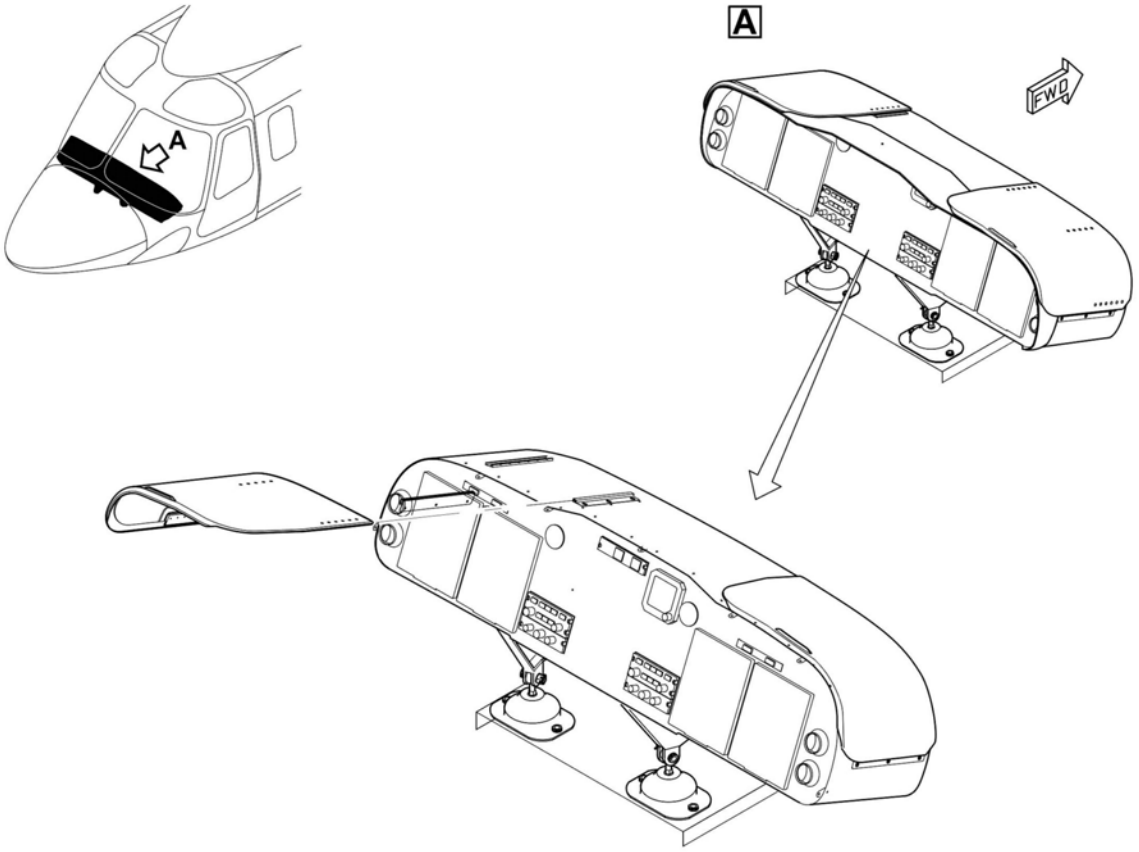
The indicating/recording systems include the subs-systems that follow:

- The instruments panel.
- The independent instruments.
- The recorders.
- The central computers.
- The central warning system.
- The central display system.

INSTRUMENTS PANEL

The instruments panel is in the cockpit in front of the pilots. On the instrument panel are installed the components that follow:

- The magnetic compass
- The master alarm panels
- The display units
- The fire detection and extinguishing control panel
- The display controllers
- The remote instrument controllers
- The area microphone
- The clocks
- The ELT remote switch panel
- The placards
- The outlet vents.



ICN-39-A-310000-G-A0126-01176-A-01-1

INSTRUMENTS PANEL

INDEPENDENT INSTRUMENTS

This section includes the components that follow:

- The clocks.
- The magnetic compass.
- The outside air thermometer.

CLOCKS

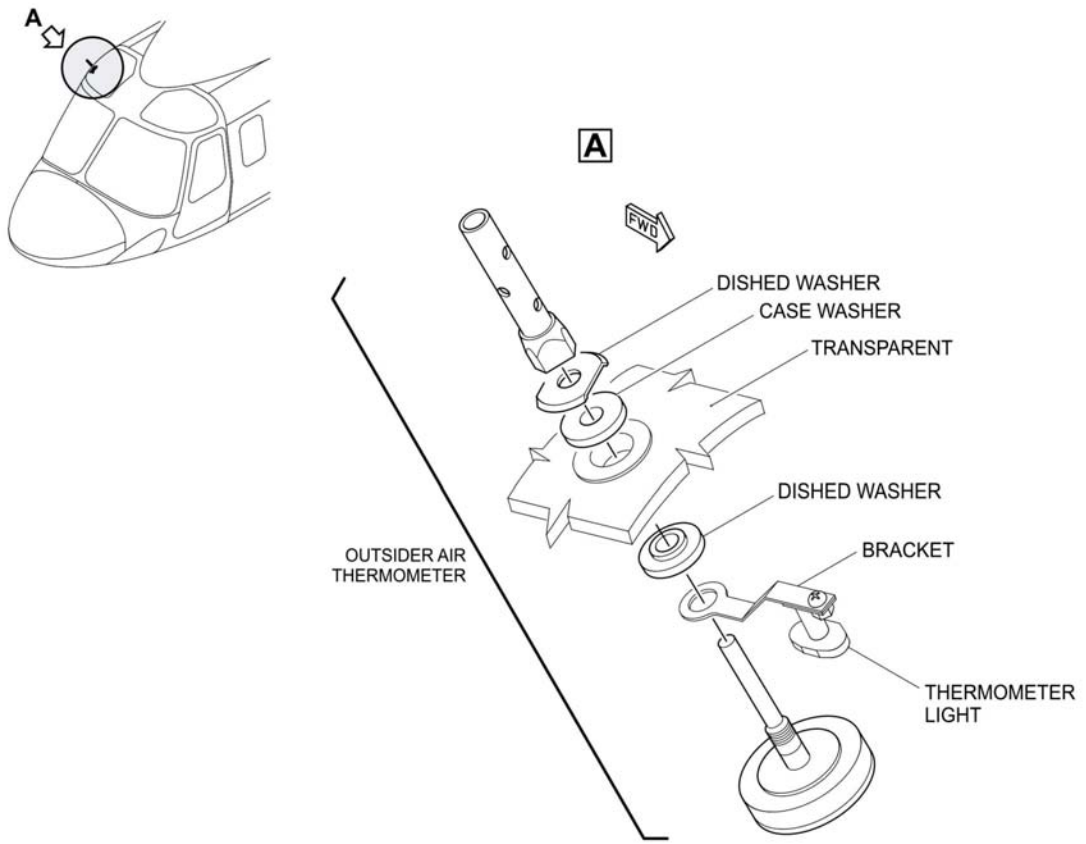
There are two clocks on the instrument panel, one for copilot and other for pilot. The clocks are chronometers that give the time and elapsed timer.

MAGNETIC COMPASS

The magnetic compass is installed on a support that is attached at the center of the glareshield on the instrument panel. The magnetic compass detects and shows the helicopter heading with reference to the magnetic North.

OUTSIDE AIR THERMOMETER

The outside air thermometer is installed on the right vision window-frame. The outside air thermometer is an aircraft instrument which measures and shows the external temperature of the air.



ICN-39-A-310000-G-A0126-01177-A-01-1

OUTSIDE THERMOMETER

RECORDERS

This section includes the systems that follow:

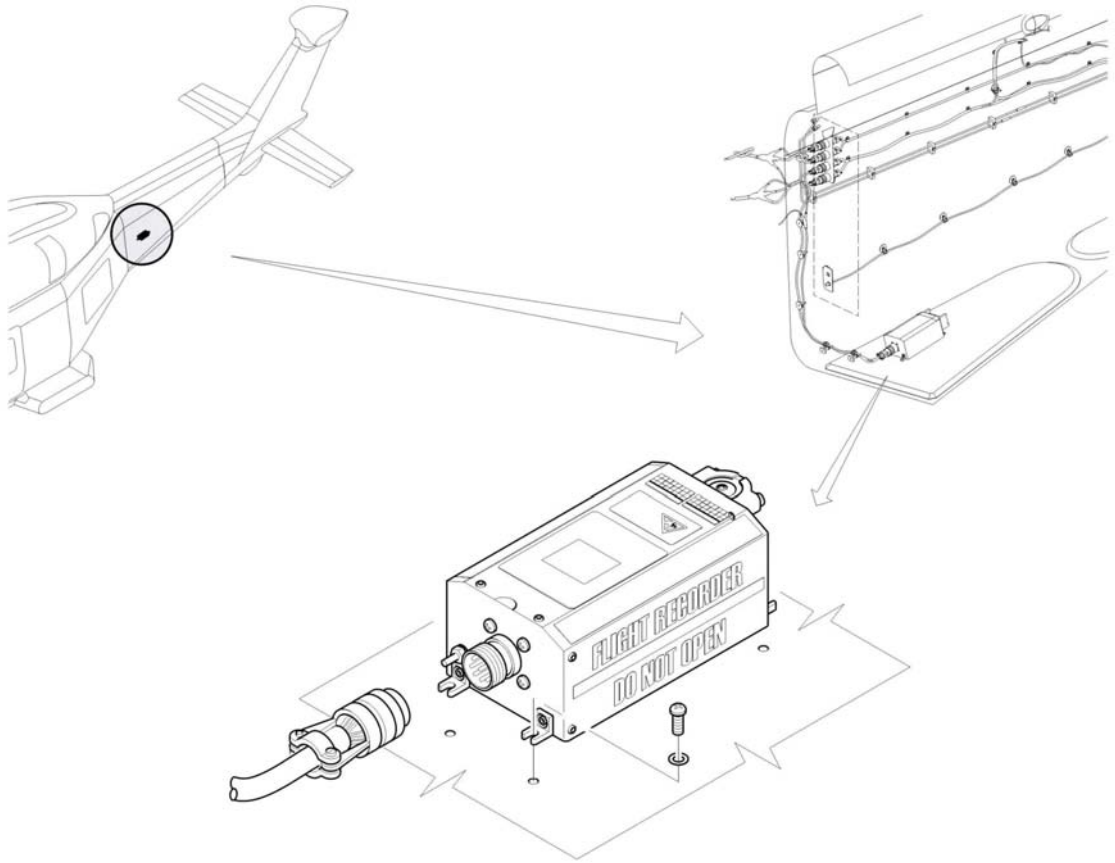
- The flight data recorder system.
- The health and usage monitoring system (HUMS) (optional).

FLIGHT DATA RECORDER SYSTEM

The flight data record system includes the Multi-Purpose Flight Data Recorder (MPFDR) The MPFDR system is composed by:

- Multi-Purpose Flight Data Recorder
- Cockpit Control Unit
- Area Microphone
- Accelerometer

The MPFDR is a crash protected airborne Multi Purpose Flight Data Recorder to meet mandatory requirements and is used to record selected aircraft parameters and 4 audio sources. The MPFDR is in the tail boom.

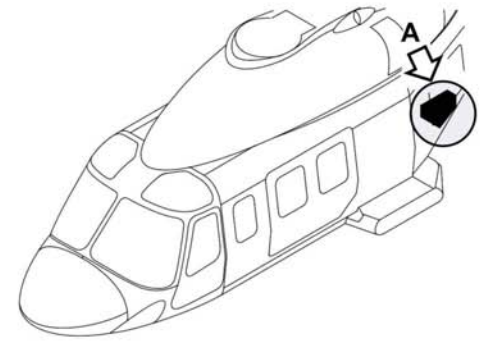


ICN-39-A-310000-G-A0126-01178-A-01-1

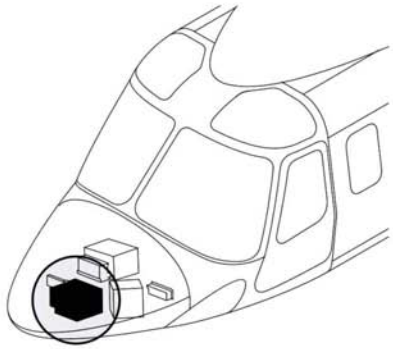
FLIGHT DATA RECORDER SYSTEM

CENTRAL COMPUTERS

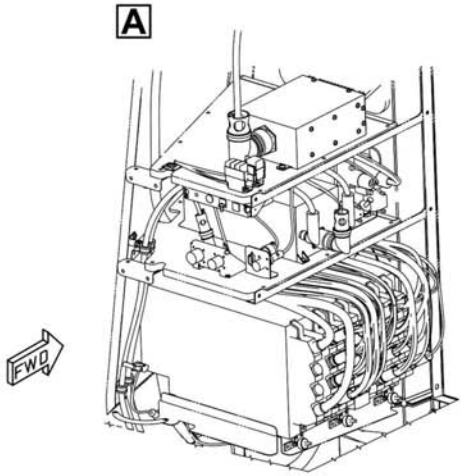
The central computer system includes two Modular Avionic units (MAU1 and MAU2). The MAUs are installed in the left and right avionic bays on the helicopters [1J]; left and right nose compartment on the helicopters [1L]. The MAUs are part of the Integrated Avionic System (PRIMUS EPIC). The PRIMUS EPIC architecture is based on the modular avionics units (MAUs).
The MAUs contain the hardware units that host the subsystems and functions necessary to operate an aircraft.

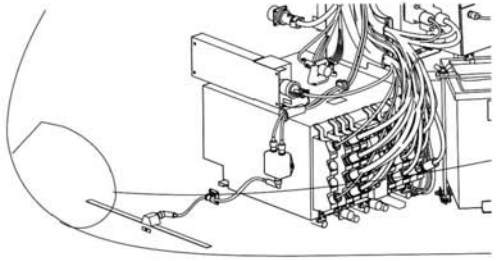


STANDARD NOSE HELICOPTER



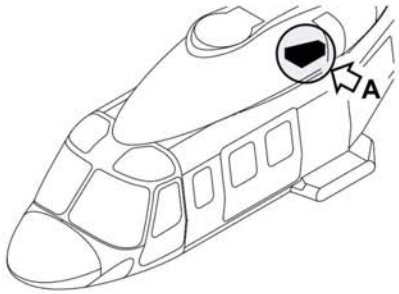
LONG NOSE HELICOPTER



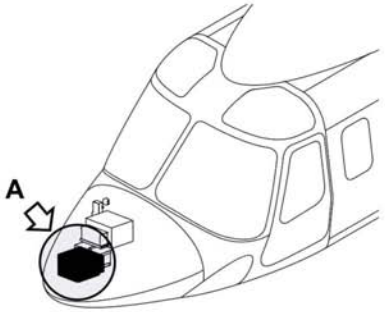


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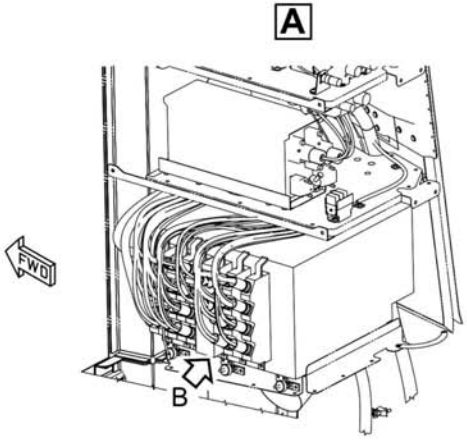
MODULAR AVIONIC UNIT (MAU) 1 INSTALLATION

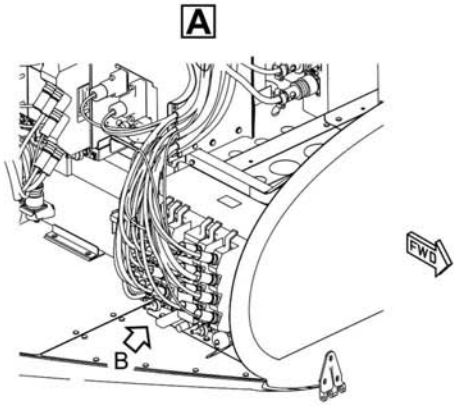


STANDARD NOSE HELICOPTER



LONG NOSE HELICOPTER



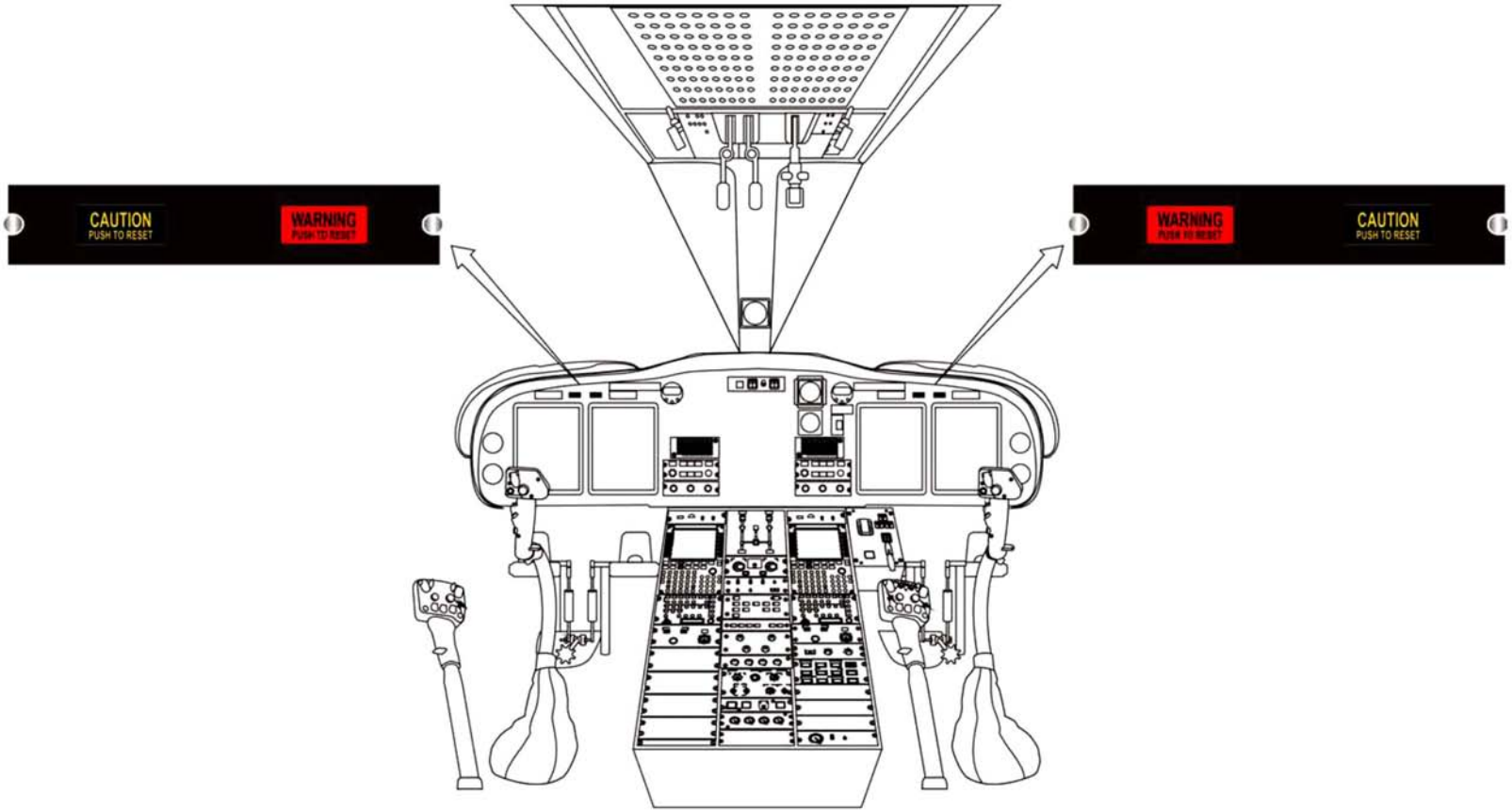


ICN-39-A-310000-G-A0126-01182-A-01-1

MODULAR AVIONIC UNIT (MAU) 2 INSTALLATION

CENTRAL WARNING SYSTEM

The Central Warning System (CWS) provides system alerts to the aircrew when unsatisfactory aircraft conditions occur. The CWS include Master Alarm Panels and Crew Alerting System (CAS) messages, visual indications and aural warning messages. The system alerts are done by the MAUs.

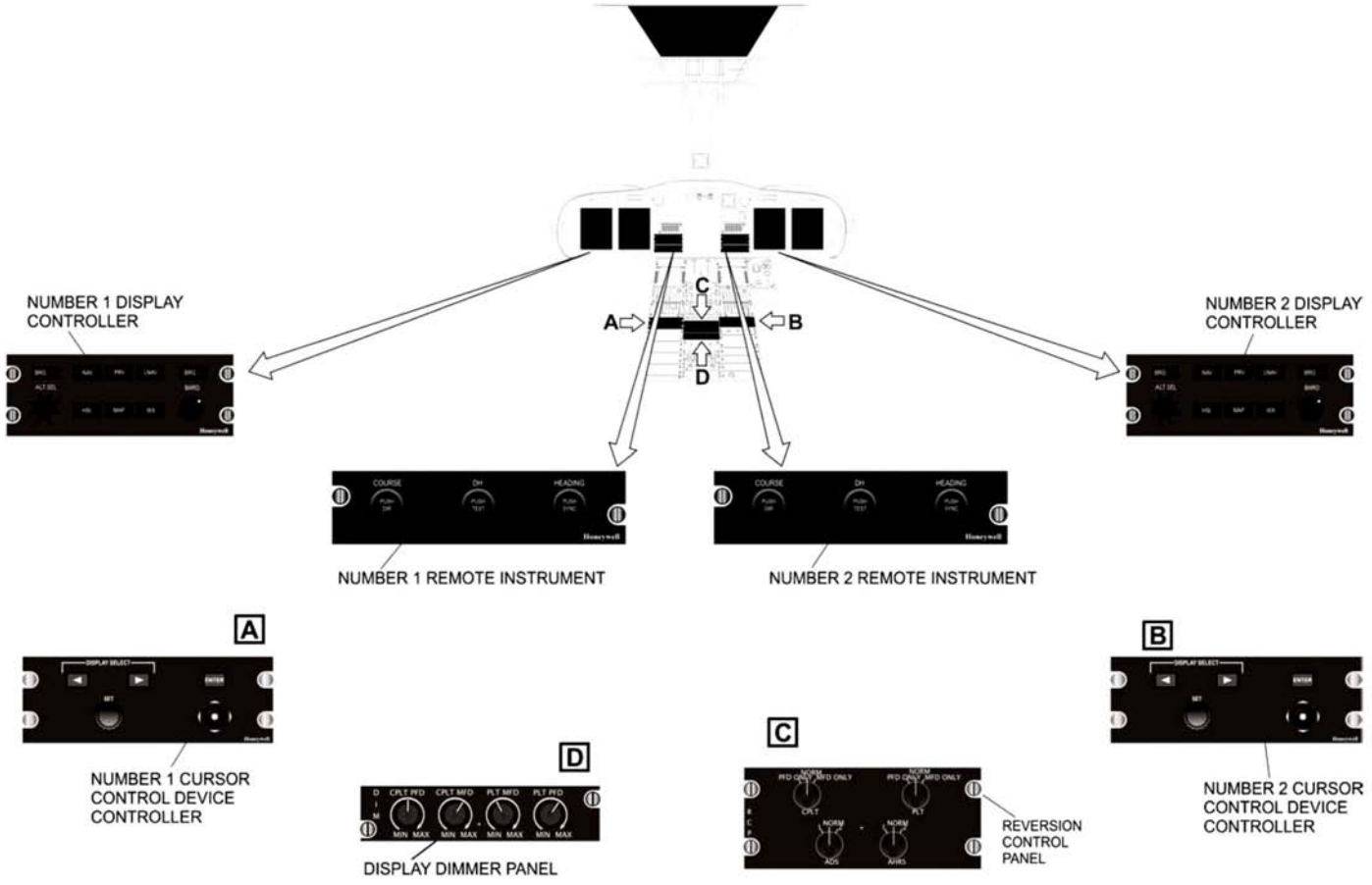


ICN-39-A-310000-G-A0126-02942-A-01-1

CENTRAL WARNING SYSTEM

CENTRAL DISPLAY SYSTEM

The Central Display System (CDS) is part of the Primus Epic® Integrated Avionics system and shows all the aircraft data to the pilots via four Display Units (DUs).Options and modes of the data displays could be selected by control panels.



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CENTRAL DISPLAY SYSTEM

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SECTION 10 - INSTRUMENT AND CONTROL PANELS

PRIMARY COMPONENTS

The instrument panel installation has the primary components that follow:

- The glareshield
- The top cover
- The instrument panel frame

GLARESHIELD

The glareshield is installed between the instrument panel frame and the top cover, at BL621.0.
Seven screws and seven washers attach the glareshield to its supports of the instruments panel frame.
The glareshield is the protection of the instrument panel from the light that goes into the cockpit thought the windscreen. Also it decreases the light reflection around the instrument panel and lets air flow on the instruments to decreases their temperature.

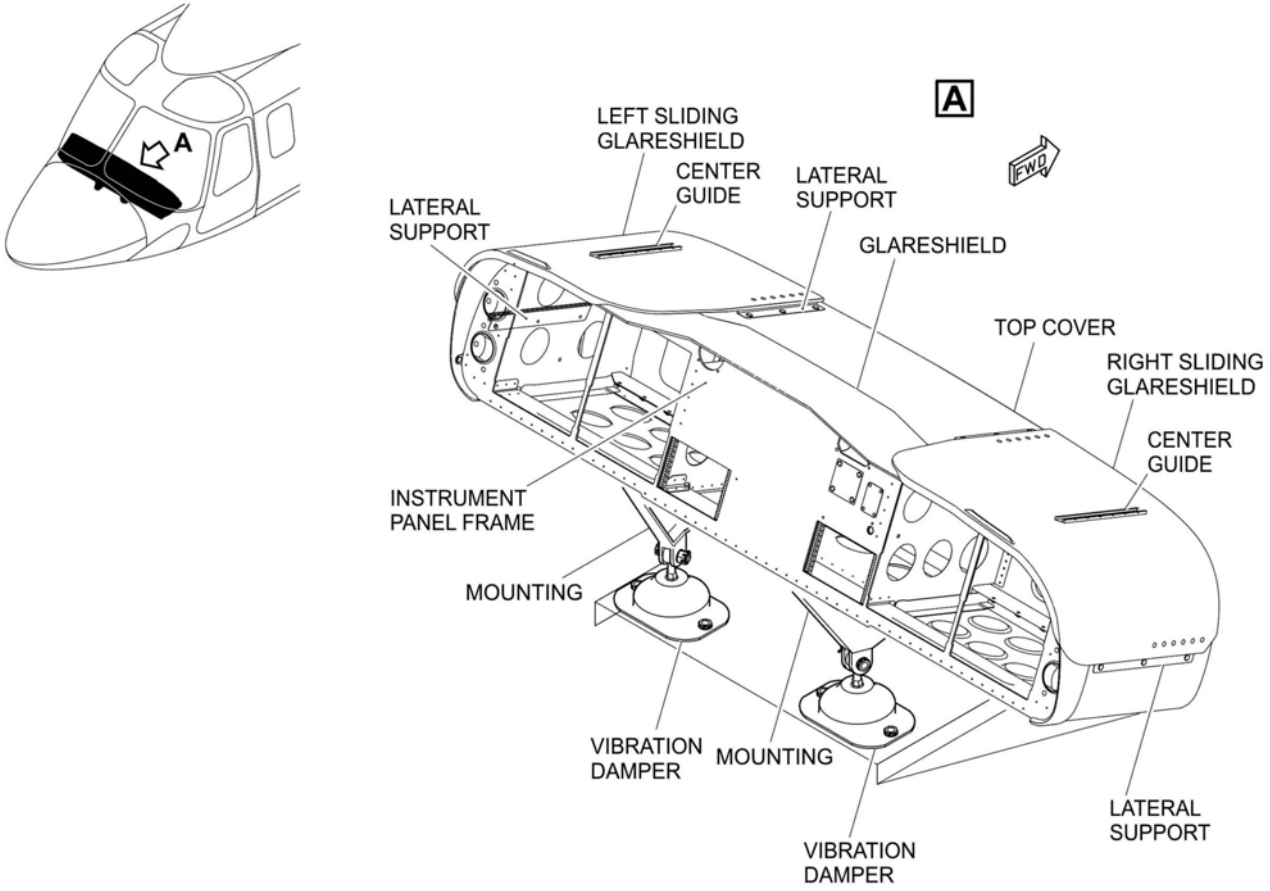
TOP COVER

The top cover is installed above the glareshield. The top cover is attached to the instrument panel as follows:

- 16 screws and 16 washers to the forward supports.
- 12 screws and 12 washers to the upper supports.
- 16 screws and 16 washers to the aft supports.

INSTRUMENT PANEL

The instrument panel is installed in the cockpit between the STA 1500.00 and STA 2065.00. The instrument panel is a frame of an aluminum-lithium alloy.
It is the frame that contains the displays, instruments panel and indicators that are in front of the flight crew.
Two mountings hold the instrument panel and attach it to two vibration dampers.
A bolt, washer and nut attach each mounting to the rod end of each vibration dampers. Each vibration damper is attached to the cockpit structure with two bolts and four washers.
The vibration dampers prevent damage from helicopter vibrations to the equipment installed in the instrument panel.



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INSTRUMENTS PANEL INSTALLATION

SECTION 20 - INDEPENDENT INSTRUMENTS

INDEPENDENT INSTRUMENTS

This section includes the maintenance data applicable to:

- The clock installation.
- The magnetic compass installation.
- The outside air thermometer installation.

PRIMARY COMPONENTS

The primary components of clock installation are:

- The Number 1 clock
- The Number 2 clock

NUMBER 1 CLOCK

The Number 1 clock (M7) is installed in the left side of the instrument panel.
The Number 1 clock has a front panel with three push-button switches, a liquid crystal display (LCD) with six digit numbers.
The Number 1 clock has these switches:

- The RST or SET push-button switch
- The MODE push-button switch
- The ST/SP or ADV push-button switch.

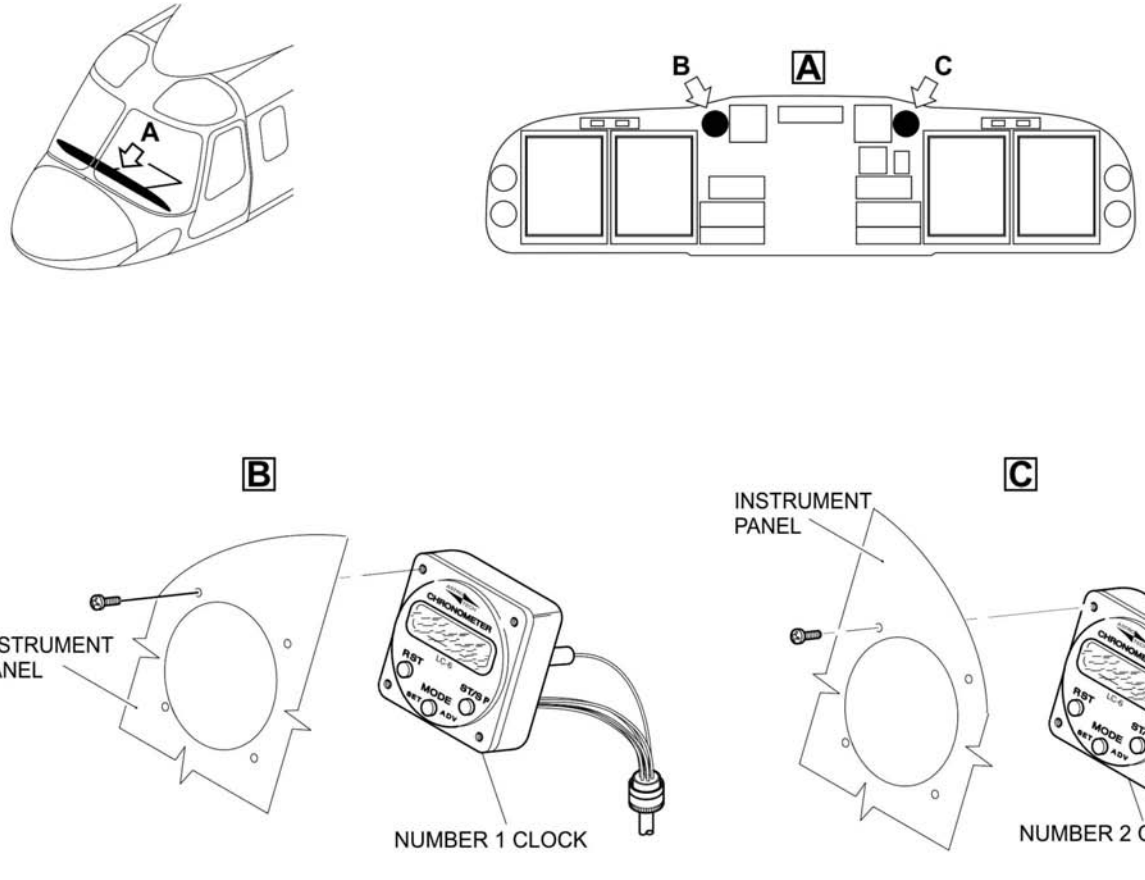
Each switch has the indication that show the related functions.
The usual power source for the Number 1 clock is from the aircraft power supply.
The Number 1 clock includes an AAA alkaline battery. The battery gives the power when the aircraft de-energized.
The Number 1 clock has back lighting for connection to the display dimmer panel. It is readable under all ambient light conditions from darkness, with internal lighting energized, to bright sunlight.
An electrical connector connect the Number 1 clock to the circuit breaker CB73.

NUMBER 2 CLOCK

The Number 2 clock (M6) is installed in the right side of the instrument panel adjacent to the pilot accessibility.
The Number 2 clock has a front panel with three push-button switches, a liquid crystal display (LCD) with six numerals and annunciators for indicating the display mode.
Four screws attach the Number 2 clock to the instrument panel.
The Number 2 clock has these switches:

- The RST or SET push-button switch
- The MODE push-button switch
- The ST/SP or ADV push-button switch.

The normal power source for the chronometer is from the aircraft power supply.
An internally mounted AAA size alkaline battery is used for keep-alive power when the aircraft power is removed.
The chronometer has back lighting for connection to the display dimmer panel. It is readable under all ambient light conditions from darkness, with internal lighting energized, to bright sunlight.
The switches have legends to indicate the function of the switch.
Four screws attach the Number 2 clock to the instrument panel. An electrical connector connect the Number 2 clock to the circuit breaker CB76.
The Number 2 clock is a chronometer that gives the time, date and elapsed timer with a hold or time-out feature.



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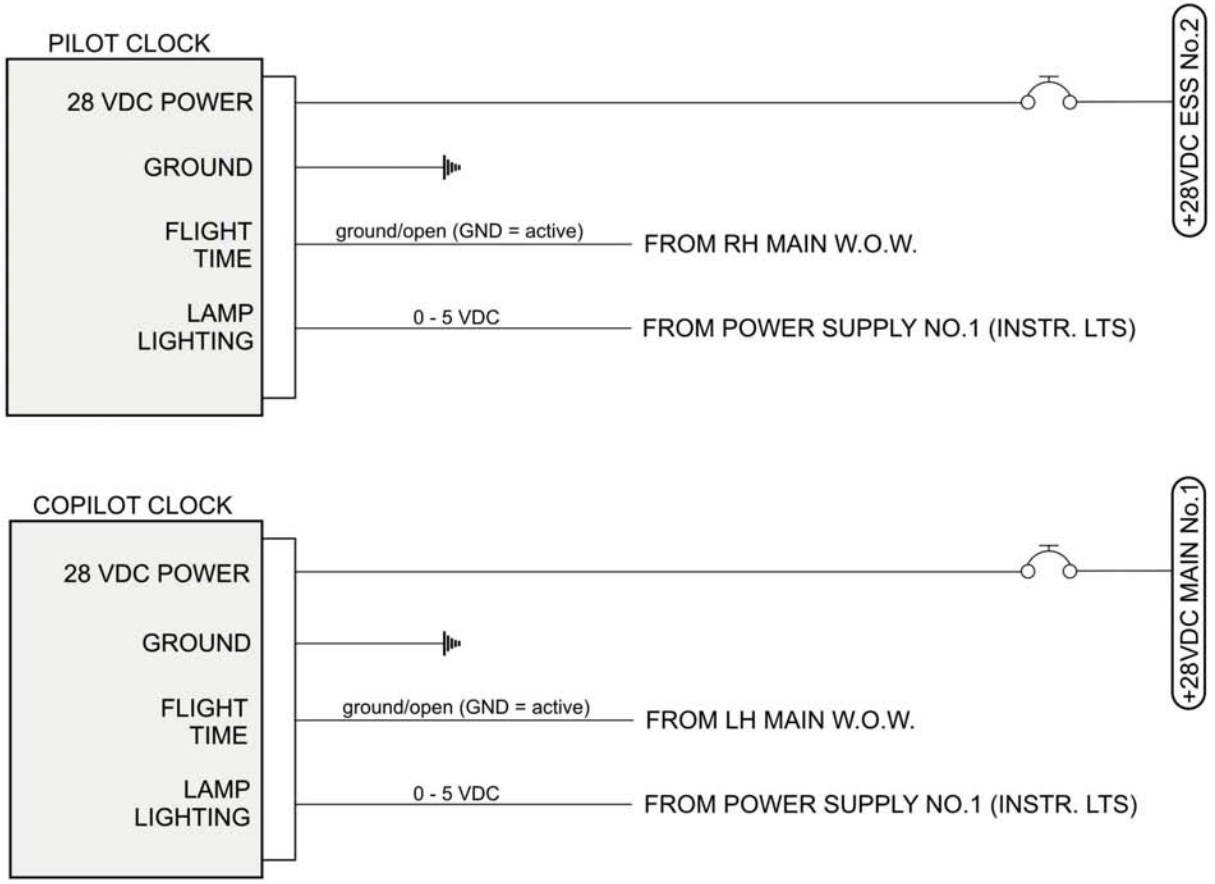
CLOCK N.1 AND N.2 INSTALLATION

SYSTEM OPERATION

The chronometer provides the following operating modes:

- Local Time (LT) of day in a 12 hour format.
- Coordinate Universal Time (UTC), formerly known as Greenwich Time, in a 24 hour format.
- Flight Time (FLT) that accumulates up to 99 hours, 59 minutes and 59 seconds; the flight time continues to run when the chronometer is in other modes.
- Stop Watch (SW) that counts up to 99 hours, 59 minutes and 59 seconds; the stop watch continues to run when the chronometer is in other modes.
- Down Counter (DC) that counts down for a maximum count of 99 hours, 59 minutes and 59 seconds; the down counter continues to run when the chronometer is in other modes.

The clock installation operates from a 28 VDC power supply. The cockpit dimming system controls the 28 VDC power supply for the clock green internal lighting.
The clock has a battery installed in it.



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SYSTEM OPERATION

CONTROLS AND INDICATORS

The chronometer is installed on the instruments panel and the controls are on the unit itself.
The controls are as follows:

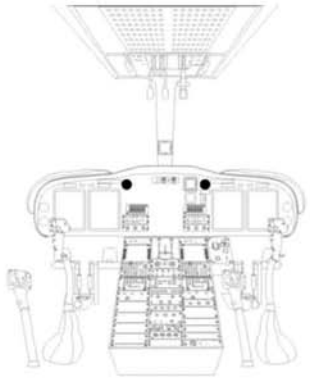
1 MODE push-button switch .... it selects the operating mode of the display and advances the annunciators across the bottom of the display. Each time the push-button is pressed the different mode is displayed (LT, UTC, FLIGHT, SW, DC).

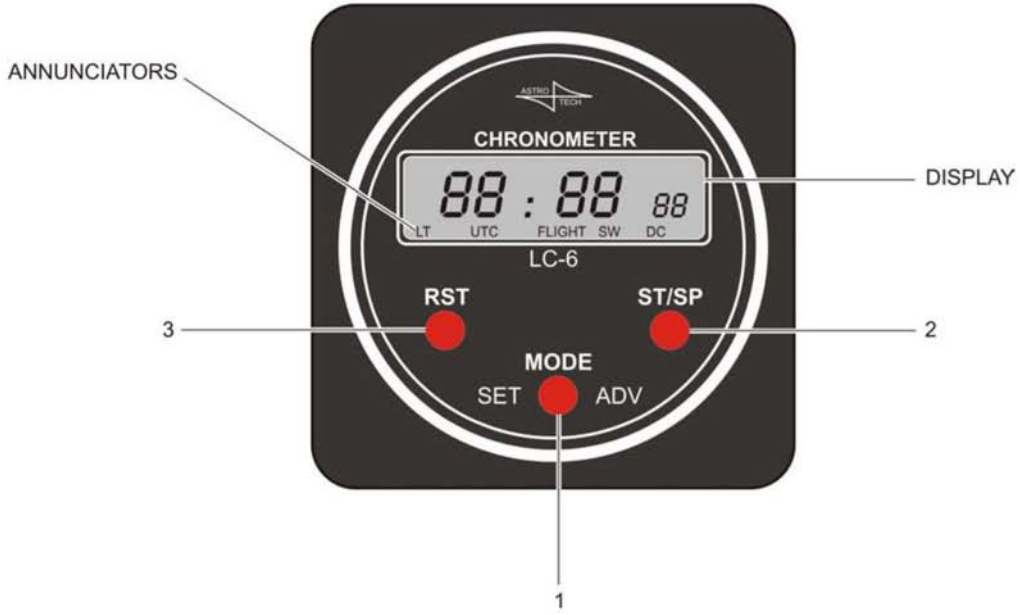
2 ST/SP - ADV push-button ..... the ST/SP push-button starts and stops the stop watch.

3 RST - SET push-button ..... pressing the RST push-button to reset to zero the stop watch.

The Mode push-button enables the following operating modes:

- LT setting
  - while in this mode, press the SET push-button switch, the hours digits are displayed and the other digits are dark, press the ADV push-button switch to advance the hours, to 12 and then go to 1.
  - Set the hours to corresponding to a time standard.
  - Press the SET switch and minutes and seconds digits are displayed and the hours digits blank, press ADV and the seconds are reset to 00 and hold, press ADV to advance the minutes, to 59 and then to 00.
  - Press the SET switch and all digits show the selected time with the seconds holding at 00.
    Start the time counting by pressing the ST/SP switch when the time standard reaches the time showing on the display.
- UTC setting
  - while in this, press the SET push-button switch, the hours digits are displayed and the other digits are blank.
  - Press the ADV to advance the hours, to 23 and then go to 00.
  - Set the hours to correspond to a time standard, press SET and minutes and seconds digits are displayed with the hours digits blank, press ADV and the seconds are reset to 00 and hold, press ADV to advance the minutes, to 59 and then go to 00.
  - SET the minutes to correspond to a time standard + 1 minute.
  - Press SET and all digits show the selected time, with the seconds holding at 00. Start the time counting by pressing the ST/SP when the time standard reaches the time showing on the display.
- FLIGHT setting
  - while in this mode, the flight timer is not running.
    The flight timer runs when the W. OFF W. condition is detected.
  - The flight timer stops when the W ON W condition is detected.
  - Press ST/SP and the flight timer is reset to zero.

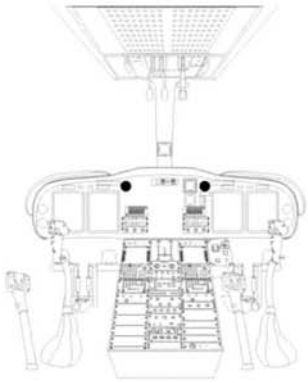


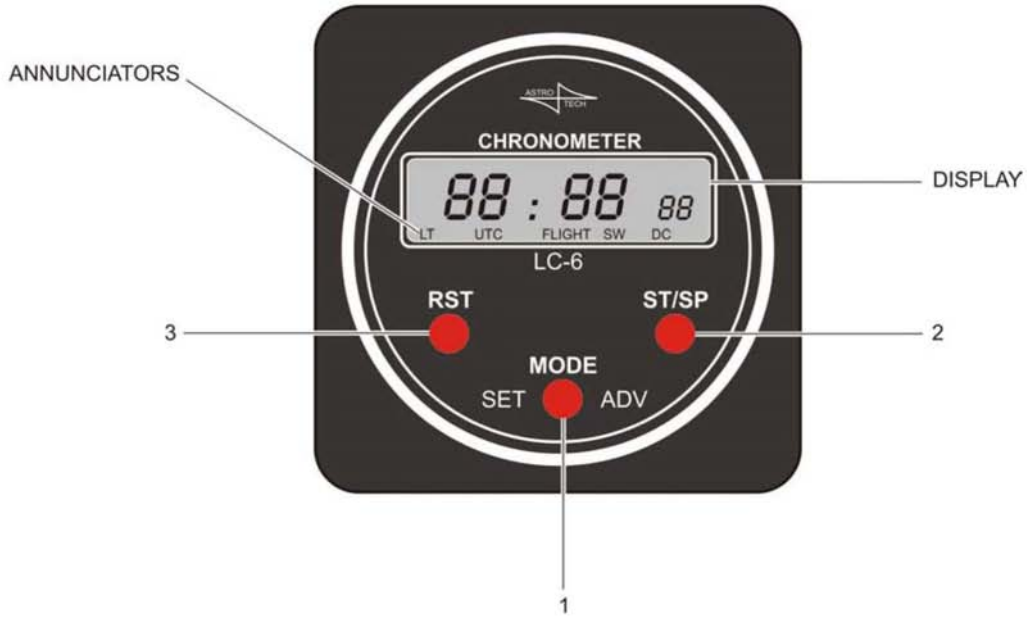


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CHRONOMETERS CONTROLS AND INDICATORS (SHEET 1 OF 2)

- STOP WATCH setting
  - while in this mode, the ST/SP push-button starts and stops the stop watch. With the stop watch not running, press RST and the stop watch is reset to zero.
  - Press ST/SP to start the stop watch.
  - If the RST is pressed, the stop watch is reset to zero and then continues to count.
- DOWN COUNTER setting
  - while in this mode, the ST/SP starts and stops the counter.
  - Press SET and the display shows 0:00:00.
  - Press SET a second time to enter the DC set condition. The hours digits are displayed and the other digits blank.
  - Press ADV to advance the hours, to 99 and then go to 0.
  - Press SET and the minutes digits are displayed with the other digits blank.
  - Press ADV and hold to advance the minutes, to 59 and then go to 00.
  - Press SET and the seconds digits are displayed with the other digits blank.
  - Press ADV and hold to advance the seconds, to 59 and then go to 00.
  - Press SET and all digits show the selected time.
  - Press ST/SP and the down-counter counts down. When the display reaches 0:00:00, it flashes every ½ second for one minute and the counter begins to count up until terminated.





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CHRONOMETERS CONTROLS AND INDICATORS (SHEET 2 OF 2)

CAPTIONS

Cautions and advisories are not involved with the chronometers.

MAINTENANCE OPERATION

The internally mounted AAA size alkaline battery should be replaced every 24 months. To replace the battery, remove the chronometer from the instrument panel first.

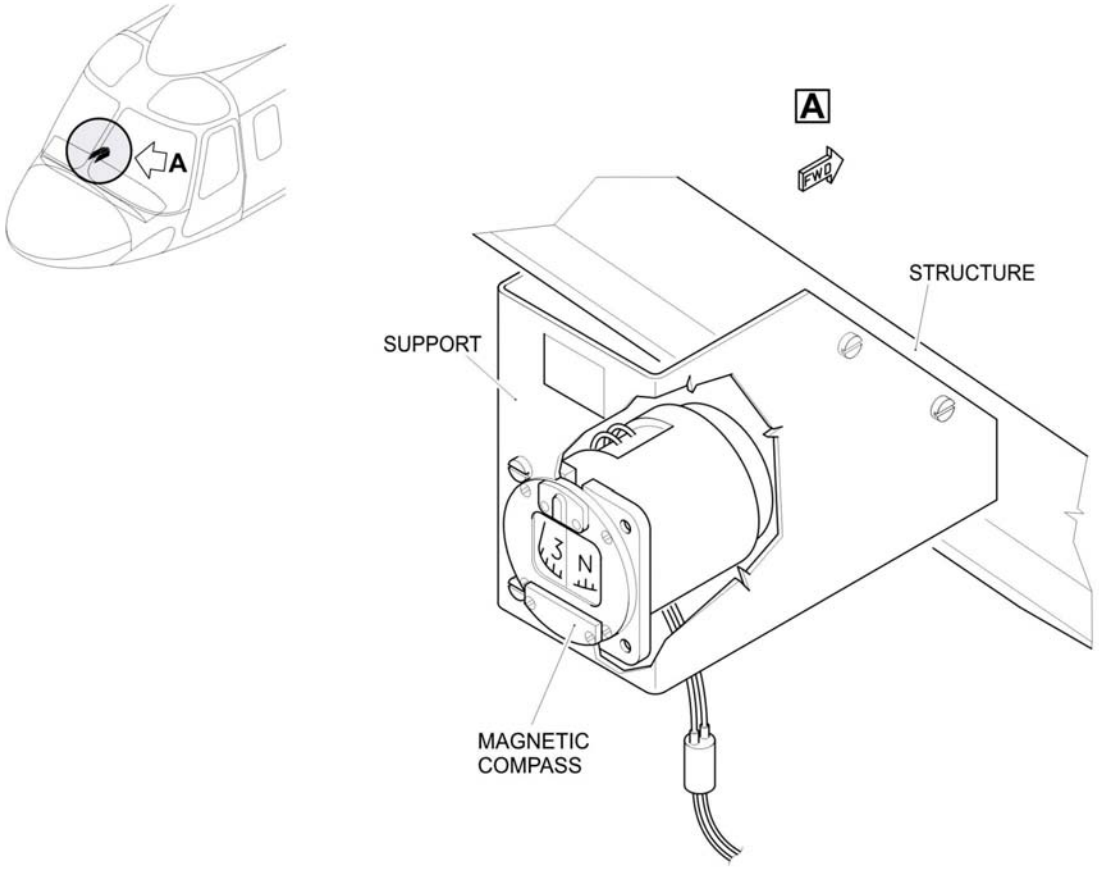
LEADING PARTICULARS

- Pilot chronometer power supply..... 28VDC ESS BUS No.2
- Time keeping accuracy..... better than one second per day under stabilized temperature conditions
- Operating temperature..... -15°C to 70°C. The low operating temperature is -40°C
- Copilot chronometer power supply..... 28VDC MAIN BUS No.1
- Timekeeping accuracy..... better than one second per day under stabilized temperature conditions
- Operating temperature..... -15°C to 70°C. The low operating temperature is -40°C

MAGNETIC COMPASS INSTALLATION

The primary component of the magnetic compass installation is the magnetic compass M3.

The magnetic compass is installed on a support that is attached at the center of the glareshield with four screws. Two electrical cables connect the magnetic compass to the display dimmer panel.
The standard magnetic compass shows the helicopter heading with reference to the magnetic North. The compass consists of a rotating compass card. Two magnetic bars attached to the compass card make the compensation system. The compass card and the magnetic bars are contained in a sealed case filled with damping fluid. The compass card has lines at 5-degree intervals and shows letters and numbers at 30-degree intervals. A lubber line on the transparent window shows the helicopter heading.
The flight crew can remove the position errors of the magnetic bars through a screw that is installed on the external side of the instrument. The compass has a backlighting system that is compatible with NVG's.
The INSTR knob on the display dimmer panel controls the lighting of the magnetic compass.



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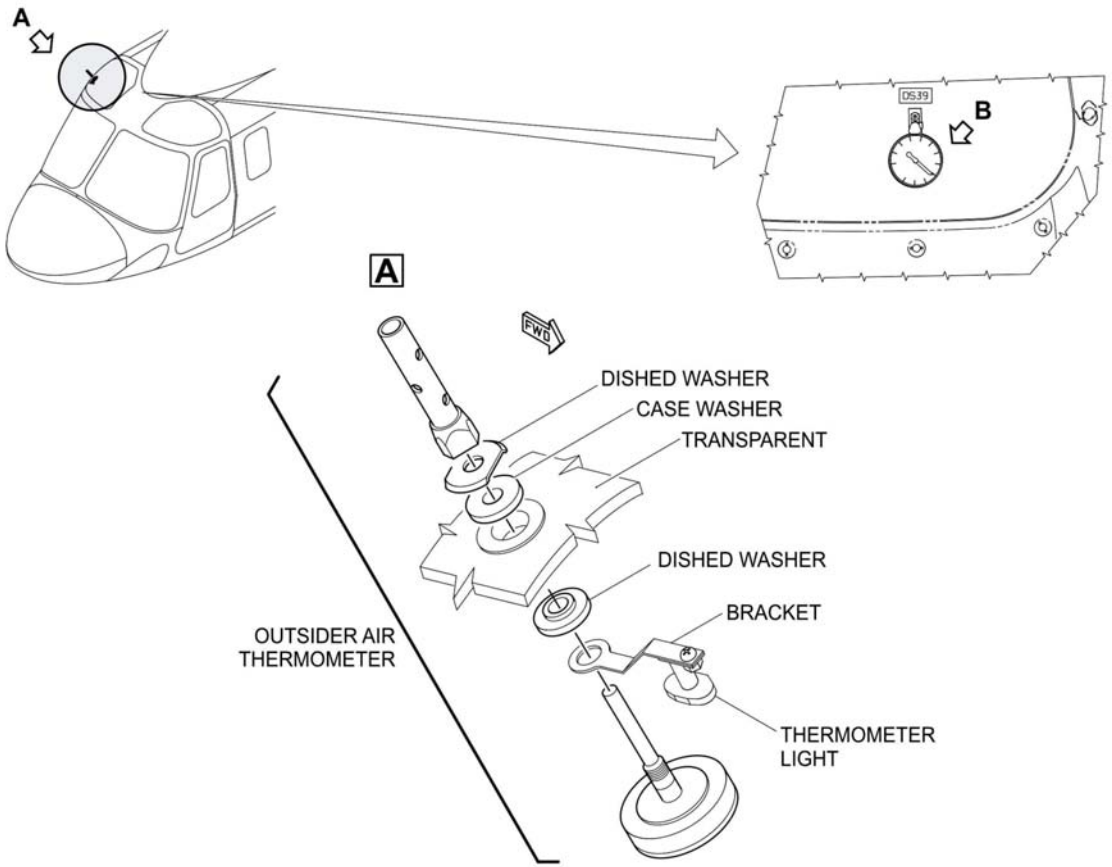
MAGNETIC COMPASS INSTALLATION

OUTSIDE AIR THERMOMETER INSTALLATION

The outside air thermometer is an aircraft instrument which measures and shows the external temperature of the air. The primary component of the outside air thermometer installation is the Outside Air Thermometer (OAT).
The OAT is installed in the top of the right direct-vision window-frame at STA2855. It includes the parts that follow:

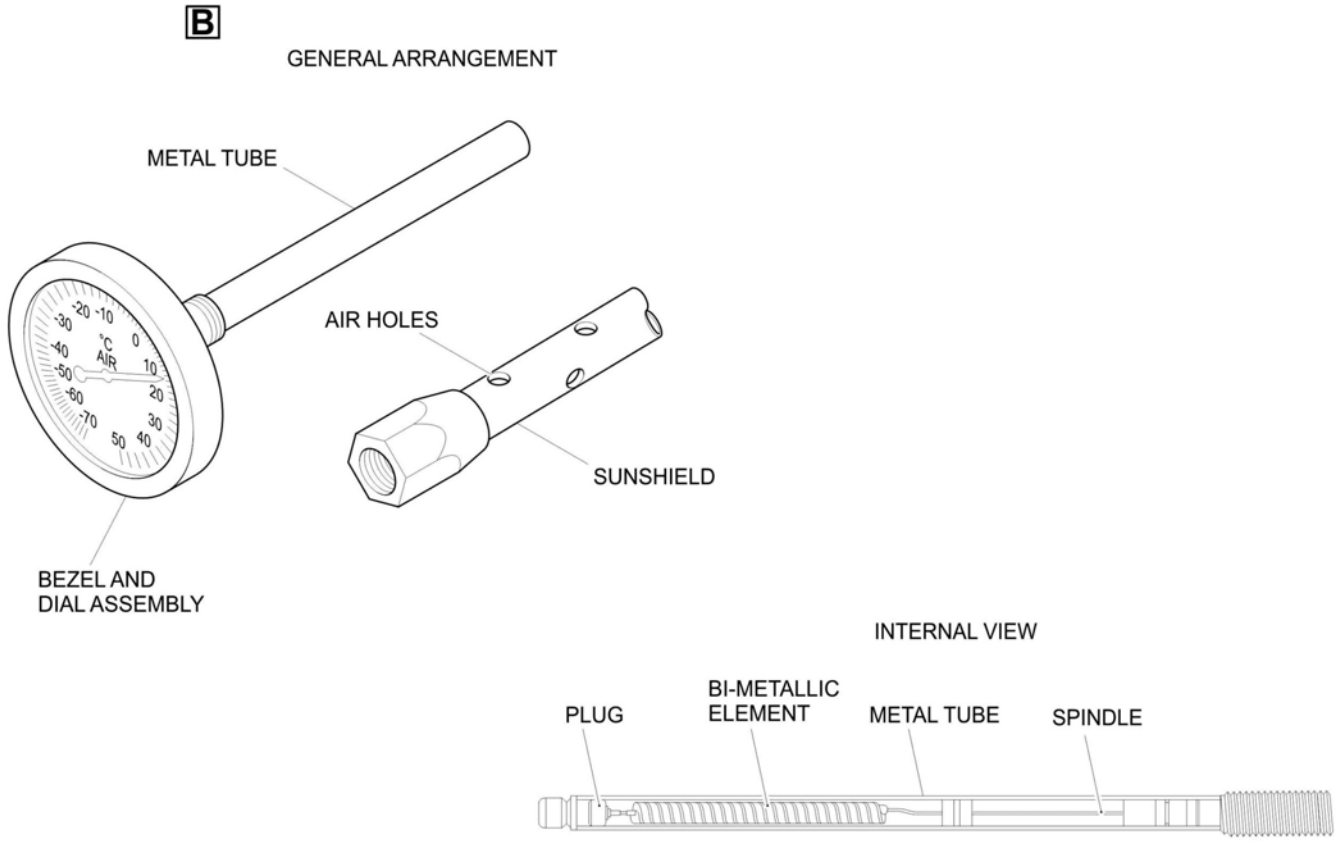
- The sunshield.
- The dished washers.
- The bracket.
- The thermometer light.
- The bezel and dial assembly.

The OAT has a temperature range of -70˚C thru +50˚C. Its dial has luminous graduations of 10˚C measured by means of a bimetallic probe which protrudes through the right overhead window.
The OAT is installed on the metal tube that is installed on the right vision window-frame. The OAT bracket is attached to the vision window-frame. The bracket holds the OAT light. A sunshield is installed on the shaft to give protection to the metal tube. The holes on the sunshield let the air flow freely around the metal tube.
The OAT has a bi-metallic element installed in a metal tube.
This element contains two strips of different metals made into a helix and welded together. One end of the helix is attached to a plug.
The other end of the helix is attached to a spindle with a pointer.
The changes of the external temperature cause a turn movement of the free end of the bi-metal element.
The OAT has a thermometer light (DS39) controlled by a pushbutton switch adjacent to the light.



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OUTSIDE THERMOMETER INSTALLATION (SHEET 1 OF 2)



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OUTSIDE THERMOMETER INSTALLATION (SHEET 2 OF 2)

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SECTION 30 - RECORDERS

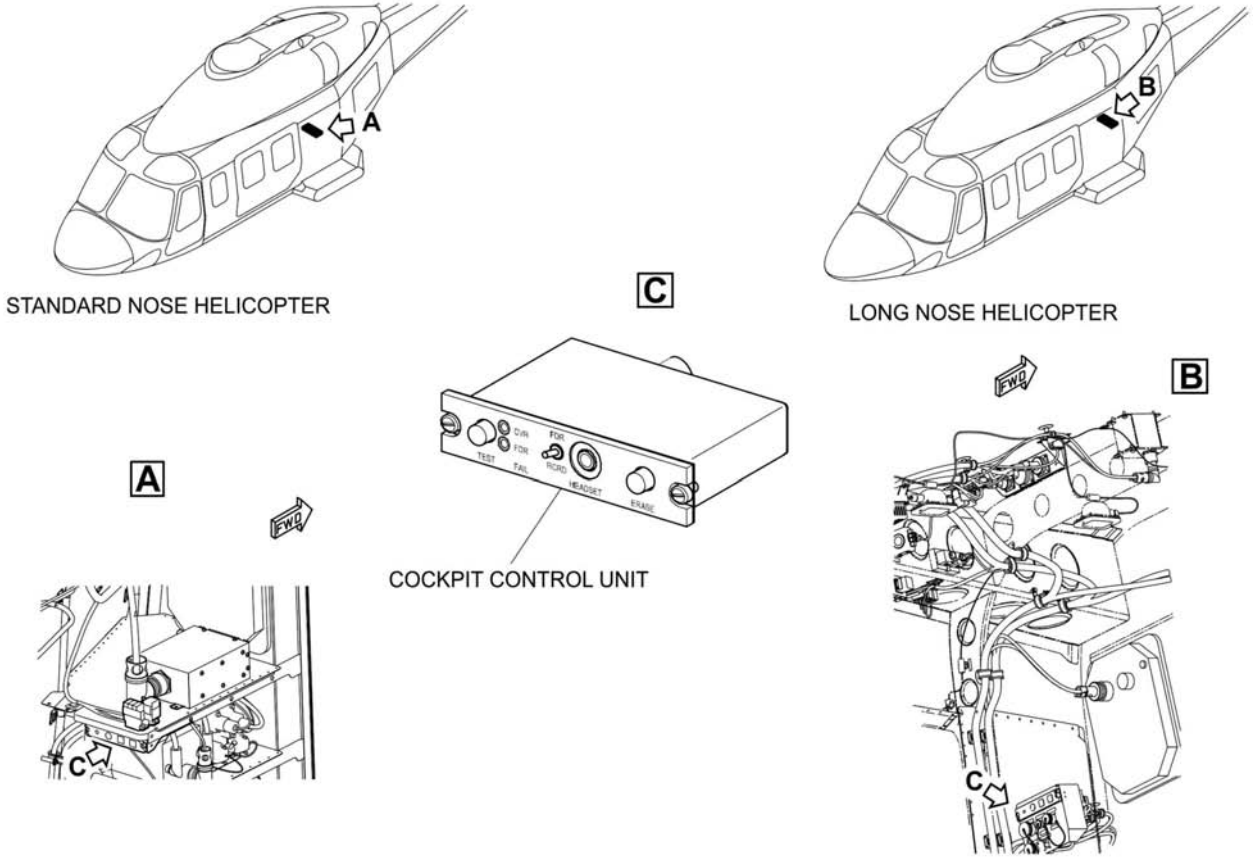
DESCRIPTION

The MPFDR system is composed by:

- Multi-Purpose Flight Data Recorder
- a Cockpit Control Unit
- an Area Microphone
- an Accelerometer

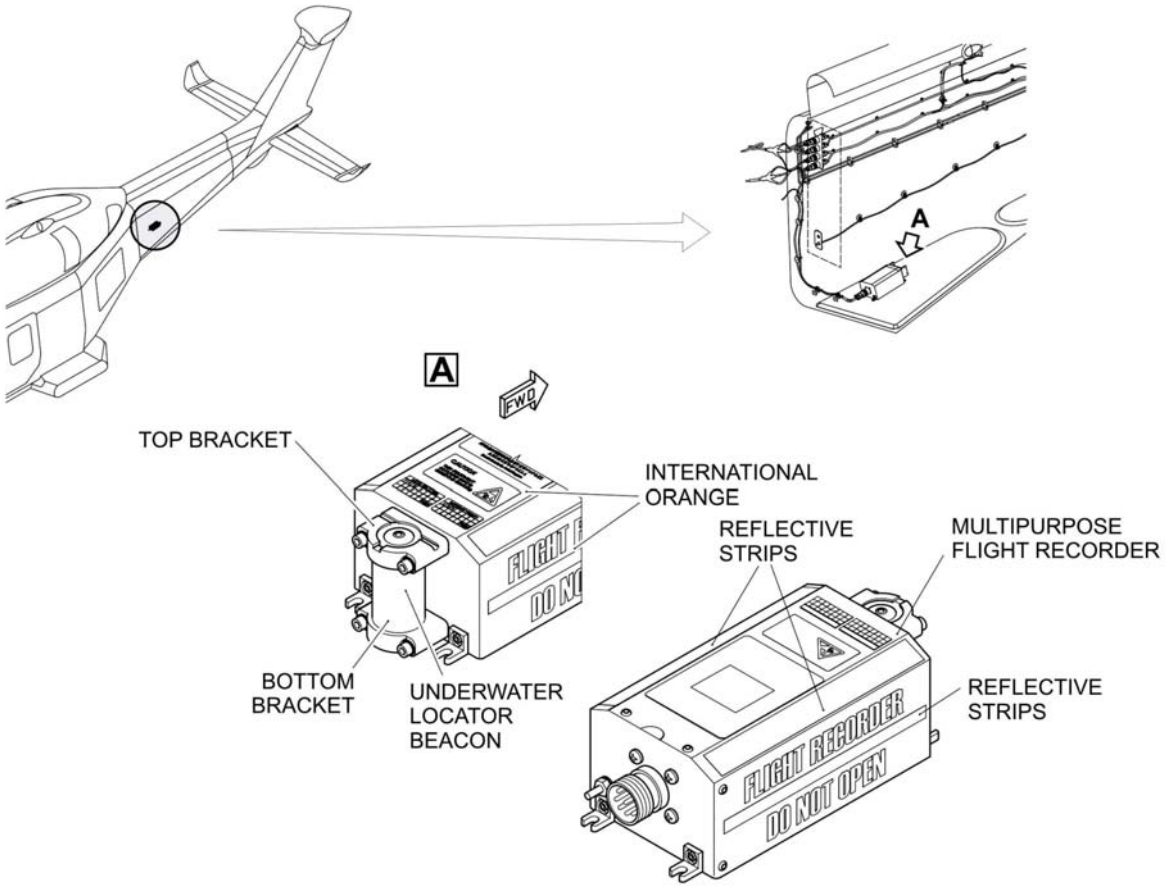
The MPFDR is a crash protected airborne Multi Purpose Flight recorder to meet mandatory requirements and is used use to record selected aircraft parameters and 4 audio sources.
The MPFDR consists of a solid state memory module and an electronic interface.
The solid state memory module is protected for crash survivability.
The electronic interface is not crash protected and is not required to survive conditions exceeding the specified operating and storage environment.
An Underwater Locator Beacon (ULB) is fitted to the Crash Survivable Memory Module as an aid to location in the event of an accident over water.
The MPFDR is painted International Orange as an aid to its location.
The MPFDR is designed to operate with the Cockpit Control Unit and Area Microphone.
The Cockpit Control Unit forms part of the Aircraft Combined Voice and Flight Data Recording System and is installed in the avionics bay.
The unit is a panel mounted enclosure containing the Area Microphone Pre-amplifier, provision for an integral or externally mounted Area Microphone, a Headphone jack socket together with pushbuttons and indicators for the self-test and voice erase facilities of the MPFDR.
The Remote Microphone is mounted in a suitable location in the cockpit on the Instrument panel, either as the main audio source or to supplement the Internal Microphone on the Control Unit.

The download mode allows the high speed recovery of all recorded aircraft data or audio parameters via LAN Connection (J182) to a normal network computer module. The download can be done on or off the aircraft. Initiating this mode ON the aircraft aborts any current MPFDR Record mode operation for the duration of the download process and it is allowed only on the GND and engines OFF.



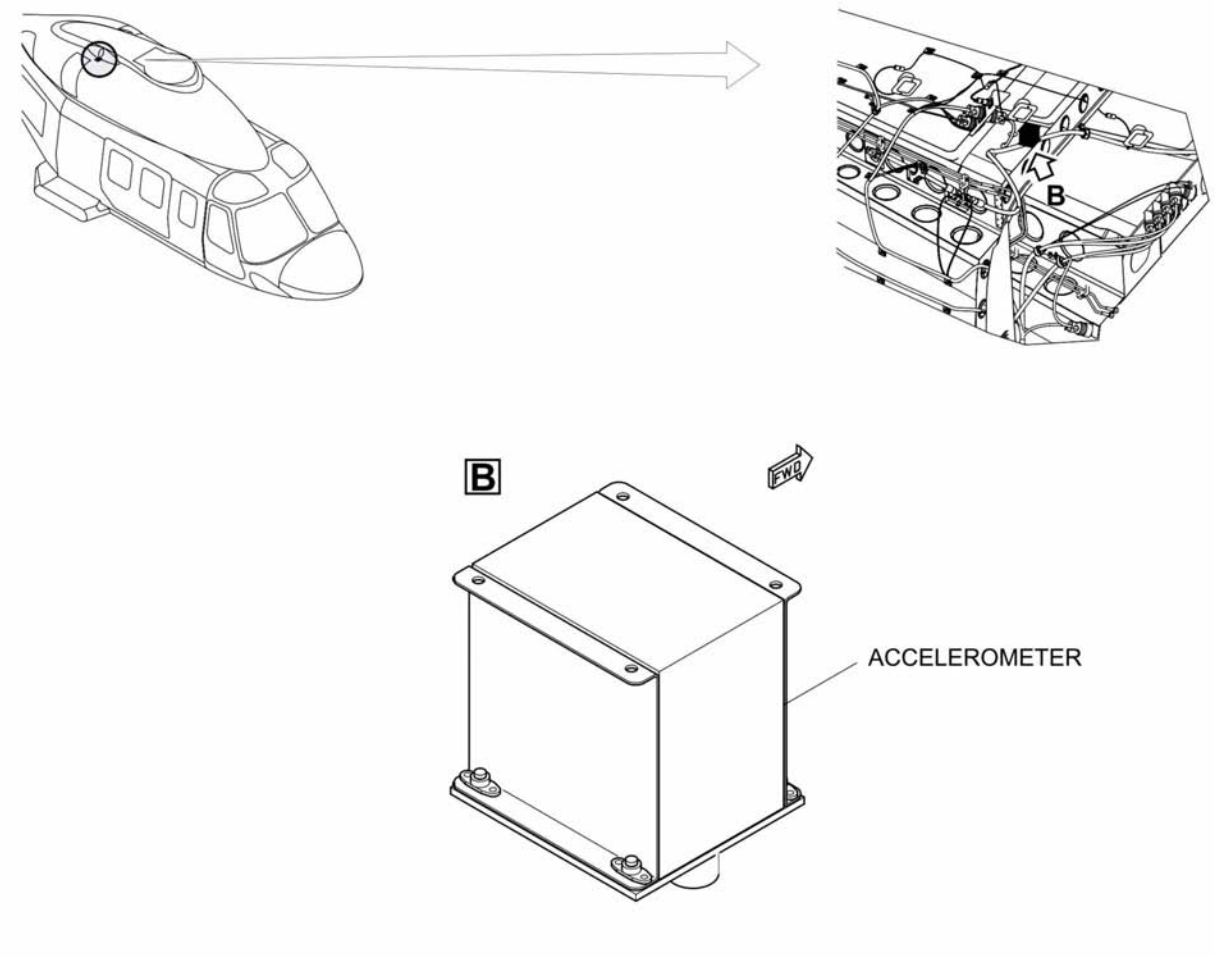
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MULTIPURPOSE FLIGHT DATA RECORDER SYSTEM COMPONENTS LOCATION (SHEET 1 OF 10)



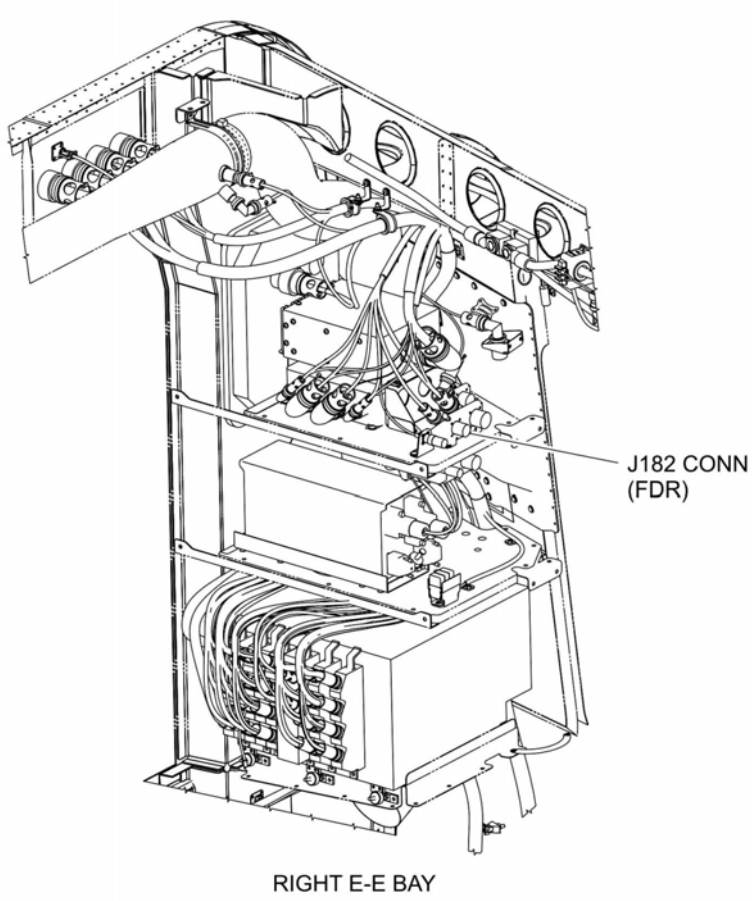
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MULTIPURPOSE FLIGHT DATA RECORDER SYSTEM COMPONENTS LOCATION (SHEET 2 OF 10)



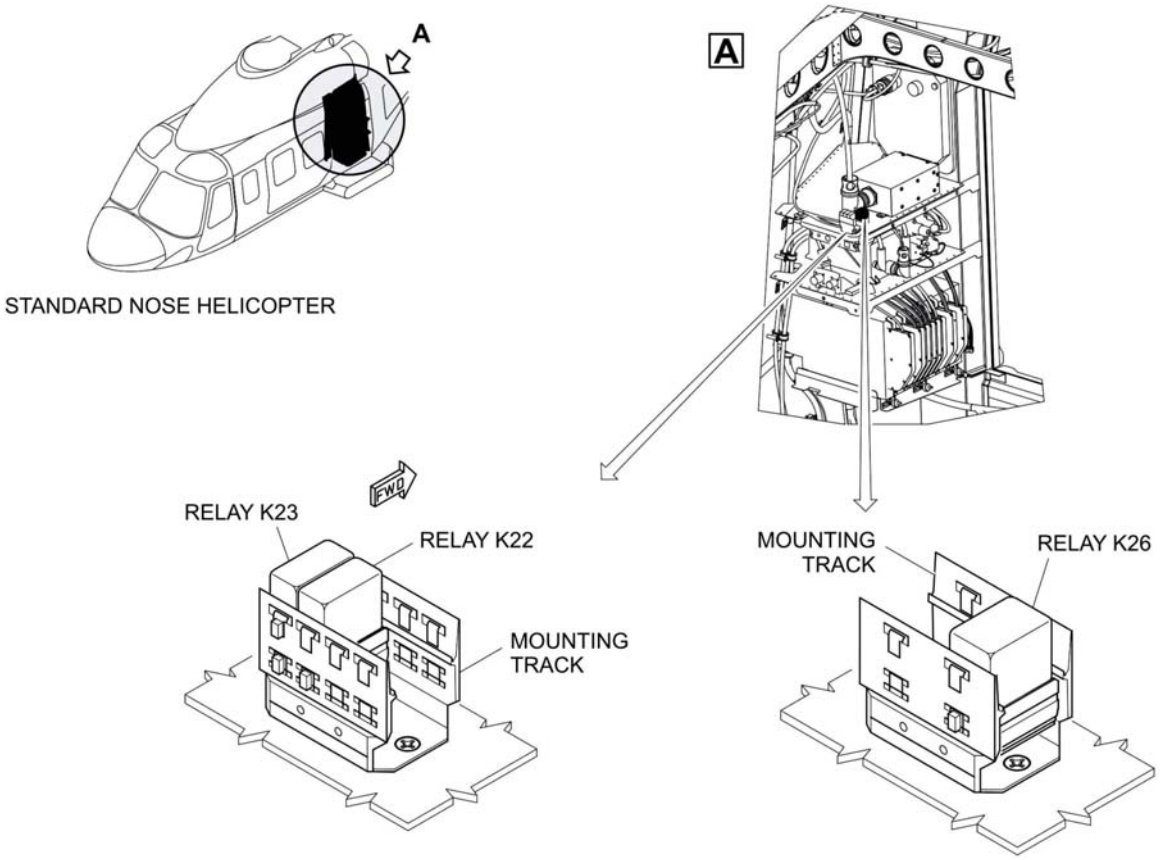
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MULTIPURPOSE FLIGHT DATA RECORDER SYSTEM COMPONENTS LOCATION (SHEET 3 OF 10)



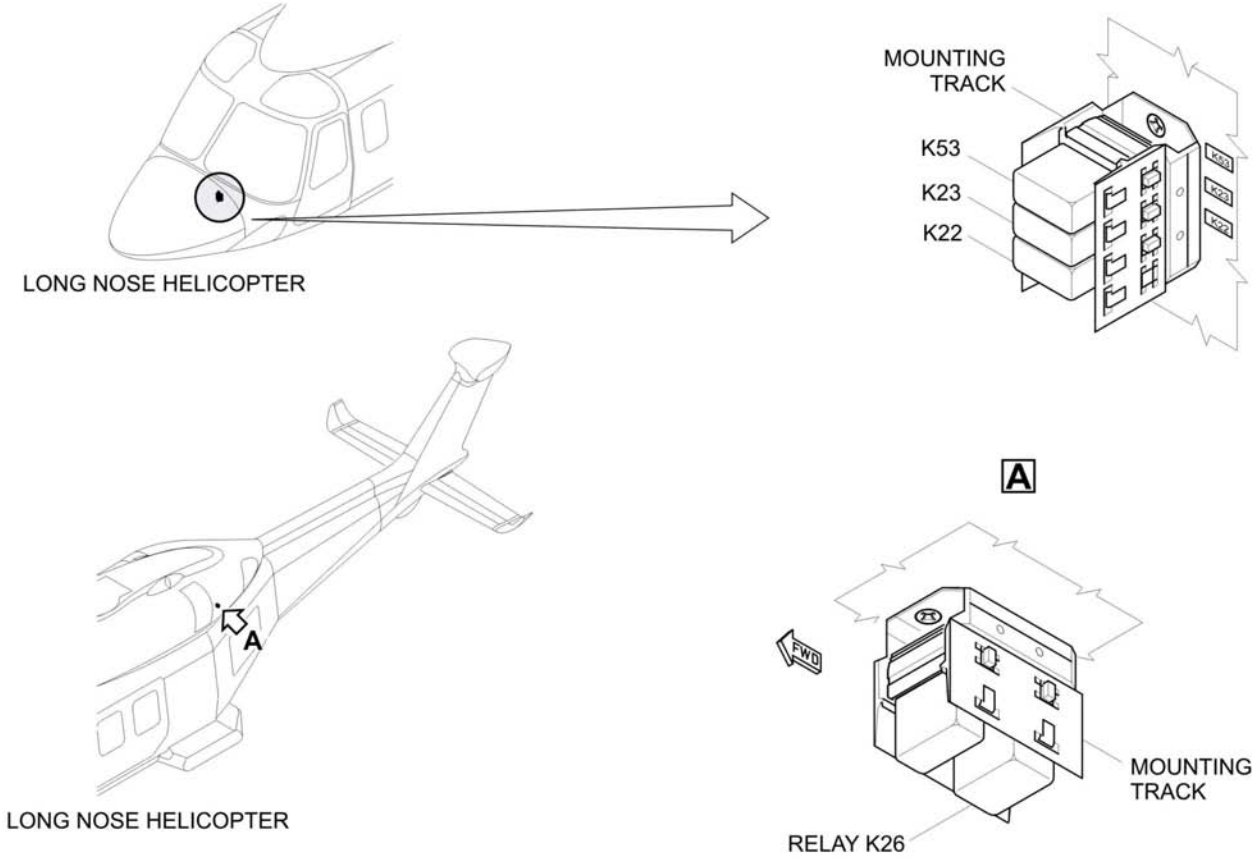
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MULTIPURPOSE FLIGHT DATA RECORDER SYSTEM COMPONENTS LOCATION (SHEET 4 OF 10)



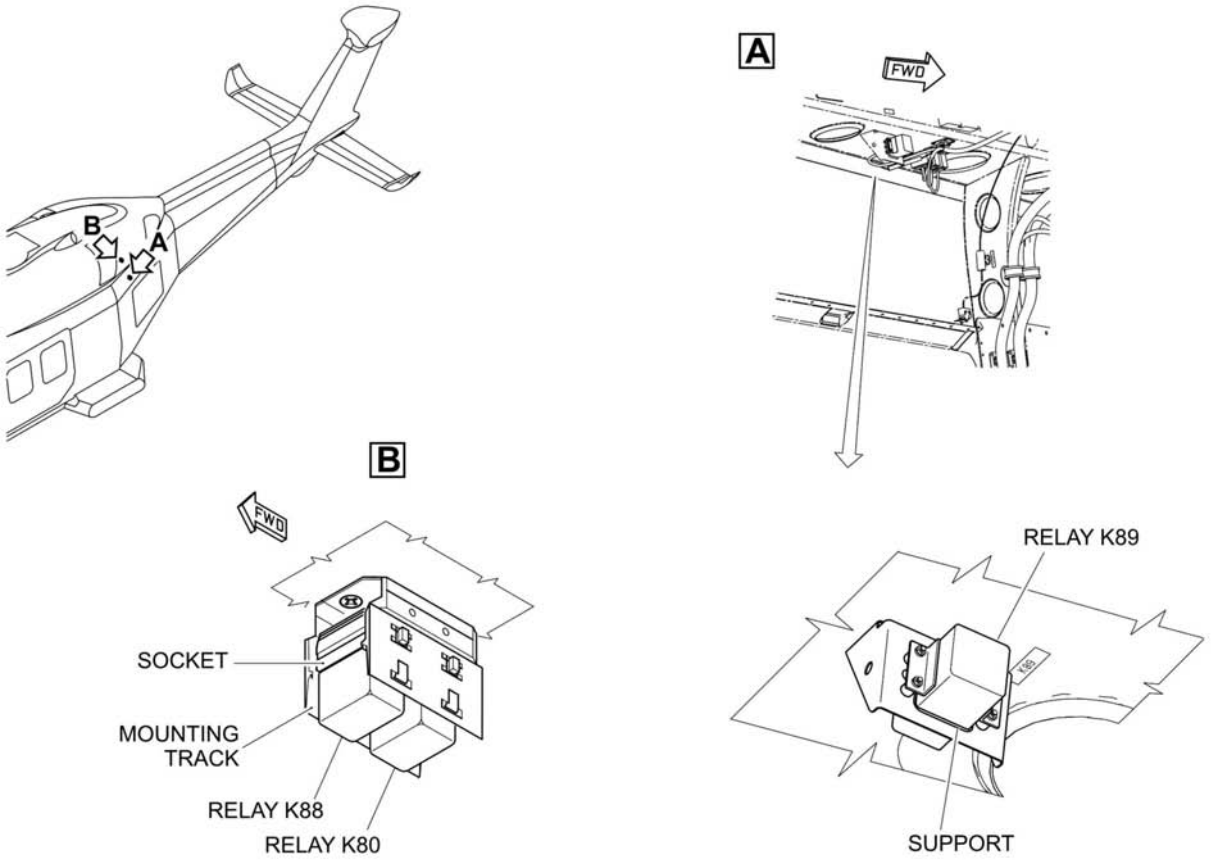
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MULTIPURPOSE FLIGHT DATA RECORDER SYSTEM COMPONENTS LOCATION (SHEET 5 OF 10)



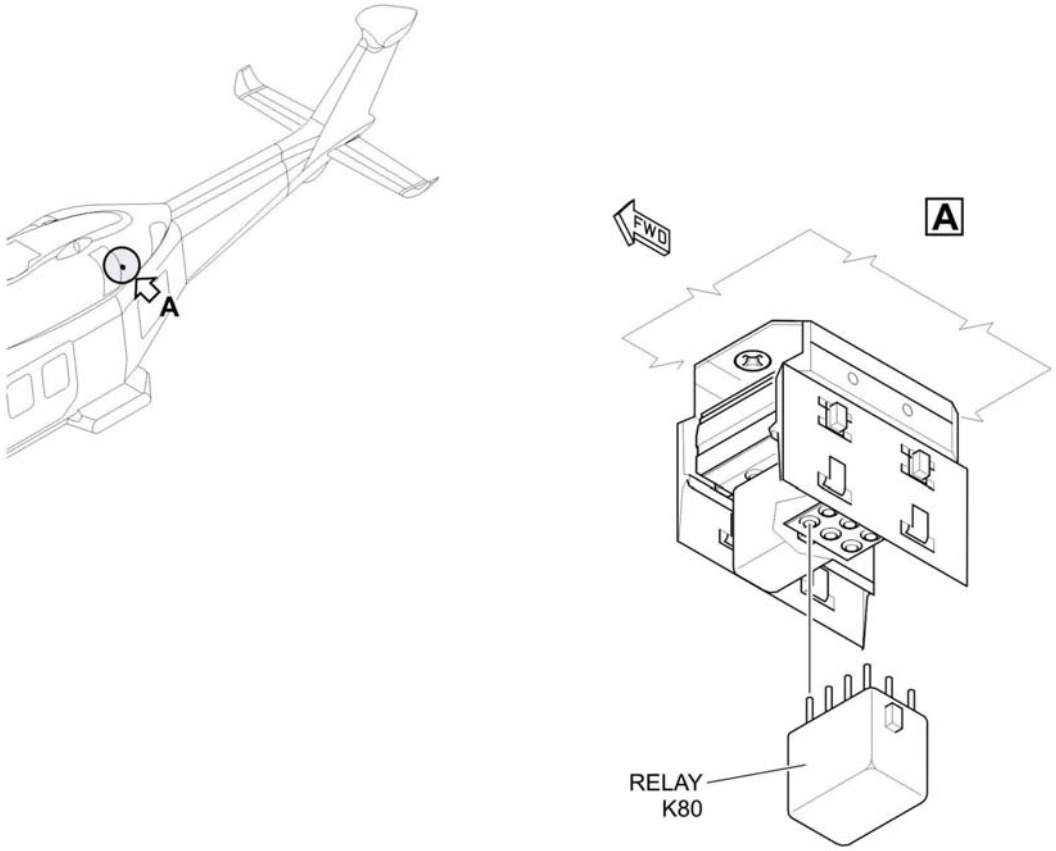
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MULTIPURPOSE FLIGHT DATA RECORDER SYSTEM COMPONENTS LOCATION (SHEET 6 OF 10)



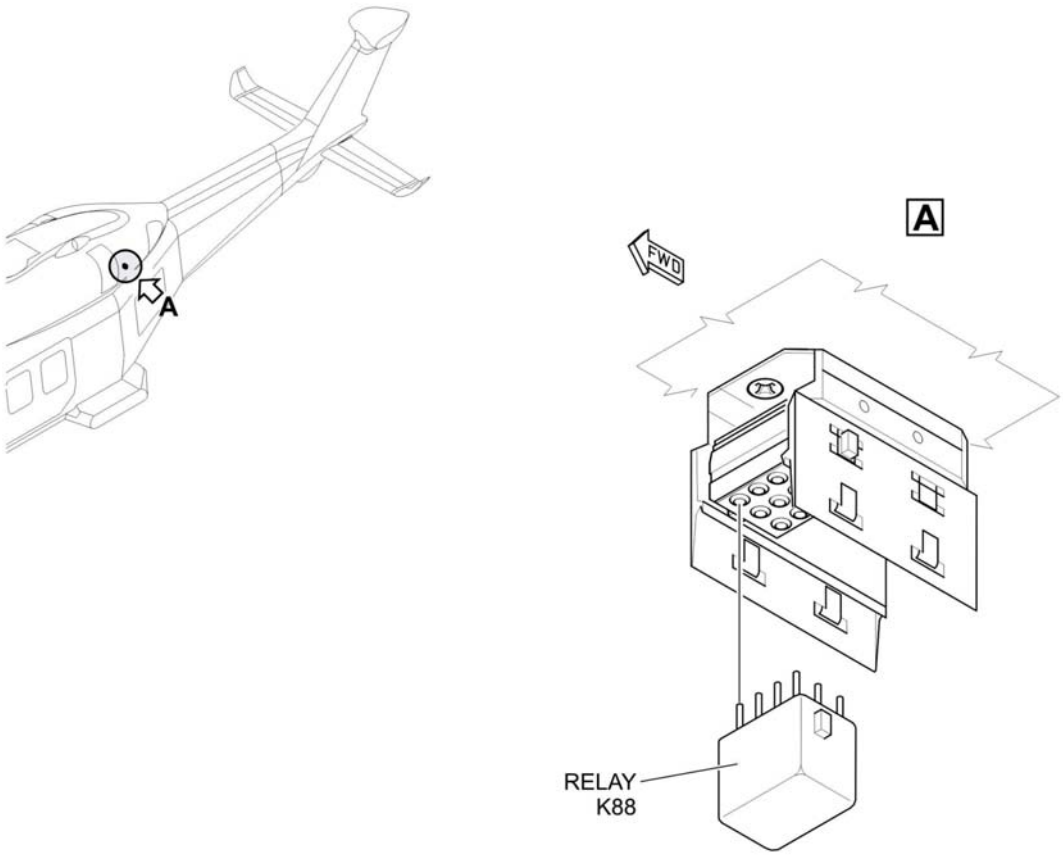
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MULTIPURPOSE FLIGHT DATA RECORDER SYSTEM COMPONENTS LOCATION (SHEET 7 OF 10)



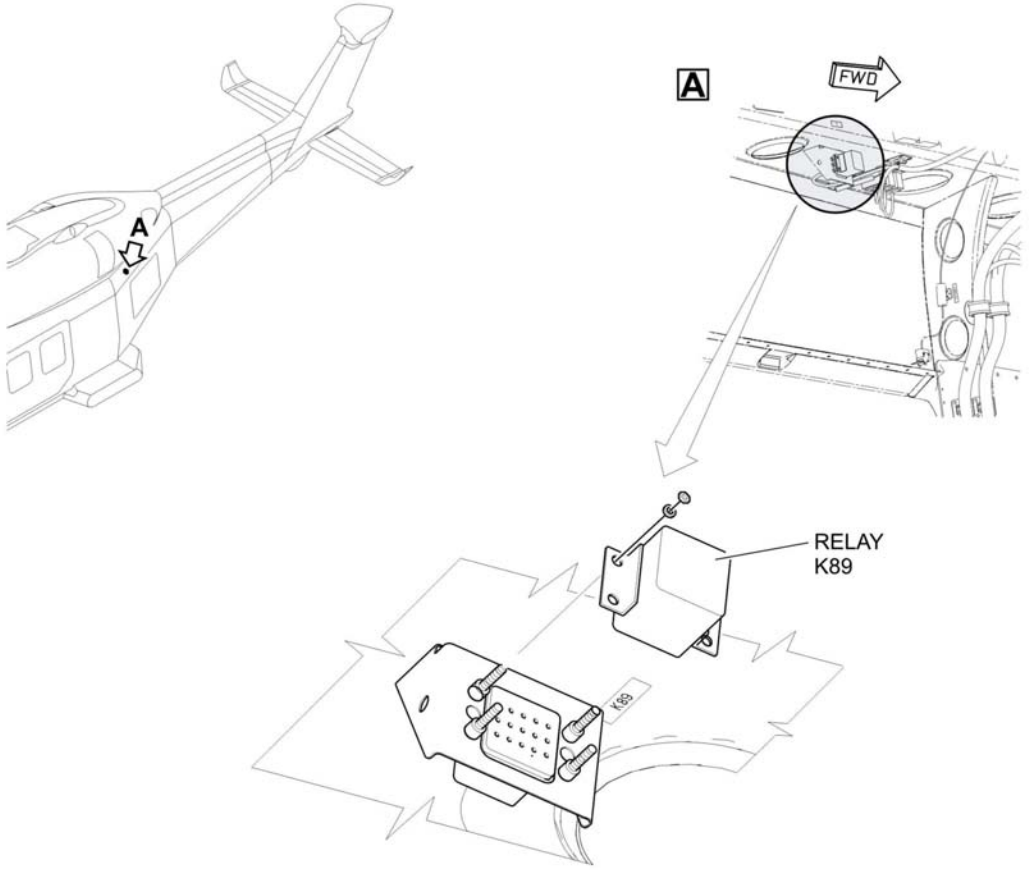
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MULTIPURPOSE FLIGHT DATA RECORDER SYSTEM COMPONENTS LOCATION (SHEET 8 OF 10)



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MULTIPURPOSE FLIGHT DATA RECORDER SYSTEM COMPONENTS LOCATION (SHEET 9 OF 10)



ICN-39-A-313000-G-A0126-01333-A-01-1

MULTIPURPOSE FLIGHT DATA RECORDER SYSTEM COMPONENTS LOCATION (SHEET 10 OF 10)

OPERATION

An MPFDR Cockpit Control Unit (CCU), installed in the Avionics Bay provides to the Flight Crews the control of the following functions:

- A TEST function of the system
- A Voice Erase of the Cockpit Voice Recording
- A FDR RCDR

None of the above controls need to be used by the pilot in Flight.

The MPFDR Cockpit Control Panel indicates to the Flight Crew the following functions:

- CVR FAIL indication visible on the CAS
- FDR FAIL indication visible on the CAS

The CVR and FDR fail lights on the CCU indicate both a recognized failure in the CVR/FDR system or a CVR/FDR non-recording mode. The CCU fail lights are not constrained by any suppression logic, they just visualize the current MPFDR status.

The CCU has a Headphone jack socket too.

The Multi Purpose Flight Data Recorder Unit

The MPFDR records selected aircraft parameters, including audio, into Solid State Non-volatile Memory. The MPFDR retains, as a minimum, either the most recent of aircraft data and the most recent 120 minutes of four audio sources.
The MPFDR is designed to operate in conjunction with the Modular Avionics Unit and suitable Portable Replay Equipment (PRE) for maintenance purpose.
An Underwater Locator Beacon (ULB) is fitted to the Crash Survivable Memory Module as an aid to location in the event of an accident over water.

The primary recording data and audio parameters of the MPFDR are the following:

- 4 Audio channel input (including Cockpit Voice Recorder input and providing also radio COMMs, NAV and Approach aids signals):
  - CH1: Cabin ICS;
  - CH2: Copilot Headset;
  - CH3: Pilot Headset;
  - CH4: Cockpit Microphone.
- Flight Data aircraft parameters input selected from MAU No. 1.
- Audio output going to the Headphone Jack via the MPFDR Control Panel to monitor the sum of all the Audio Channel input on the Ground only.

The Cockpit Voice Recording starts automatically on the application of power to the MPFDR and stops within 10 minutes if the now recording condition persists. The CVR recording restarts for 10 minutes if the TEST pushbutton is activated.
If one engine not OFF or aircraft not on ground, the FDR/ CVR data recording start automatically.
When the aircraft is on ground and both engines OFF, the FDR data recording stops immediately while the CVR data recording stops within 10 minutes.

Cockpit Area Microphone

The Remote Microphone records the conversation of the Flight Crew and the different noises in the Cockpit. It is mounted on the Instrument Panel in a suitable location to provide the main audio source or to supplement the Internal Microphone on the Control Unit.

Accelerometer

The Accelerometer sends data to the MAU No. 1 the different acceleration of the three axes: lateral, longitude, vertical. it is mounted inside the baggage

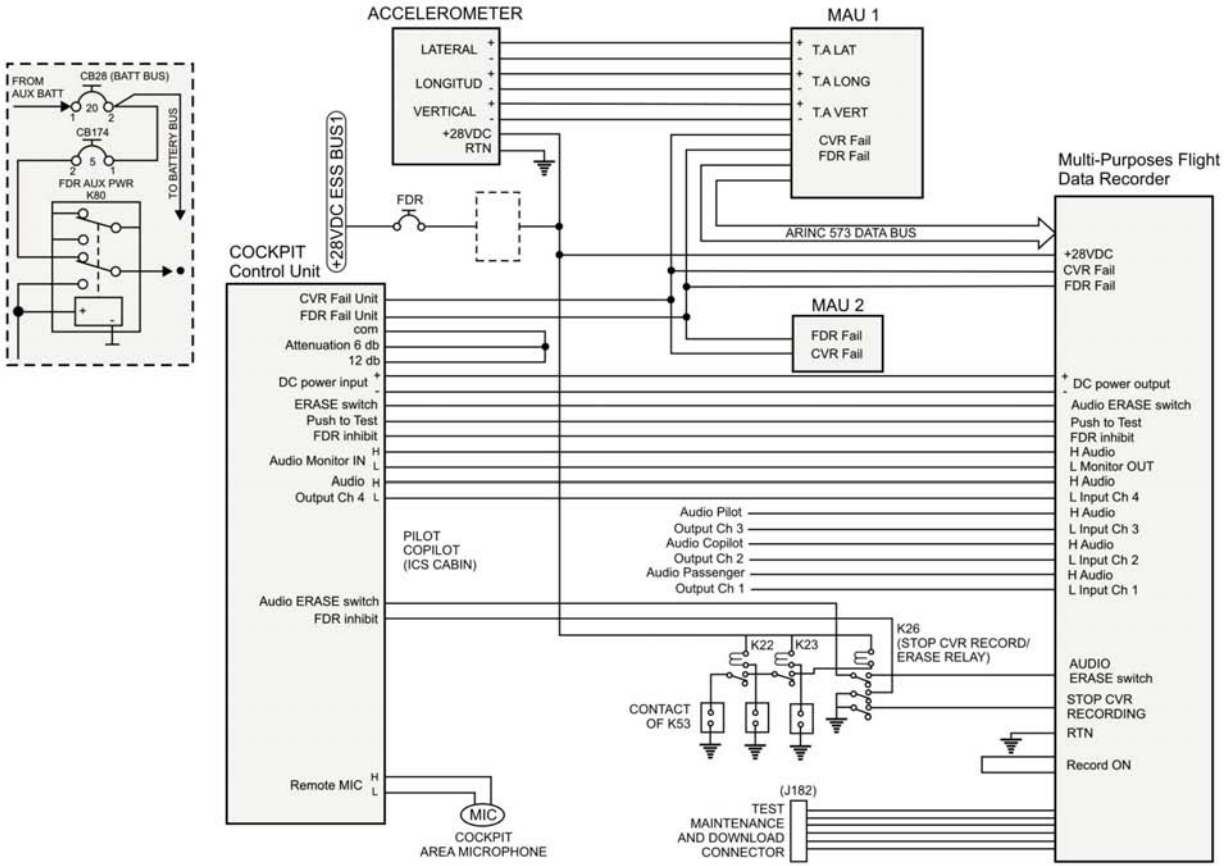
compartment and it is supplied from the same power supply of the MPFDR. Its ranges are:

- Vertical: +6G to -3G;
- Longitudinal: +/- 1G;
- Lateral: +/- 1G.

FDR, CVR system check for maintenance purpose

It is possible to download the MPFDR, CVR information from the MPFDR on to a copy PC. The downloaded data can be transferred to an analysis PC for investigation.

The MPFDR System Check maintenance is carried out with the MPFDR ON or removes it from the aircraft and not in Flight.



ICN-39-A-313000-G-A0126-01334-A-01-1

MULTIPURPOSE FLIGHT DATA RECORDER ELECTRICAL SCHEMATIC

MPFDR OPERATION	
FDR CAPACITY	• 25 hours Flight parameters • 2 hours Audio Voice data
AUDIO VOICE RECORDING STARTS AFTER A/C POWER APPLIED	
AUDIO RECORDING WILL STOP AFTER 10 MINS UNLESS	• Weight On Wheels is false, or • Eng 1 is not OFF, or • Eng 2 is not OFF
AUDIO VOICE DATA RECORDER FOR 2 HOURS	
AUDIO RECORDING WILL CONTINUE FOR 10 MINS AFTER A/C SHUT DOWN	
FLIGHT DATA RECORDING WILL BE PERFORMED ONLY WHEN	• Weight On Wheels is false, or • Eng 1 is not OFF, or • Eng 2 is not OFF
MODULAR AVIONIC UNIT #1 TRANSFER FLIGHT DATA VIA ARINC 573 DATABUS TO FDR	

The MAU No.1 provides an ARINC 573 interface to send to the FDR the information listed below.

NO.	PARAMETER	TYPE OF SIGNAL	SOURCE
1	Time	Digital	MAU1

NO.	PARAMETER	TYPE OF SIGNAL	SOURCE
2a	Pressure Altitude (ALT)	ASCB-D	ADC1, ADC2
2b	Vertical Speed (VS)	ASCB-D	ADC1, ADC2
3	Indicate Airspeed (IAS)	ASCB-D	ADC1, ADC2
4	Magnetic Heading	ASCB-D	AHRS1, AHRS2
4	Magnetic Variation	ASCB-D	AHRS1, AHRS2
5a	Normal Acceleration	Analog	Tri-axial Accelerometer
5b	Tri-Axial Accelerometer Valid	Analog	Tri-axial Accelerometer
5c	Normal Acceleration (AHRS)	ARINC-429	AHRS1, AHRS2
6	Pitch Attitude	ASCB-D	AHRS1, AHRS2
7a	Pitch Altitude	ASCB-D	AHRS1, AHRS2
8a	Radio PTT key (pilot)	Discrete	PTT Switch (pilot)
8b	Radio PTT key (co-pilot)	Discrete	PTT Switch (co-pilot)
9a	NF1 & NF2	RS-422A & Anal.	EEC1, EEC2 and NF Sensor
9b	Torque 1 & Torque 2	RS-422A & Anal.	EEC1, EEC2 and NF Sensor
9c	ITT1 & ITT2	RS-422A & Anal.	EEC1, EEC2 and T5 Sensor
9d	NG1 & NG2	RS-422A	EEC1, EEC2 and NG Sensor
9e	EEC1, EEC2 Mode	RS-422A	EEC1, EEC2

NO.	PARAMETER	TYPE OF SIGNAL	SOURCE
9f	Power Index (displayed parameter)	Digital	PFD PLT (NG, ITT or TQ)
9g	Power Index (displayed equivalent value)	ASCB-D	PFD PLT PI
9h	Engine 1&2 Oil Temp	Anal.	Sensor
10a	Rotor RPM (NR)	RS-422A &Anal.	EEC1, EEC2 and NR Sensor
10b	Rotor Brake	Discrete	Switch
11a	Collective Pitch	Rs-422A	EEC1, EEC2
11b	Longitudinal Cyclic	Analog	RVDT (dual sensor – AFCS)
11c	Lateral Cyclic	Analog	RVDT (dual sensor – AFCS)
11d	Tail Rotor Pedal	Analog	RVDT (dual sensor – AFCS)
11e	Hydraulic 1 Selection	Discrete	Switch
11f	Hydraulic 2 Selection	Discrete	Switch
12a	Hydraulic 1 Low Pressure	Discrete	Pressure Switch
12b	Hydraulic 2 Low Pressure	Discrete	Pressure Switch
12c	Hydraulic 1 / 2Oil Temp	Analog	Sensor
13	T1, OAT	RS-422A 6 Anal.	EEC, EEC2 and OAT Probe 1&2
14	AFCS Mode and Engagement	Digital (ASCB)	AFCS Mode Selector
15a	SAS1 On	Discrete	SA1 Controller

NO.	PARAMETER	TYPE OF SIGNAL	SOURCE
15b	SAS2 On	Discrete	SAS2 Controller
16	Main Gearbox Oil Pressure	Analog	MGB Pressure Sensor
17	Main Gearbox Oil Temperature	Analog	MGB Temperature Sensor
17a	IGB Oil Temperature	Analog	IGB Temperature Sensor
17b	TGB Oil Temperature	Analog	TGB Temperature Sensor
18	Yaw Rate	ASCB-D	AHRS1, AHRS2
20a	Longitudinal Acceleration	Analog	Tri-axial Accelerometer
20b	Longitudinal Acceleration (AHRS)	ARINC-429	AHRS1, AHRS2
21a	Lateral Acceleration	Analog	Tri-axial Accelerometer
21b	Lateral Acceleration (AHRS)	ARINC-429	AHRS1, AHRS2
22a	Radio Altitude (pilot PFD)	Analog	Radio Altimeter
22b	Radio Altitude (copilot PFD)	Analog	Radio Altimeter n.2 when installed
22c	RALT Validity	Discrete	1,2 RAD ALT validity discretes
23	Glide Slope 1 & 2 Deviation	ASCB-D	MRC1, MRC2
24	Localizer 1 & 2 Deviation	ASCB-D	MRC1, MRC2
25	Marker Beacon 1 & 2	ASCB-D	MRC1, MRC2
26a	MWL On	Digital	MAU1

NO.	PARAMETER	TYPE OF SIGNAL	SOURCE
26b	MCL On	Digital	MAU1
26c	Main Gearbox Low Pressure	Discrete	Pressure Switch 1 & 2
26d	SAS1 & SAS2 Failure	ASCB-D	MAU1, MAU2
26e	Each “Red” Warning	Discrete	Switch MAU for High/Low NR
26f	Generator 1 & 2 Failure	Discrete	Switch
26g	Inverter 1 & 2 Failure	Discrete	Switch
26h	EEC1 & EEC2 Failure	RS-422A & Discrete	EEC1, EEC2
26i	DSCWD1, DSCWD2, NCFUR1, NCFUR2, CFUR1	RS-422	EEC1, EEC2
26j	ADC1 & ADC2 Valid	ASCB-D	ADC1, ADC2
26k	AHRS1 & AHRS2 Valid	ASCB-D	AHRS1, AHRS2
26l	NAV1 & NAV2 Valid	ASCB-D	MRC1, MRC2
26m	DME Valid	ASCB-D	MRC1
26n	FMS Valid	ASCB-D	FMS (if installed)
26o	All “Amber” Cautions	Digital	MAU (acquired each 4 seconds)
26p	MAU 1 / 2 failure on PFDs	Digital	DU
26q	1 / 2 CAS Miscomp. on PFDs	Digital	MAU
27	VOR/ILS 1 & 2 Frequency	ASCB-D	MRC1, MRC2

NO.	PARAMETER	TYPE OF SIGNAL	SOURCE
28a	DME on Pilot PFD Distance	ASCB-D	MRC2
28b	DME on Cplt PFD Distance	ASCB-D	MRC2
29a	Latitude/Longitude	ASCB-D	FMS (if installed)
29b	Drift Angle	ASCB-D	FMS (if installed)
29c	Wind Speed	ASCB-D	FMS (if installed)
29d	Wind Direction	ASCB-D	FMS (if installed)
30a	WOW1	Discrete	Switch
30b	WOW2	Discrete	Switch
30c	Syntetic Ground	Discrete	MAU
33a	Fuel Contents 1 / 2	ARINC-429	FCU
33b	Fuel Flow 1 / 2	ASCB-D	MAU
33c	Fuel Press 1 / 2	Analog	Sensor
34	Altitude Rate	ASCB-D	ADS1 – ADS2
35	Ice Detection	Discrete	Sensor (when installed)
36	HUMS Data	ARINC-429	HUMS Computer (if installed)
38a	Baro Set (Pilot)	ASCB-D	PFD (Pilot)
38b	Baro Set (Co-Pilot)	ASCB-D	PFD (Co-Pilot)

NO.	PARAMETER	TYPE OF SIGNAL	SOURCE
39a	Selected Altitude (Pilot)	ASCB-D	PFD (Pilot)
39b	Selected Altitude (Co-Pilot)	ASCB-D	PFD (Co-Pilot)
40a	Selected Speed (Pilot)	ASCB-D	PFD (Pilot)
40b	Selected Speed (Co-Pilot)	ASCB-D	PFD (Co-Pilot)
42a	Selected Vertical Speed (Pilot)	ASCB-D	PFD (Pilot)
42b	Selected Vertical Speed (Co-Pilot)	ASCB-D	PFD (Co-Pilot)
43	Selected Heading	ASCB-D	PFD (Pilot)
44a	Selected Course (Pilot)	ASCB-D	PFD (Pilot)
44b	Selected Course (Co-Pilot)	ASCB-D	PFD (Co-Pilot)
44c	Selected flight path (All pilot selectable course of operation: VOR 1 / 2, LOC 1 / 2, FMS 1 / 2	ASCB-D	PFD (Pilot)
44d	Selected flight path (All Co-Pilot selectable course of operation: VOR 1 / 2, LOC 1 / 2, FMS 1 / 2	ASCB-D	PFD (Co-Pilot)
45a	Selected Decision Height (Pilot)	ASCB-D	PFD (Pilot)
45b	Selected Decision Height (Co-Pilot)	ASCB-D	PFD (Co-Pilot)
46a	PFD Format (Pilot)	ASCB-D	PFD (Pilot)
46b	PFD Format (Co-Pilot)	ASCB-D	PFD (Co-Pilot)
47a	MFD Format (Pilot)	ASCB-D	MFD (Pilot)

NO.	PARAMETER	TYPE OF SIGNAL	SOURCE
47b	MFD Format (Co-Pilot)	ASCB-D	MFD (Co-Pilot)
47c	MFD Pilot Config (map, pwr, plant etc.)	Digital	MFD (Pilot)
47d	MFD Co-Pilot Config (map, pwr, plant etc.)	Digital	MFD (Co-Pilot)
48a	Loadmeter Main Battery	Analog	Relevant Shunt
48b	Loadmeter Aux Battery	Analog	Relevant Shunt
48c	Loadmeter Generator 1	Analog	Relevant Shunt
48d	Loadmeter Generator 2	Analog	Relevant Shunt
49	TCAS alarms	Digital	TCAS
50	AWG Regrade	Discrete	Switch
51	NVG Mode	Discrete	Switch
52	Ground Speed	Digital	GPS
53	Day	ASCB-D	MAU
54	Month	ASCB-D	MAU
55	Year	ASCB-D	MAU
56	AC Code	Digital	APM
57	EGPWS alarms	Digital	EGPWS

MPFDR SYSTEM CONTROLS

The MPFDR controls are housed on the Cockpit Control Unit located on the left E-E Bay (central shelf). The CAS message window No.1 displays the caution messages.
The CCU provides the following functions:

1 TEST pushbutton ..... when pressed, initiates its requested Built-In-Test function. The two leds illuminates for 8 sec. about, after which each is asserted only if its associated test is failed. This action also initiates the CVR recording for 10 minutes.

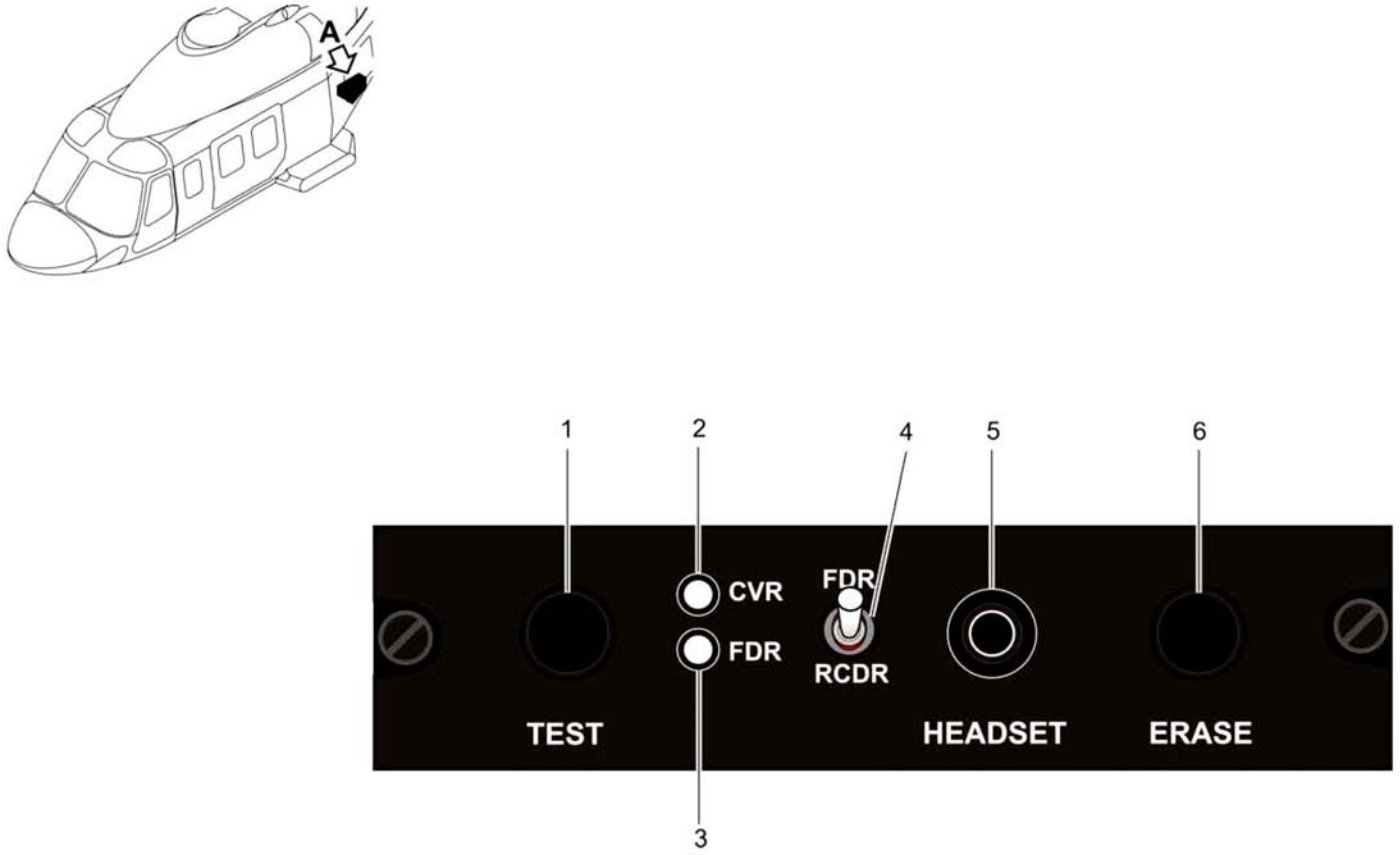
2 CVR FAIL yellow led ..... illuminates steady or flashes. steady when a fail has been detected after BIT sequence or when the MPFDR has been put into a non-recording mode.
Flashes during data recovery or Voice Erase modes. This led is OFF if no problem exists (recording mode).

3 FDR FAIL yellow led ..... illuminates steady or flashes. Steady when a fail has been detected after BIT sequence or when the MPFDR has been put into a non-recording mode. Flashes during data recovery or Voice Erase modes. This led is OFF if no problem exists (recording mode).

4 FDR RCRD spring loaded momentary toggle switch: ..... when set to RCRD, the system is forced recording parameter on ground with engines OFF for maintenance purpose.

5 HEADSET jack socket ..... this receptacle is used to test the audio coming from cockpit area microphone and flight crew interphones. This function is used only on ground with engines OFF.

6 ERASE pushbutton ..... when pressed for more than two seconds, initiates the Cockpit Voice Recorder erase cycle only. This function is enable on ground with engines OFF. The erase lasts for approximately 5 sec during which period the CVR FAIL yellow led will flash.



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MULTIPURPOSE FLIGHT DATA RECORDER SYSTEM CONTROLS

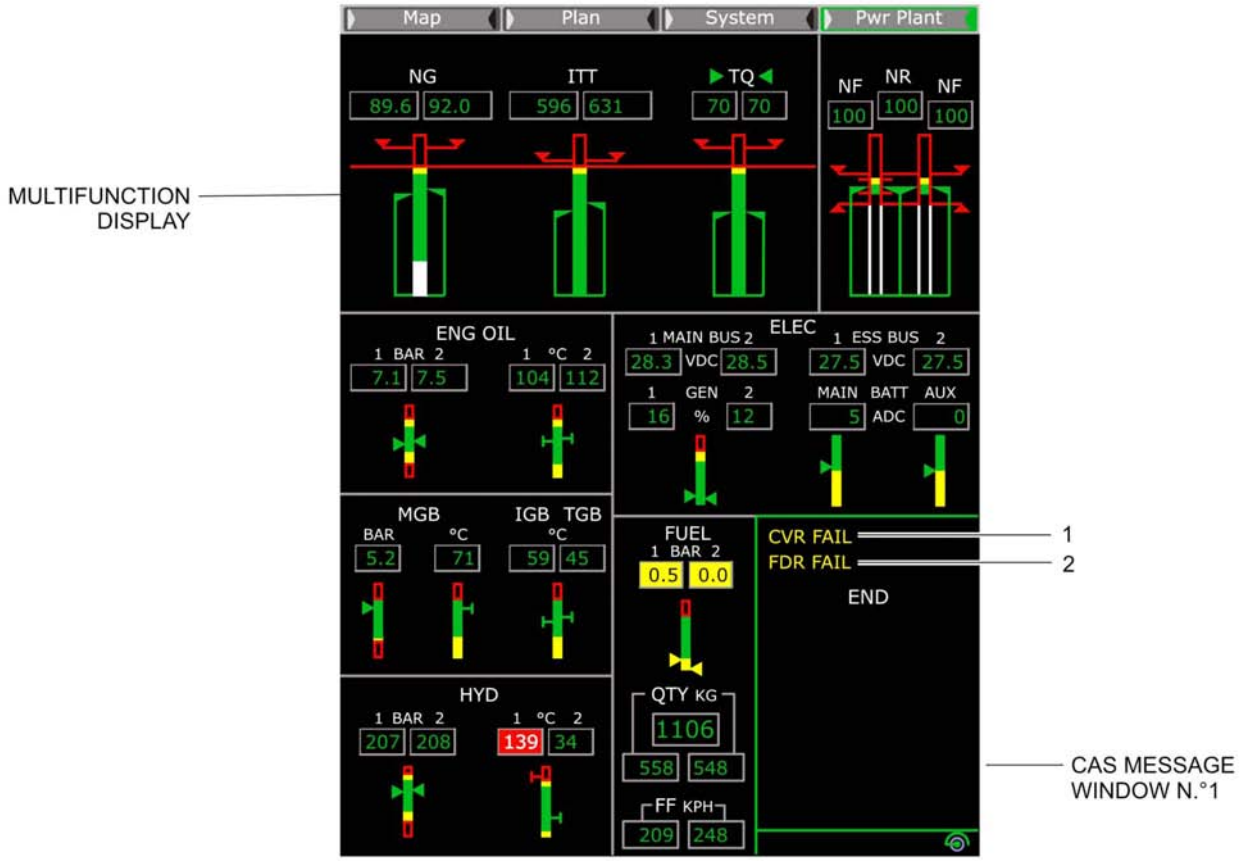
MPFDR SYSTEM INDICATING

The caution provided in the CAS message window is as follows:

1	CVR FAIL	when a malfunction of the CVR or a non-recording mode with at least one engine running is detected.
2	FDR FAIL	when a malfunction of FDR or a non-recording mode with at least one engine running is detected.

NOTE A

The above cautions are suppressed on ground if engines OFF or starting not active.



ICN-39-A-313000-G-A0126-01340-A-01-1

MULTIPURPOSE FLIGHT DATA RECORDER SYSTEM INDICATING

LEADING PARTICULARS

- Multipurpose Flight Data Recorder
  - Nominal supply voltage..... 28VDC (ESS BUS No.1)
- Cockpit Control Unit
  - Nominal supply voltage..... 18VDC ±1VDC (from the MPFDR)
  - Current consumption..... 100 mA max
- Accelerometer
  - Nominal supply voltage..... 28VDC (ESS BUS No.1)
- Underwater acoustic beacon
  - Operating frequency..... 37.5 KHz ±1 KHz
  - Operating depth..... surface to 20.000 ft
  - Pulse length..... not less than 9 milliseconds
  - Pulse repetition rate..... not less than 0.9 pulse per second
  - Useful life..... six years
  - Operating life..... 30 days (minimum)
- Battery characteristics..... single cell, six year life

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SECTION 40 - GENERAL COMPUTERS

GENERAL DESCRIPTION

The Central Computer includes the systems that follow:
The number 1 modular avionics unit system.
The number 2 modular avionics unit system.
The PRIMUS EPIC® system is an Integrated Avionic System whose architecture is based on the modular avionics units (MAUs).
The two Modular Avionics Units:

- handle, compute and distribute data of all aircraft systems to the four Display Units
- interface mechanical and virtual controllers to control the displays and the avionics
- integrate the functions of crew alerting, navigation and autoflight systems

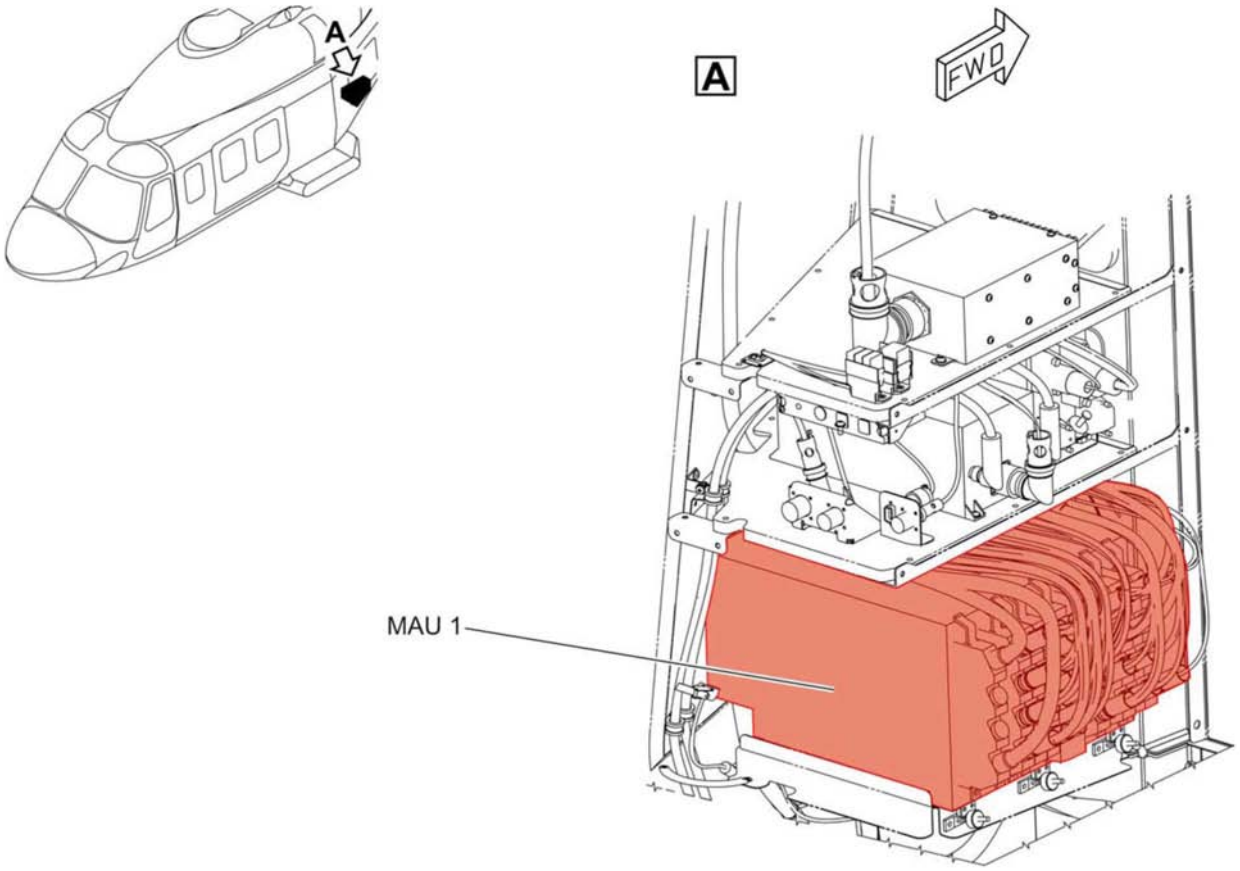
The MAUs contain the hardware units that host the subsystems and functions necessary to operate an aircraft. All the hardware units, installed inside the MAUs, are called line replaceable modules (LRM).The Modular Avionics Unit is a cabinet that contains line replaceable modules that integrate the different MAU functions.
The most important modules are duplicated in MAU 1 and MAU 2 for redundancy, while other modules are single.
The MAU uses a Digital Engine Operating System (DEOS) and data is made available through a bus (called Virtual Back Plane Peripheral Component Interconnect (VbPCI) bus) that connects the NIC to all the modules in the MAU and to the ASCB-D.
The Power Supply module in each MAU is supplied with two 28 VDC inputs for redundancy: one power input only is used at a time.
Two circuit breakers are therefore provided for each MAU:

- MAU 1 PRI PWR connected to MAIN 1 bus
- MAU 1 AUX PWR connected to MAIN 2 bus
- MAU 2 PRI PWR connected to ESS 1 bus
- MAU 2 AUX PWR connected to ESS 2 bus

NOTE A

MAU 1 power inputs are connected to ESS 1 and ESS 2 buses when the aircraft is on the ground or when starting an engine.

Two electrical fans are installed in each MAU cabinet to keep the electronic modules cool; they are automatically operated and monitored.



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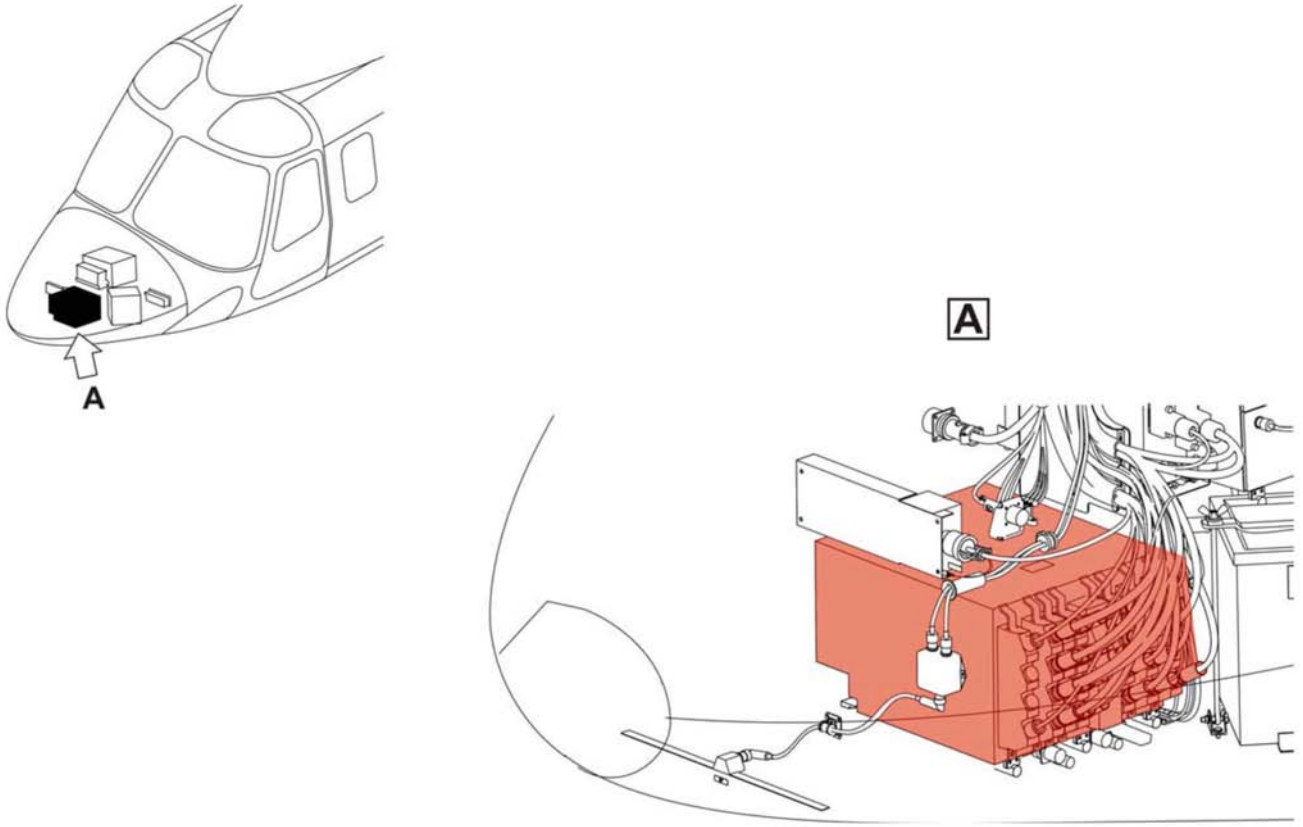
SHORT NOSE – MODULAR AVIONIC UNIT (MAU 1) INSTALLATION

DESCRIPTION AND OPERATION OF MAJOR COMPONENTS

MAU 1

The MAU 1 is installed in the left avionic bay (helicopters [1J]/left nose compartment (helicopters [1L].
The MAU1 system includes the primary components that follow:

- The Number 1 Modular Avionic unit system (MAU1) cabinet
- The Number 1 Power Supply (PS1) module
- The Number 1 Custom Input/Output (CSIO1) module
- The Control Input/Output (CIO1) module
- The Air Management module
- The Number 1 Network Interface Controller/processor (NIC/PROC1) module
- The Number 1 Actuator Input/output Processor (AIOP1) module
- The Number 3 Actuator Input/output Processor (AIOP3) module
- The Central Maintenance Computer (CMC)
- The Number 1 fan assembly
- The Configuration Module.



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LONG NOSE – MODULAR AVIONIC UNIT (MAU 1) INSTALLATION

NUMBER 1 MODULAR AVIONICS UNIT (MAU1) CABINET

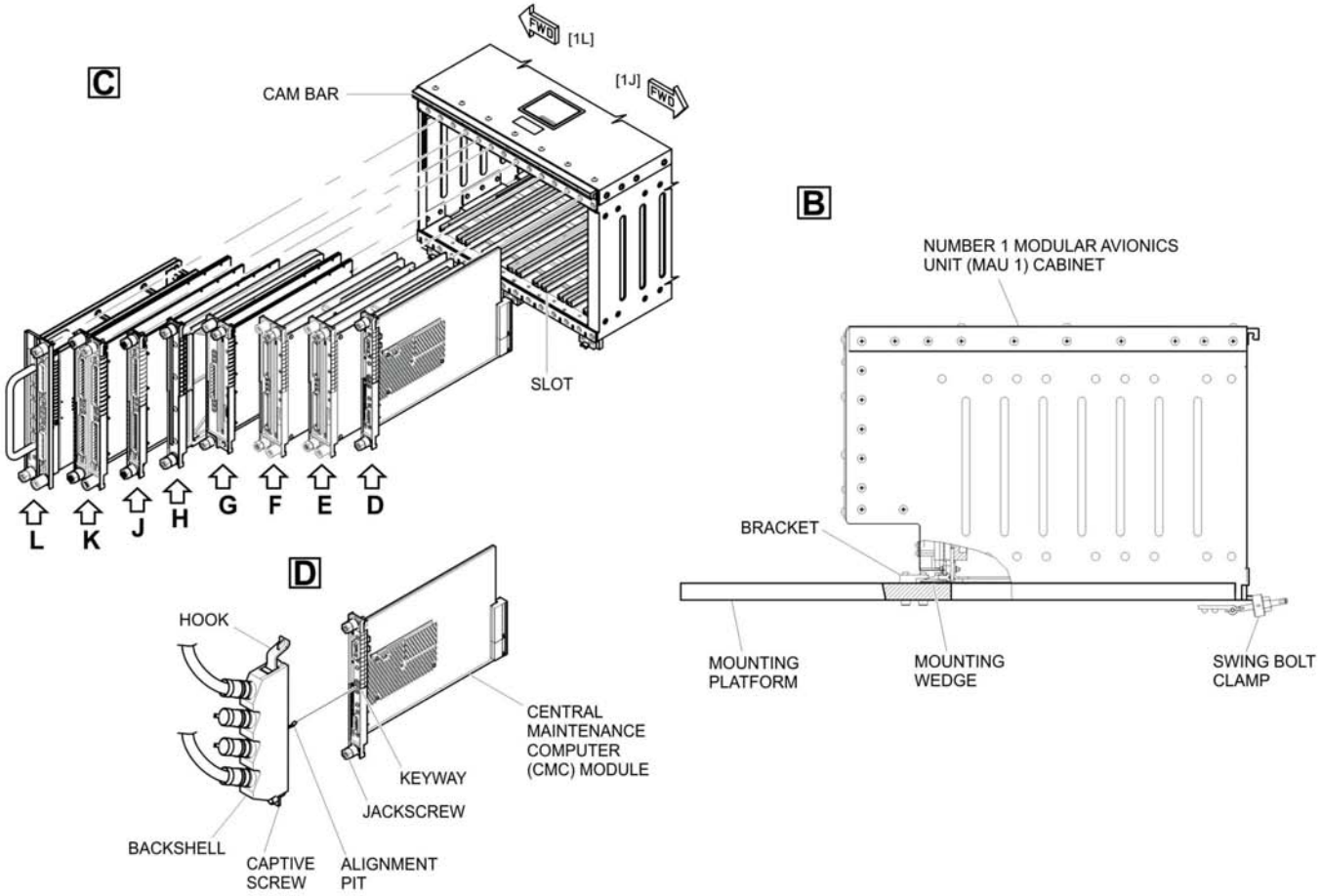
The MAU1 cabinet is installed in the left avionic bay (helicopters [1J]/left nose compartment (helicopters [1L].
The three swing bolts, clamps and a bracket attach the MAU cabinet to its mounting platform. The bracket holds the mounting wedge.
The MAU1 cabinet contains the modules that follow:

- The Power Supply module
- The NIC/PROC module
- The six User Modules.

It has slots to hold the modules.
The MAU1 cabinet also contains the back plane bus, slots, cam bar, the Number 1 fan assembly and the related support.

CENTRAL MAINTENANCE COMPUTER MODULE (CMC)

The CMC module is installed in the MAU1 cabinet. It is installed on the left side of the cabinet. The module it is aligned to the number 9 mark on the cam bar. Two jackscrews attach the CMC module to the MAU1 cabinet. The CMC module has a backshell, which is attached to the module with two captive screws. On top of the backshell there is a hook that engages the cam bar of the MAU1 cabinet, thus the backshell cannot move.
The backshell also has an alignment pin that engages the keyway in the module to align the two components.



ICN-39-A-314000-G-A0126-01394-A-01-1

NUMBER 1 MODULAR AVIONIC UNIT (MAU 1) (SHEET 1 OF 4)

NUMBER 3 ACTUATOR INPUT/OUTPUT PROCESSOR (AIOP3) MODULE

The AIOP3 module is installed in the MAU1 cabinet, on the left side. Two jackscrews hold the AIOP3 module in its slot in the MAU1 cabinet.
The AIOP3 module has a backshell, which is attached to the module with two captive screws. On top of the backshell there is a hook that engages the cam bar of the MAU1 cabinet, thus the backshell cannot move.
The backshell also has an alignment pin that engages the keyway in the module to align the two components.

NUMBER 1 ACTUATOR INPUT/OUTPUT PROCESSOR (AIOP1) MODULE

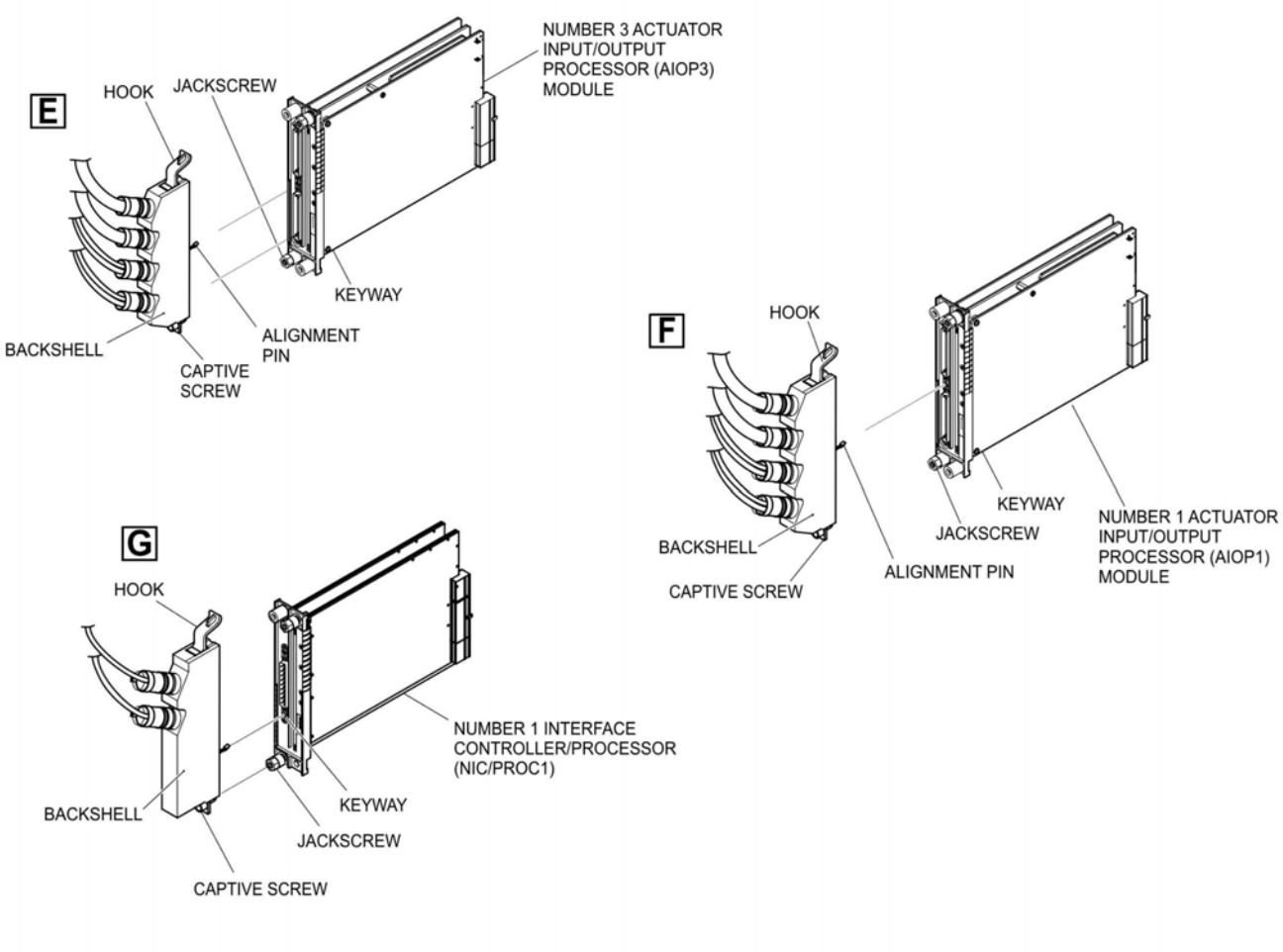
The AIOP1 module is installed in the MAU1 cabinet, on the left side. Two jackscrews hold the AIOP1 module in its slot in the MAU1 cabinet.
The AIOP1 module has a backshell, which is attached to the module with two captive screws. On top of the backshell there is a hook that engages the cam bar of the MAU1 cabinet, thus the backshell cannot move.
The backshell also has a alignment pin that engages the keyway in the module to align the two components.

NUMBER 1 NETWORK INTERFACE CONTROLLER/ PROCESSOR (NIC/PROC1) MODULE

The NIC/PROC1 module is installed in the MAU1 cabinet. It is installed on the left side of the cabinet, adjacent to the air management module. Two jackscrews attach the NIC/PROC1 module to the MAU1 cabinet.
The NIC/PROC1 module has a backshell, which is attached to the module with two captive screws. On top of the backshell there is a hook that engages the cam bar of the MAU1 cabinet, thus the backshell cannot move.
The backshell also has an alignment pin that engages the keyway in the module to align the two components.

CONFIGURATION MODULE

The configuration module is installed in the NIC/PROC backshell connector and connects to the NIC/PROC module. The configuration module memorizes equipment data, rigging data and the primary identification data.



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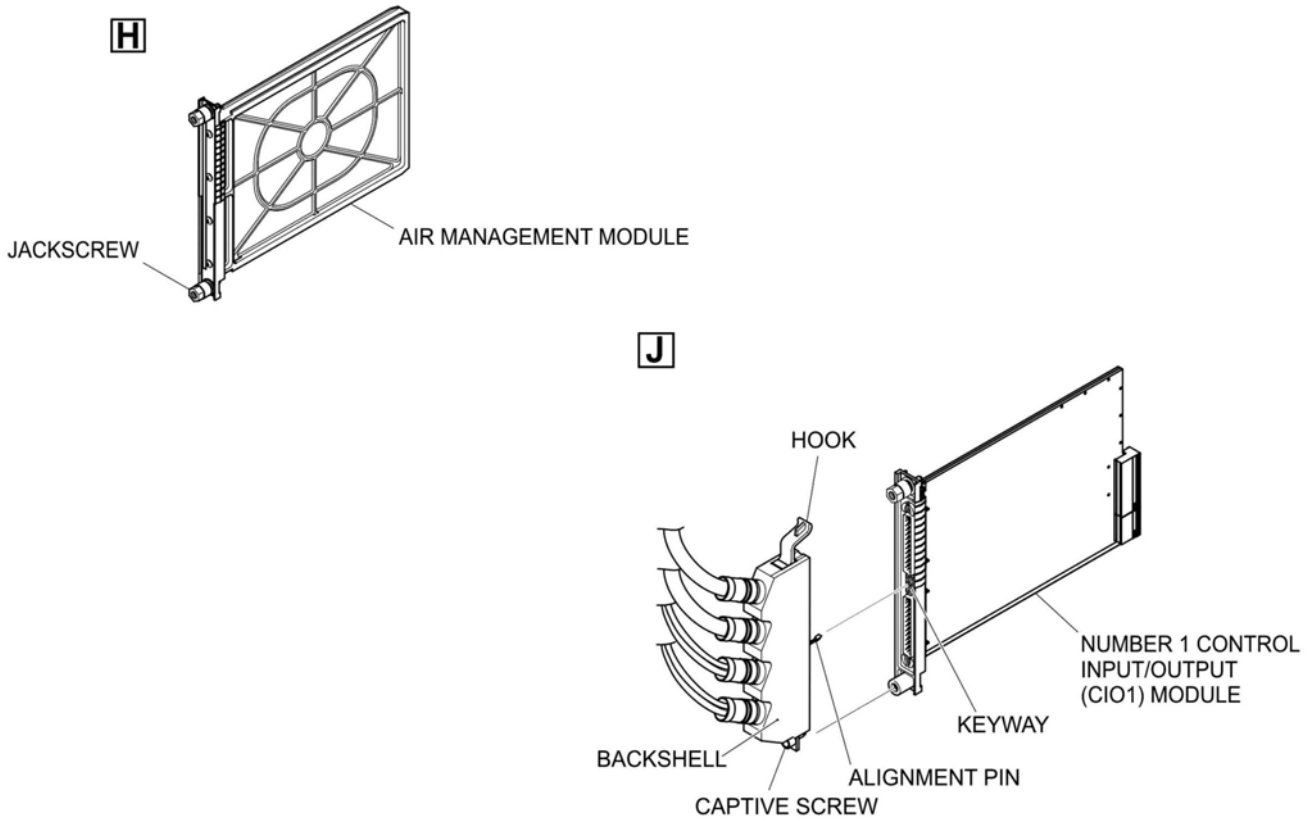
NUMBER 1 MODULAR AVIONIC UNIT (MAU 1) (SHEET 2 OF 4)

AIR MANAGEMENT MODULE

An Air Management Module (AMM) is installed in the MAU1 cabinet. It is installed on the right side of the MAU1 cabinet adjacent to the CIO1 module. Two jackscrews hold the AMM module in its slot in the MAU1 cabinet.

NUMBER 1 CONTROL INPUT/OUTPUT (CIO1) MODULE

The CIO1 module is installed in the MAU 1 cabinet. It is installed on the right side of the cabinet, adjacent to the CSIO1 module. Two jackscrews attach the CIO1 module to the MAU1 cabinet.
The CIO1 module has a backshell, which is attached to the module with two captive screws. On top of the backshell there is a hook that engages the cam bar of the MAU1 cabinet, thus the backshell cannot move.
The backshell also has an alignment pin that engages the keyway in the module to align the two components.



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NUMBER 1 MODULAR AVIONIC UNIT (MAU 1) (SHEET 3 OF 4)

NUMBER 1 POWER SUPPLY (PS1) MODULE

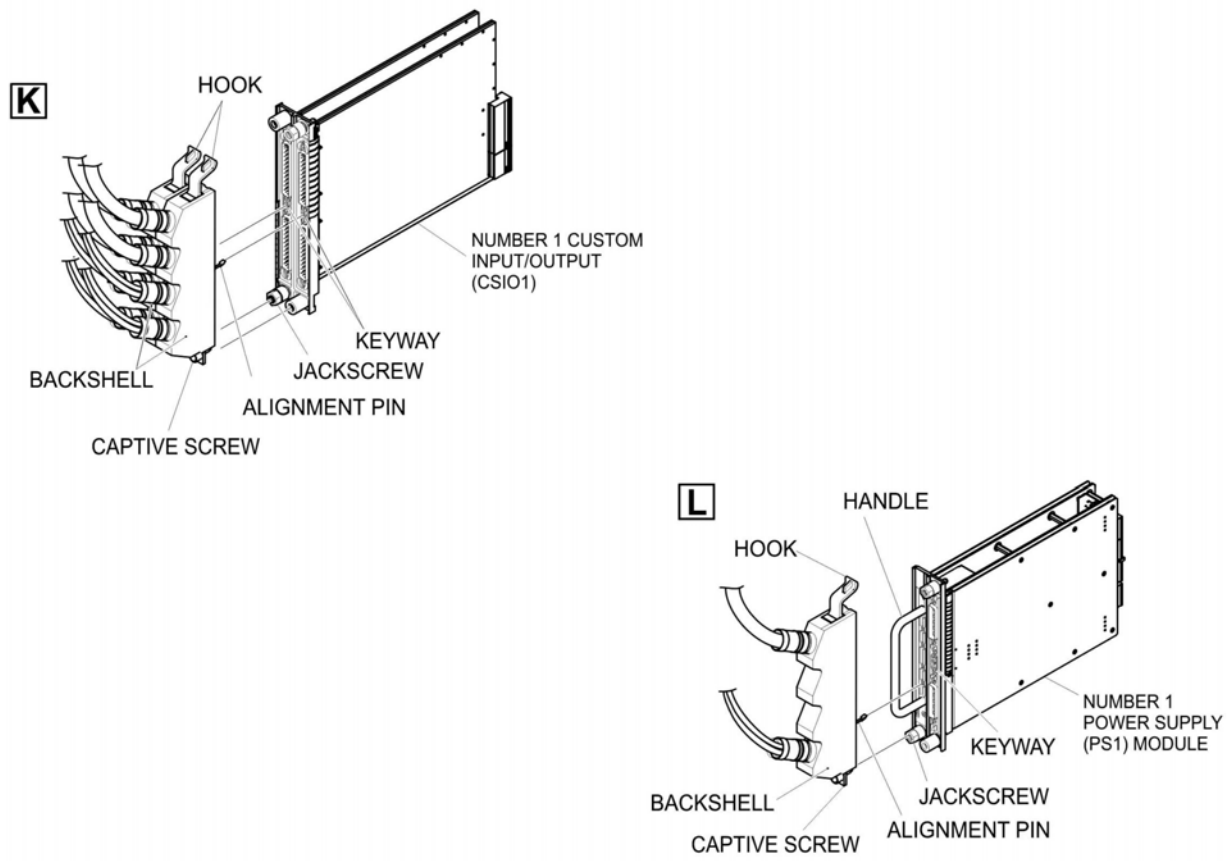
The PS1 module is installed in the MAU1 cabinet, on the right side. Two jackscrews hold the PS1 in its slot.
The PS1 module has a backshell, which is attached to the module with the two captive screws. On top of the backshell there is a hook that engages the cam bar of the MAU1 cabinet, thus the backshell cannot move.
The backshell also has an alignment pin that engages the keyway in the module to align the two components.
A handle is installed on the module front to make its installation and removal easier.

NUMBER 1 CUSTOM INPUT/OUTPUT (CSIO1) MODULE

The CSIO module is installed in the MAU1 cabinet. It is installed on the right side of the cabinet, adjacent to PS1 module.
Four jackscrews attach the CSIO1 module to the MAU1 cabinet.
The CSIO1 module has two backshell, which are attached to the module with four captive screws.
On top of the backshell there is a hook that engages the cam bar of the MAU1 cabinet, thus the backshell cannot move.
Each backshell also has an alignment pin that engages the keyway in the module to align the two components.

NUMBER 1 FAN ASSEMBLY

The Number 1 fan-assembly is installed in the rear side of the MAU1 cabinet.
Eighteen screws attach the Number 1 fan-assembly to the housing assembly.
Two electrical connectors connects the Number 1 fan-assembly to the MAU1 cabinet. The PS1 module energizes the Number 1 fan assembly.



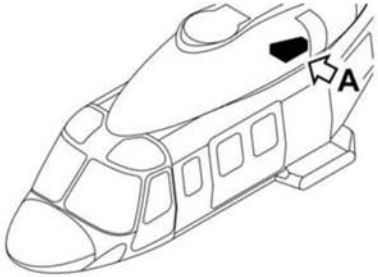
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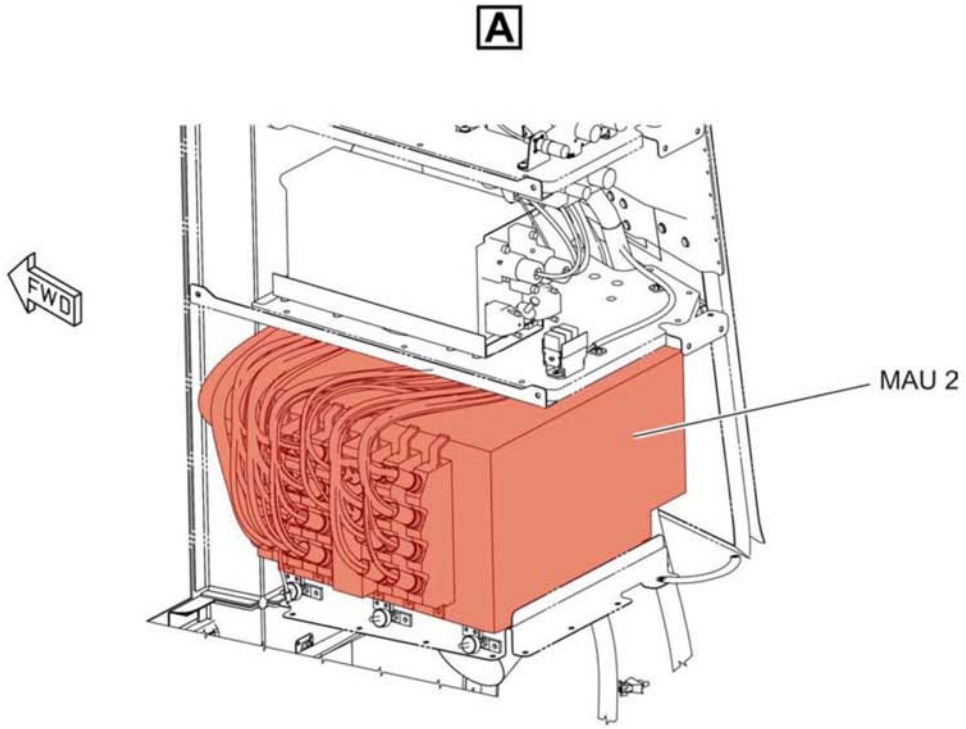
NUMBER 1 MODULAR AVIONIC UNIT (MAU 1) (SHEET 4 OF 4)

MAU 2

The MAU 2 is installed in the right avionic bay (helicopters [1J])/right nose compartment (helicopters [1L]).
The Number 2 Modular avionics unit system (MAU2) system includes the primary components that follow:

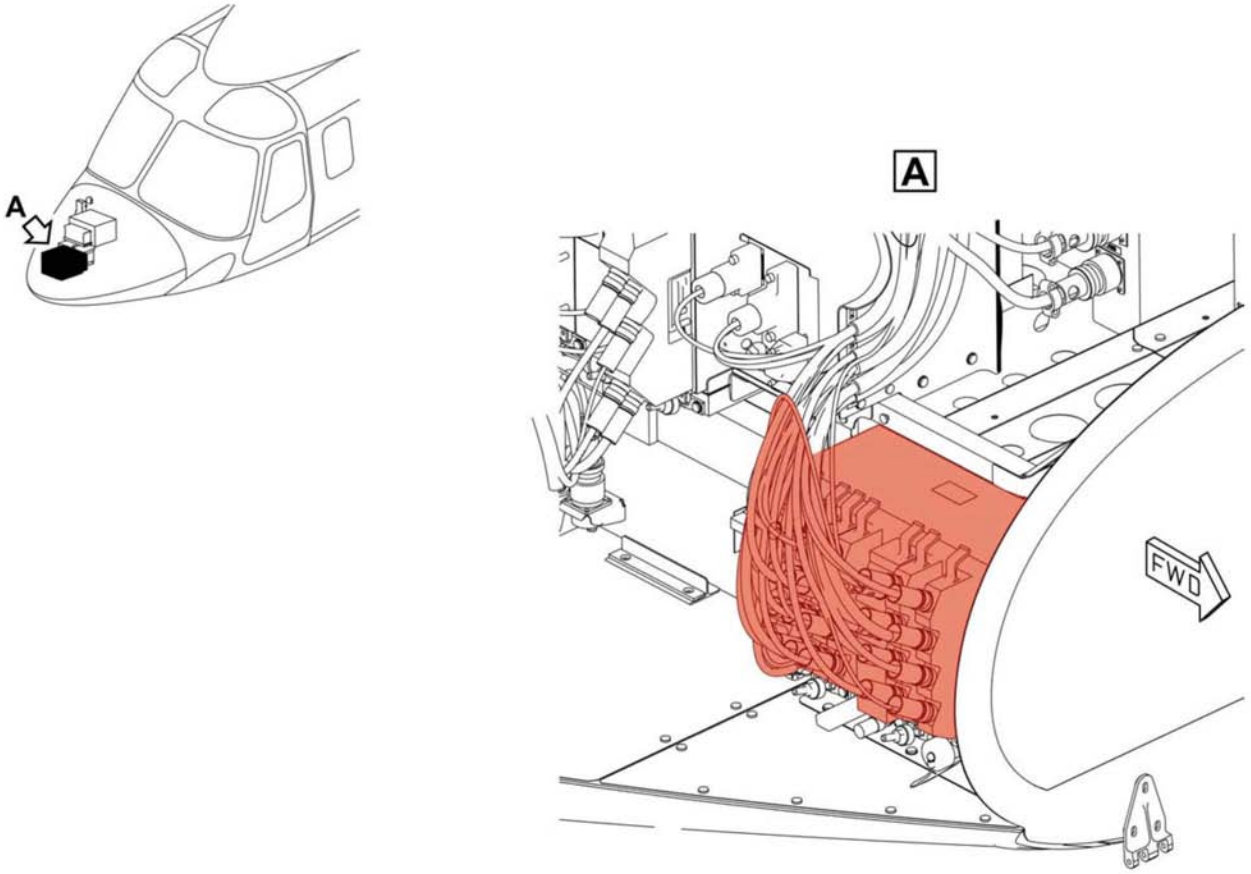
- The Number 2 Modular Avionic unit system (MAU2) cabinet
- The Number 2 Power Supply (PS2) module
- The Number 2 Custom Input/Output (CSIO2) module
- The Data base module (DBM)
- The Number 2 Network Interface Controller/processor (NIC/PROC2) module
- The Number 2 Actuator Input/output Processor (AIOP2) module
- The Number 4 Actuator Input/output Processor (AIOP4) module
- The Global Positioning System (GPS) (optional).





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SHORT NOSE – MODULAR AVIONIC UNIT (MAU 2) INSTALLATION



ICN-39-A-314000-G-A0126-02950-A-01-1

LONG NOSE – MODULAR AVIONIC UNIT (MAU 2) INSTALLATION

NUMBER 2 MODULAR AVIONIC UNIT SYSTEM (MAU2) CABINET

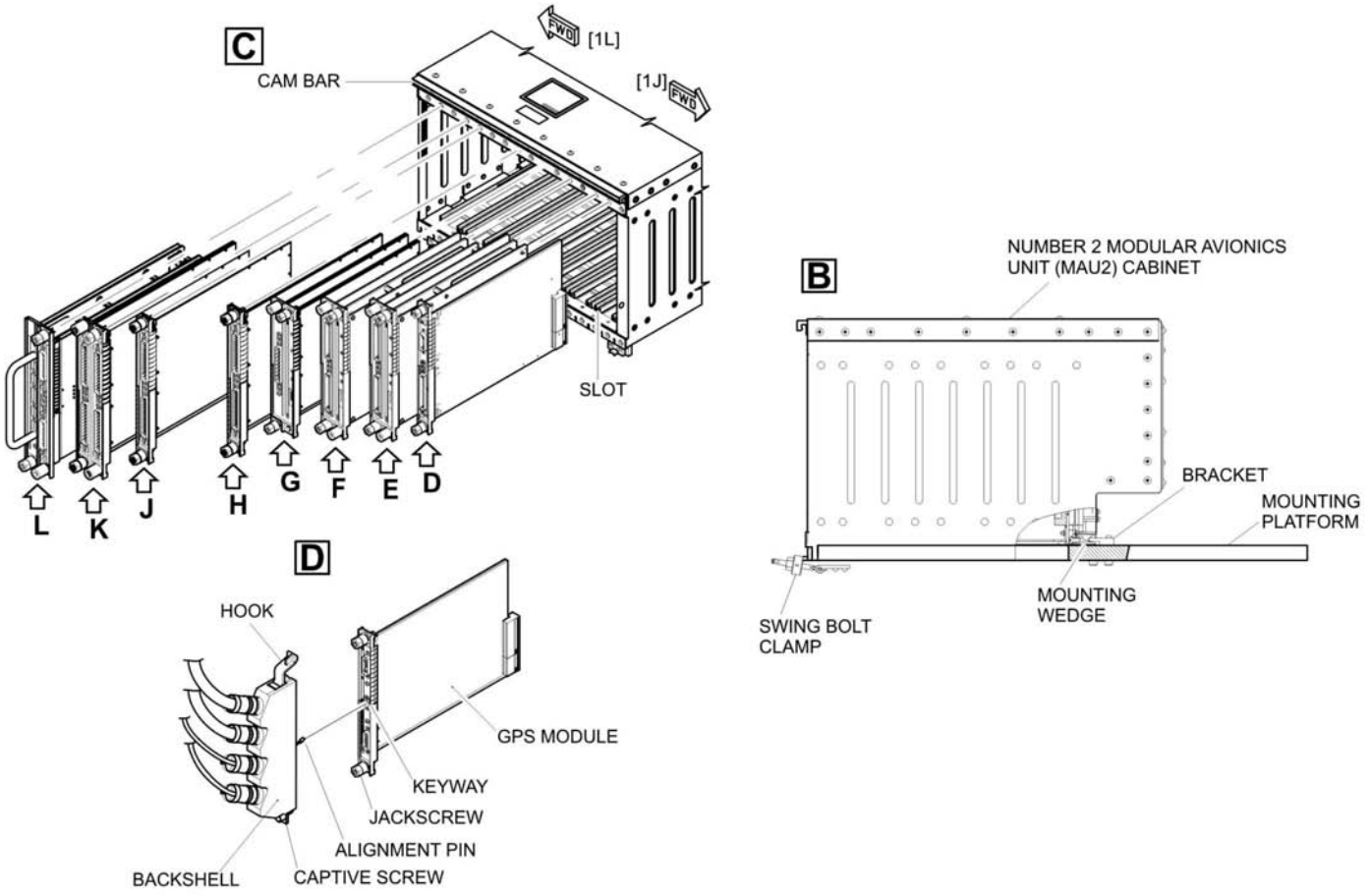
The MAU2 cabinet is installed in the right avionic bay helicopters [1J] right nose compartment helicopters [1L].
The three swing bolts, clamps and a bracket attach the MAU2 cabinet to its mounting platform. The bracket holds the mounting wedge.
The MAU2 cabinet has slots to hold the modules that follow:

- The Power Supply module
- The NIC/PROC module
- The six User Modules.

The MAU2 cabinet also contains the back plane bus, slots, cam bar, the Number 2 fan assembly and the related support.

GLOBAL POSITIONING SYSTEM (GPS) MODULE

The GPS module is installed in the MAU2 cabinet. It is installed on the left side of the cabinet, adjacent to the AIOP4 module.
Two jackscrews hold the GPS module in its slot in the MAU2 cabinet.
The GPS module has a backshell, which is attached to the module with two captive screws. On top of the backshell there is a hook that engages the cam bar of the MAU2 cabinet, thus the backshell cannot move.
The backshell also has an alignment pin that engages the keyway in the module to align the two components.



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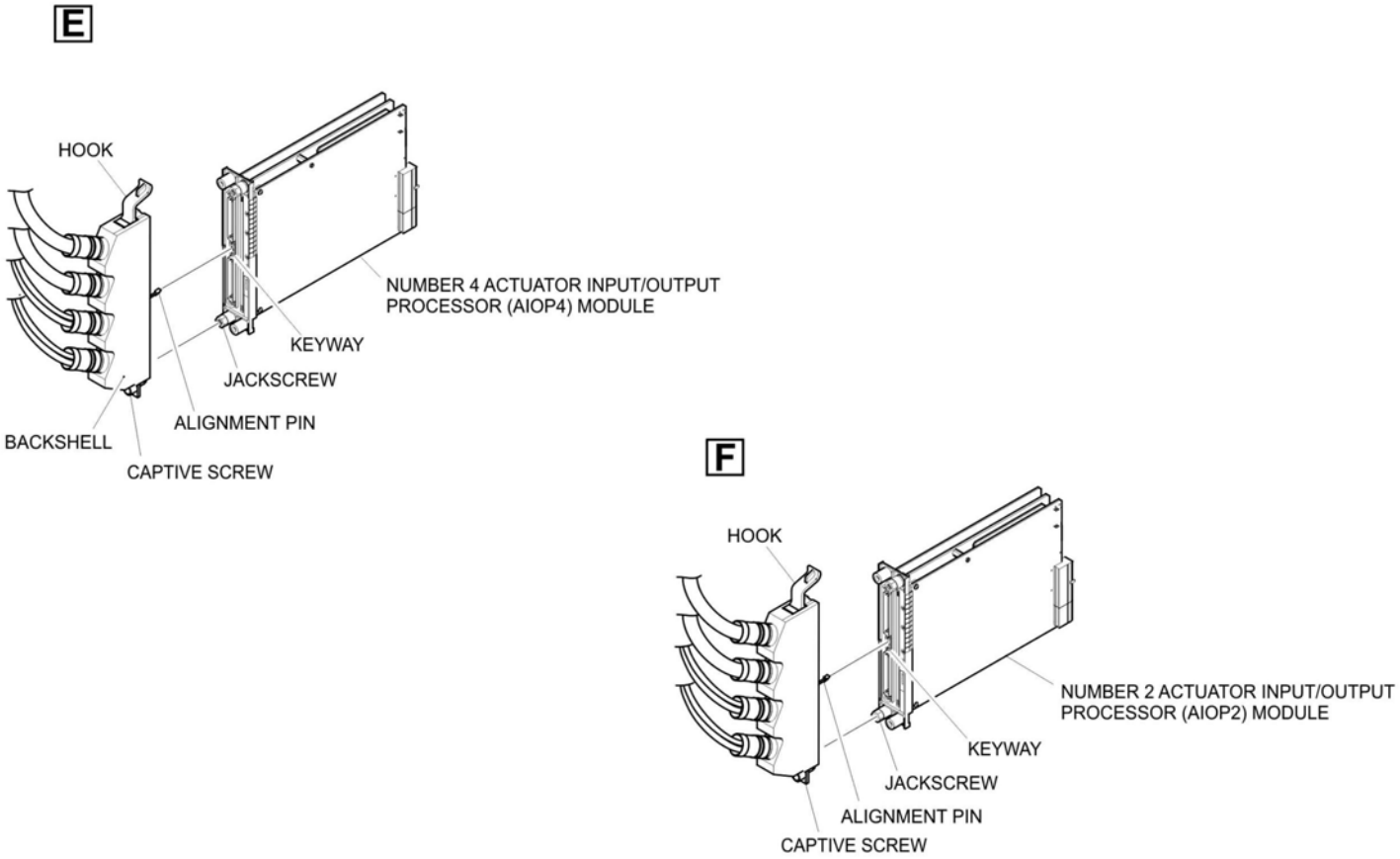
NUMBER 2 MODULAR AVIONIC UNIT (MAU 2) (SHEET 1 OF 4)

NUMBER 4 ACTUATOR INPUT/OUTPUT PROCESSOR (AIOP4) MODULE

The AIOP4 module is installed in the MAU2 cabinet. It is installed on the left side of the cabinet, adjacent to the AIOP2 module. Two jackscrews hold the AIOP4 module in its slot in the MAU2 cabinet.The AIOP4 module has a backshell, which is attached to the module with two captive screws. On top of the backshell there is a hook that engages the cam bar of the MAU2 cabinet, thus the backshell cannot move.The backshell also has an alignment pin that engages the keyway in the module to align the two components.

NUMBER 2 ACTUATOR INPUT/OUTPUT PROCESSOR (AIOP2) MODULE

The AIOPA2 module is installed in the MAU2 cabinet. It is installed on the left side of the cabinet, adjacent to the NIC/PROC2 module. Two jackscrews hold the AIOP2 module in its slot in the MAU2 cabinet.
The AIOP2 module has a backshell, which is attached to the module with two captive screws. On top of the backshell there is a hook that engages the cam bar of the MAU2 cabinet, thus the backshell cannot move.The backshell also has an alignment pin that engages the keyway in the module to align the two components.



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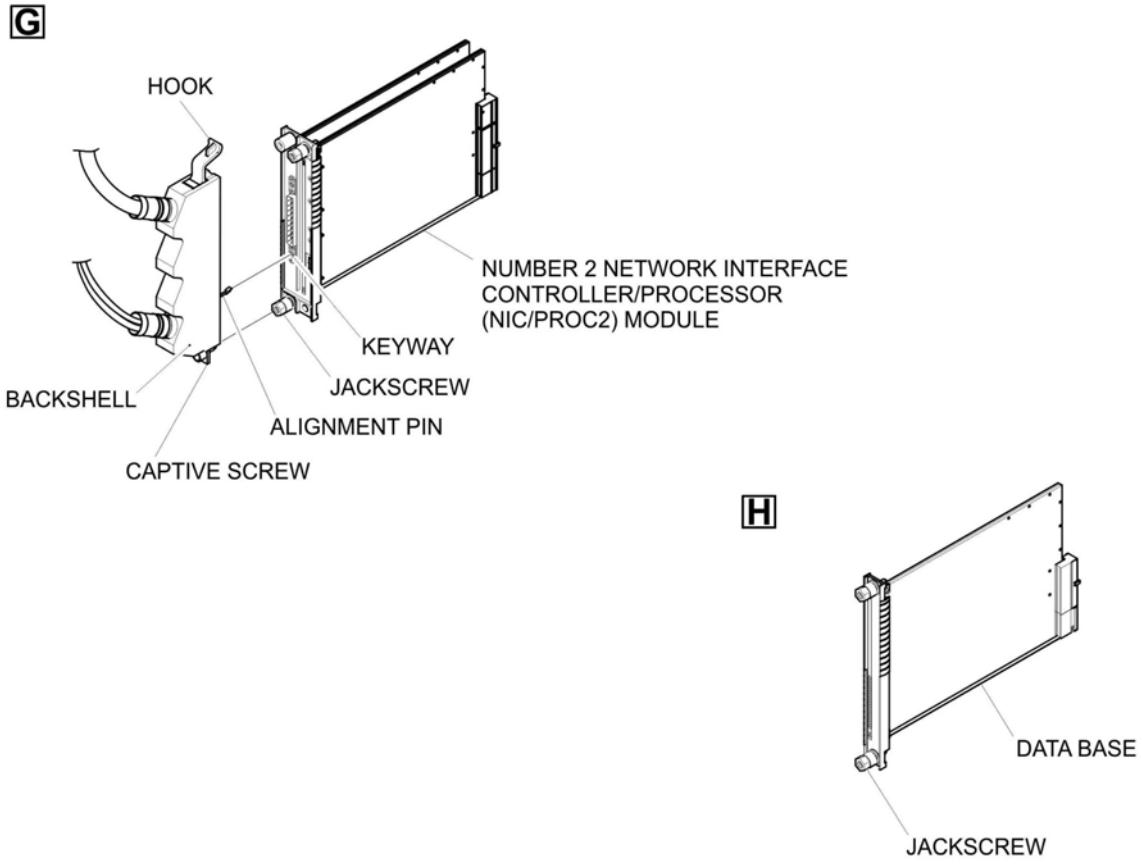
NUMBER 2 MODULAR AVIONIC UNIT (MAU 2) (SHEET 2 OF 4)

NUMBER 2 NETWORK INTERFACE CONTROLLER/ PROCESSOR (NIC/PROC2) MODULE

The NIC/PROC2 module is installed in the MAU2 cabinet. It is installed on the left side of the cabinet, adjacent to the database module. Two jackscrews attach the NIC/PROC2 module to the MAU2 cabinet.
The NIC/PROC2 module has a backshell, which is attached to the module with two captive screws. On top of the backshell there is a hook that engages the cam bar of the MAU2 cabinet, thus the backshell cannot move.
The backshell also has an alignment pin that engages the keyway in the module to align the two components.

DATA BASE MODULE

The data base module is installed in the MAU2 cabinet. It is installed on the right side of the cabinet, adjacent to the CIO2 module. Two jackscrews attach the data base module to the MAU2 cabinet.



ICN-39-A-314000-G-A0126-01406-A-01-1

NUMBER 2 MODULAR AVIONIC UNIT (MAU 2) (SHEET 3 OF 4)

NUMBER 2 CONTROL INPUT/OUTPUT (CIO2) MODULE

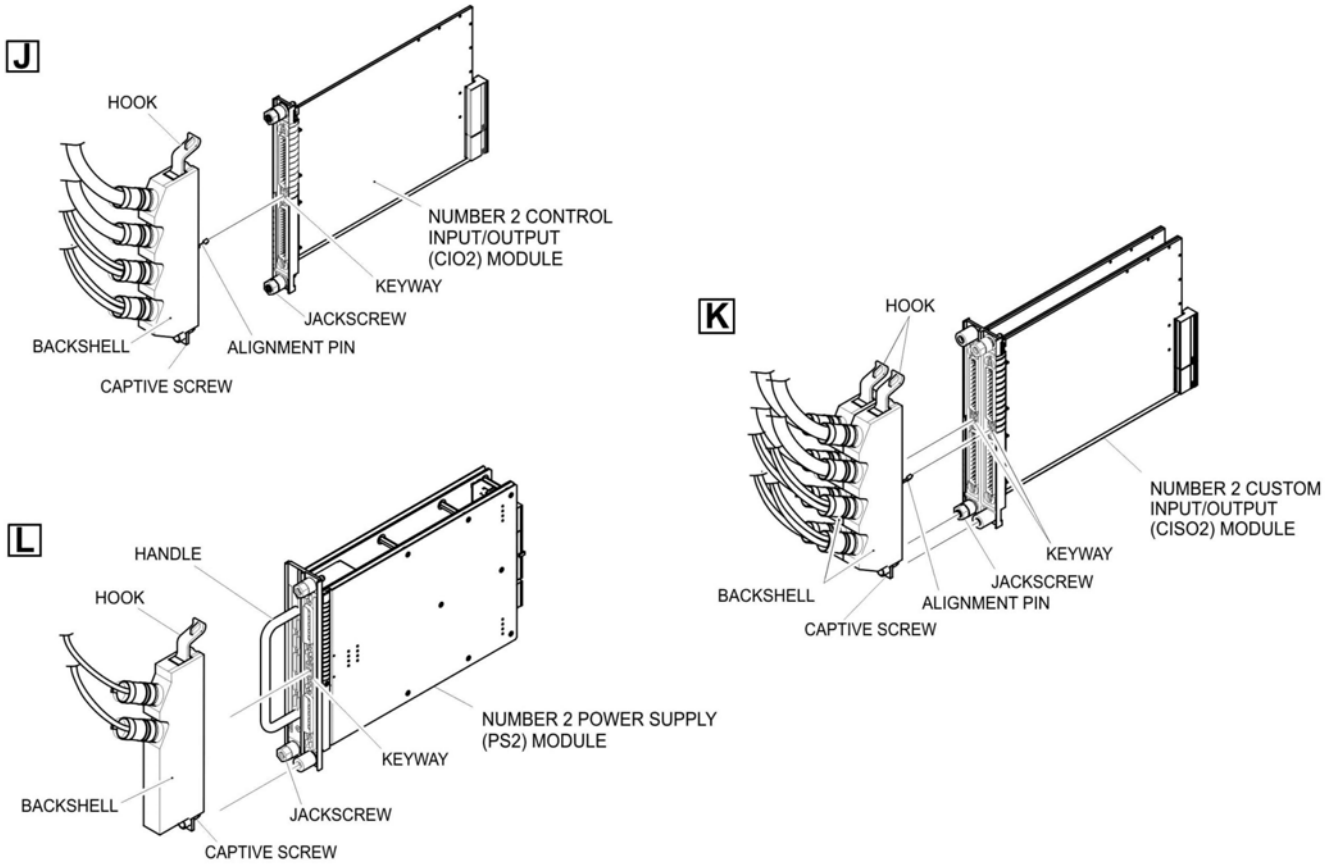
The CIO2 module is installed in the MAU2 cabinet. It is installed on the right side of the cabinet, adjacent to the CSIO2 module. Two jackscrew attach the CIO2 module to the MAU2 cabinet.
The CIO2 module has a backshell, which is attached to the module with two captive screws. On top of the backshell there is a hook that engages the cam bar of the MAU2 cabinet, thus the backshell cannot move.
The backshell also has an alignment pin that engages the keyway in the module to align the two components.

NUMBER 2 CUSTOM INPUT/OUTPUT (CSIO2) MODULE

The CSIO2 module is installed in the MAU2 cabinet. It is installed on the right side of the cabinet, adjacent to the PS2 module.
Two jackscrews attach the CSIO2 module to the MAU2 cabinet.
The CSIO2 module has two backshells, which are attached to the module with four captive screws. On top of each backshell there is a hook that engages the cam bar of the MAU2 cabinet, thus the backshell cannot move.
Each backshell also has an alignment pin that engages the keyway in the module to align the two components.

NUMBER 2 POWER SUPPLY (PS2) MODULE

The PS2 module is installed in the right side of the MAU2 cabinet. Two jackscrews hold the PS2 in its slot.
The PS2 has a backshell, which is attached to the module with two captive screws. On top of the backshell there is a hook that engages the cam bar of the MAU2 cabinet, thus the backshell cannot move.
The backshell also has an alignment pin that engages the keyway in the module to align the two components.A handle is installed on the module front to make its installation and removal easier.

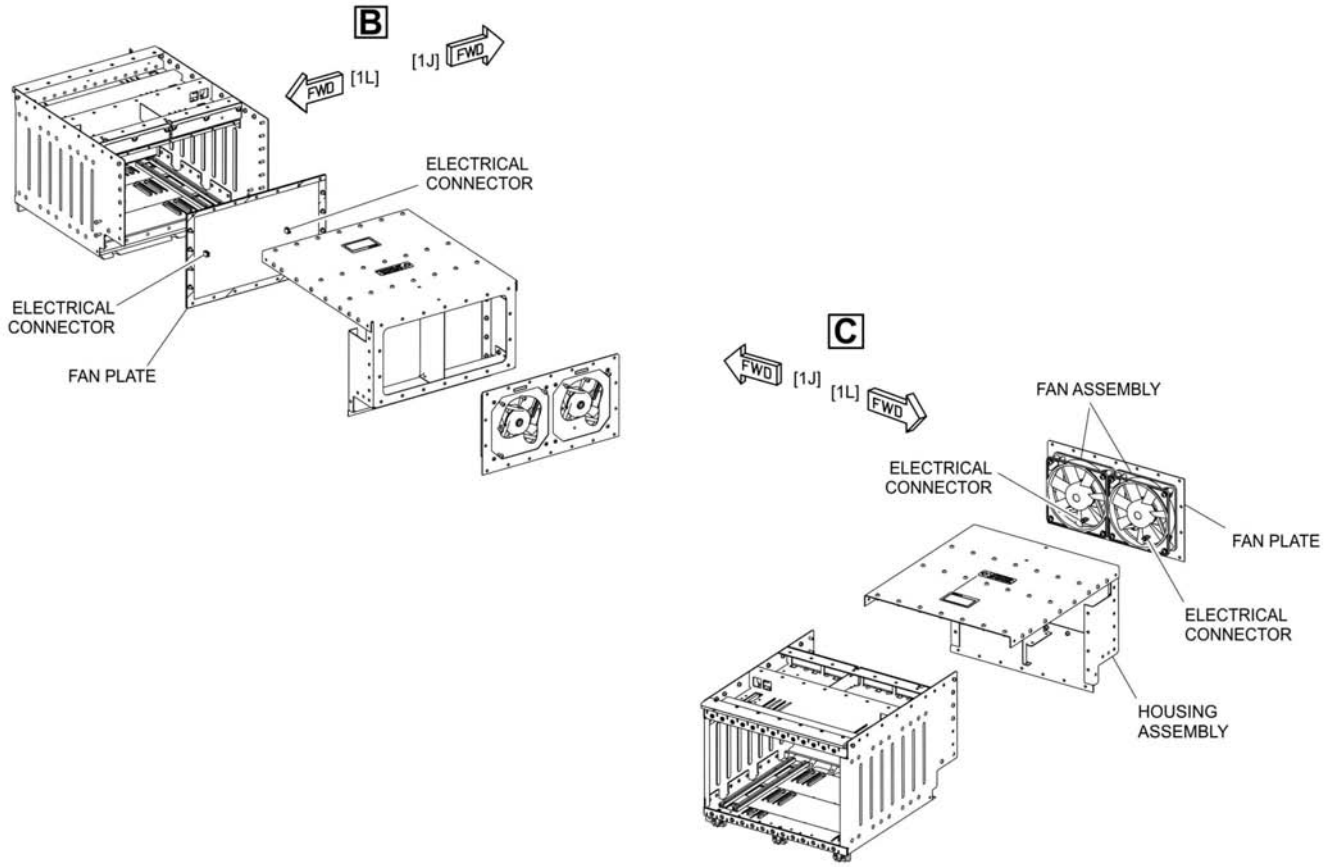


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NUMBER 2 MODULAR AVIONIC UNIT (MAU 2) (SHEET 4 OF 4)

NUMBER 2 FAN ASSEMBLY

The Number 2 fan-assembly is installed in the rear side of the MAU2 cabinet. Eighteen screws attach the Number 2 fan-assembly to the housing assembly. Two electrical connectors connects the Number 2 fan-assembly to the MAU2 cabinet. The PS2 module energized the Number 2 fan assembly.



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FAN ASSEMBLY

OPERATION

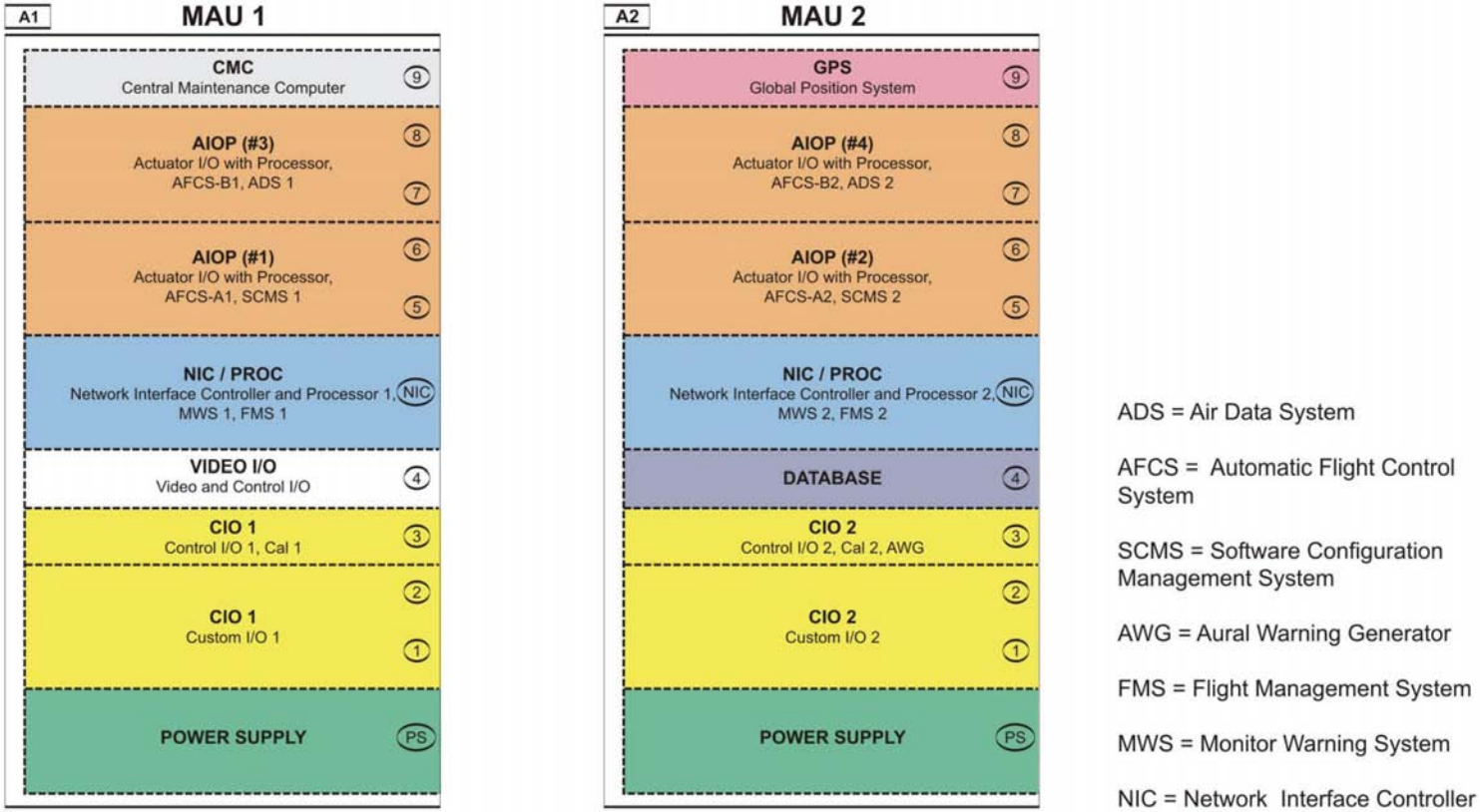
The Modular Avionics Unit system (MAU) transmits and receives data through the ASCB-D bus and the LAN bus. The Custom I/O (CSIO), Control I/O (CIO) and the other modules send the data from the sensors and the systems to the processor modules. The processor modules give at output the data to control and monitor the helicopter.
The most important module in the MAU is the NIC/PROC (Network Interface Controller and Processor) module which:

- Handles, processes and distributes all the data via the ASCB D
- Provides the Monitoring Warning Function
- Houses the Flight Management System (FMS) software

Failure of the NIC/PROC module causes the total failure of the MAU that is annunciated by the 1(2) MAU caution message in the bottom left corner of both PFDs.
Interface with aircraft systems, controllers and sensors is provided by two modules in each MAU:

- CIO (Control Input/Output) module
- CSIO (Custom Input/Output) module

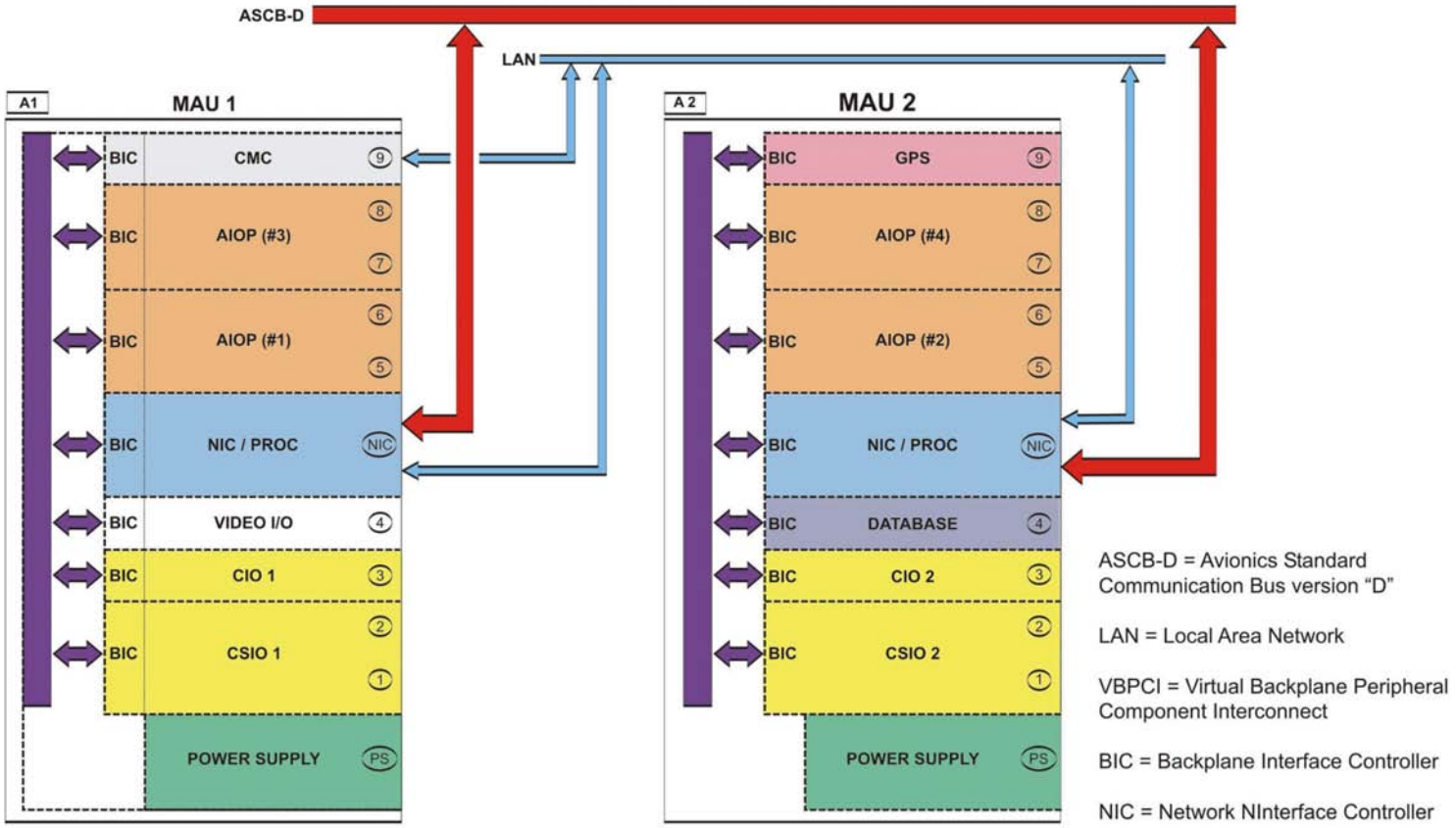
Only the CSIO module in MAU 2 provides Aural Warning Generator function (see chapter 31-50).
Each MAU also contains two AIOP (Actuator Input/Output with Processor) modules: they provide AFCS functions (see chapter 22 00).
The GPS module and the DATABASE module are only provided in MAU 2 (see chapter 34 00).
The CMC (Central Maintenance Computer) module and the optional VIDEO I/O module are only provided in MAU 1.



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MODULAR AVIONIC UNIT SYSTEM (SHEET 1 OF 2)

The MAU system is a hardware cabinet that contains the line replaceable modules (LRM). The LRMs connect to the avionics-standard communications bus, version D (ASCB-D). The ASCB-D network interchanges data between the primary-system computers. All modules are connect to the high-speed parallel backplane bus in the MAU. Only the PS and GPS modules are not connect to backplane bus. The GPS module only receives power and serial data from the backplane bus. To transmit data between the ASCB-D and the internal modules, the MAU channel uses a NIC/PROC module. The NIC/PROC supplies a high-integrity, high availability procedure for connection of the MAU and the displays to the ASCB-D. Each NIC/PROC connects to three of the four ASCB-D buses. Each NIC/PROC transmits and receives on one onside-primary and one onside-backup ASCB-D bus and receives only on the cross-side-primary ASCB-D bus. The NIC/PROC modules in the MAU connect to a configuration module to receive configuration data.
Each module uses the BIC to connect to the backplane bus in the MAU.
The NIC/PROC module includes a LAN. The LAN connects to the CMC for system maintenance and to the RMT for software and database updates. The LAN gives the access to tests tools. The MAU has the input/output (I/O), processing, and data base storage modules. These modules are linked to the Avionics Standard Communication Bus (ASCB) of the Network Interface Controller with the Processor (NIC/PROC) module and Power Supply Module (PSM).
Each module has a standard interface (the Backplane Interface Controller (BIC)) to supply data to the MAU.
The MAU sends and receives data through the ASCB-D and the Local Area Network (LAN).



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MODULAR AVIONIC UNIT SYSTEM (SHEET 2 OF 2)

POWER SUPPLY (PS) MODULE

The PS module energizes the MAU with 28 V DC power. Two 28 V DC inputs are available for each MAU, by aircraft power buses.

CUSTOM INPUT/OUTPUT (CSIO) MODULE

The CSIO module uses I/O conversion routines to change external I/O data to and from the ASCB-D network. The CSIO module uses two MAU slots. The module is only connected to one of the backplanes in the two slots. The main circuit card in the CSIO is a generic I/O card that supplies ARINC 429, discrete, and analog I/O interfaces.

- The generic I/O card has the inputs that follows:
- The ground/open discrete input
- The single-ended DC analog input
- The differential DC analog input
- The thermistor/resistive thermal device input
- The potentiometer Input.

The generic I/O card also has the interfaces that follow:

- The multiple discrete Inputs (ground/open and/or 28 V DC/open)
- The multiple ARINC 429 receivers (high or low speed)
- The multiple ground/open, 200 mA discrete inputs.

The second circuit card in the CSIO module has more inputs and custom interfaces than those available on the generic I/O card. The second circuit card is connected to the generic I/O card through a mezzanine connector. The processor on the generic I/O card controls the hardware on the custom I/O card.

- The custom I/O card has the circuits that follow:
- The thermocouple inputs
- The RS-422 receivers
- The low-level DC inputs
- The tachometer inputs
- The discrete inputs
- The AC analog inputs
- The Linear Variable Displacement Transformer (LVDT) inputs.

CIO MODULE

The CIO module is an interface for the external I/O data on the ASCB-D. The CIO module is an interface for the system display controllers, cursor control devices (CCD) and the multifunction control display units (MCDU). The CIO2 module also has an aural warning circuitry that cause warning to be heard in the cockpit.
The module also supplies the interfaces that follow:

- The RS-422 receivers
- The RS-422 transmitters
- The RS-232 receiver
- The RS-232 transmitter
- The ARINC 429 receivers
- The analog audio generator with two buffered outputs (for aural warning)
- The ground/open discrete input
- The 28 V DC/open discrete input (for aural warning)
- The versatile discrete inputs (any combination of ground/open or 28 V DC/open)
- The ground/open discrete outputs (500 mA each)

- The 28 V DC/open discrete output (used for guidance controller)
- The +5 V DC/ground power output (used for guidance controller).

AIR MANAGEMENT MODULE

The air management module is a card without data on it that has a front plate and is attached to the MAU with jackscrews. This module is installed in each empty slot of the MAU to help control the airflow to the modules.

DATA BASE MODULE

The database module is installed in the MAU2 cabinet It has a central area that stores the data. It also stores a backup copy of the MAU flight software and of the maintenance database. The backup copies let the software be installed when an LRM is replaced. The database module uses a solid-state non-volatile memory.

NETWORK INTERFACE CONTROLLER WITH PROCESSOR (NIC/PROC)

The NIC/PROC transmits and receives data from the ASCB-D and the LAN, and makes them available to the other modules in the MAU. The data are made available through a special Peripheral Component Interconnect (PCI) backplane that has this name: Virtual back Plane Peripheral Component Interconnect (VbPCI) bus. The VbPCI is a 32-bit wide, passive, parallel backplane bus. The system uses this bus to connect the NIC to all the modules in the MAU on the ASCB-D. All module data in these units (I/O included) are available to all other modules or to the ASCB-D in the integrated avionics system.
A standard hardware interface the name of which is Backplane Interface Controller (BIC) supplies the connection to the MAU VbPCI. The BIC is installed on the MAU modules. The BIC stores the ASCB-D and LAN data it receives from the NIC, and sends them to the modules in sequence. But the transmission occurs only when the modules are prepared to receive the data. The hardware mechanism that stores, transmits and receives the data in sequence is the BIC frame buffer. The BIC frame buffer has dual ports and uses Random Access Memory (RAM). The NIC and the MAU can read and write to the BIC frame buffer.
To decode and encode the data to and from the BIC, the MAU modules use a software function of the module processing circuit. This is a standard function the name of which is Periodic Device Driver (PDD). The PDD is a high-integrity, table-driven routine. It applies ASCB-D data to software functions available on the modules that are synchronized with the ASCB-D network.
The NIC/PROC modules supply the interface between the internal MAU backplane, the ASCB-D, and the LAN.
The NIC/PROC controls all the data interchanged between the modules and the ASCB-D/LAN. The NIC/ PROC module uses two slots in the MAU cabinet. The NIC/PROC module does the functions of a processor module. The primary function of the NIC/PROC is to let the MAU modules access the ASCB-D and the LAN.
The NIC/PROC also:

- Synchronizes the ASCB-D network
- Broadcasts System Configuration
- Supplies date/time to the System
- Does a check of channel configuration
- Monitors the power module when power is applied
- Monitors MAU fan operation.

The processor of the NIC/PROC module uses a digital engine operating system (DEOS). The NIC/PROC supplies partitioned space and time resources for multiple functions as those of:

- The flight management system

- The monitor warning system.

ACTUATOR INPUT/OUTPUT PROCESSOR (AIOP) MODULES

The AIOP module do the functions of the Automatic-Flight Control-System (AFCS) and have interfaces with:

- The six linear actuators
- The trim actuators
- The position sensors

Each AIOP module has an interface with three of the six linear actuators. The interfaces are one for the pitch axis, one for the roll axis and one for the yaw axis. The AIOP module also has an interface with three (or four) rotary actuators. The three rotary actuators include pitch, roll, and yaw trim actuators with an option for collective parallel actuator. The pulse with modulation PWM controls the rotary actuator.
The AIOP module is a two-slot module with one connection to the backplane. The module has an I/O card and a processor card. The I/O card contains no software and the processor card communicates directly with the I/O card through the Peripheral Component Interconnect (PCI) local bus. The primary use for the I/O is for control of actuators, excitation/demodulation of position sensors, and control of functional enable-and-disable control lines.
The I/O parts of the AIOP modules contain the circuits that follow:

- ARINC 429 receivers
- Discrete I/Os
- AC sensor excitation circuitry for LVDT and resolver type sensors
- Analog input processing circuits, and AC demodulation circuits
- Control Area Network (CAN) buses to interface with the linear actuators
- PWM circuits for trim actuator functions.

CENTRAL MAINTENANCE COMPUTER MODULE (CMC)

The CMC module receives the fault data from part systems and transmits commands to them. The fault data are store in a fault-history database on the CMC module. The CMC has different maintenance modes that the user sets through the Remote Maintenance Terminal (RMT).
The CMC functions are as follow:

- To transmit data to the central display system (CDS) and to the RMT
- To transmit data to the central display system (CDS)
- To do the built-in test (BIT)
- To download the fault-data memory of the member systems
- To give the serial numbers and part numbers of the member systems
- To monitor the continuous BIT
- To monitor the parameters for the rigging tests.

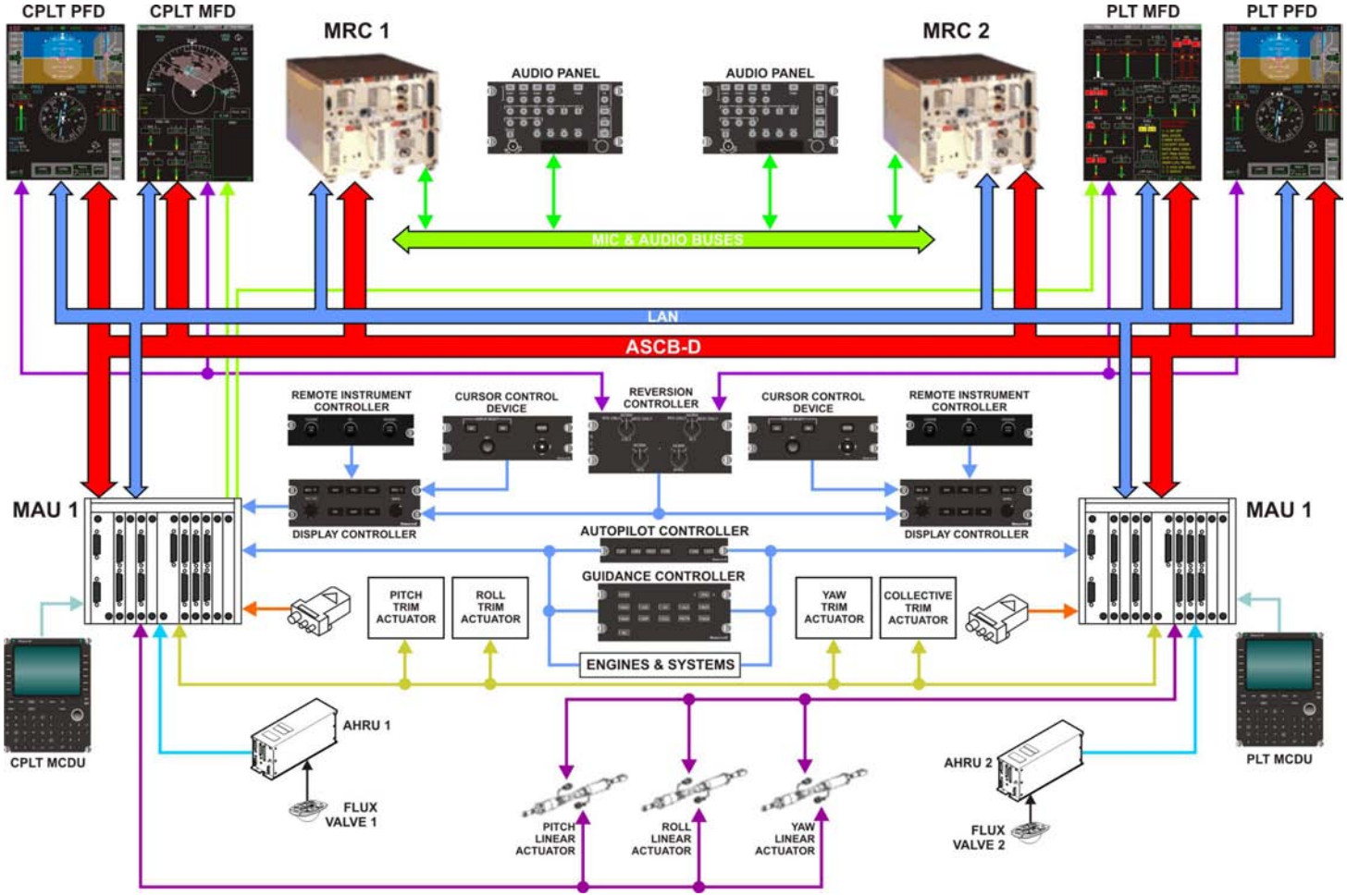
The data loading function loads the program software and data into storage devices or downloads the fault-history data. The RMT is connected to the LAN when necessary.

GLOBAL POSITIONING SYSTEM (GPS)

The GPS module is installed in the MAU2 cabinet. The GPS module is a stand-alone satellite navigation sensor. The GPS module received GPS satellite signals with the input transmit power (RF port) of -121.0 (minimum) and -86.0 dBm (maximum).

FAN ASSEMBLY

The MAU has the fan assembly. It blows cool air on the modules to decrease their temperature. The two cooling fans receive power from the PS module.



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PRIMUS EPIC SYSTEM BLOCK DIAGRAM

CAPTIONS

A temperature switch triggers the 1(2) MAU OVHT caution message in the CAS if the MAU temperature exceeds a set value following dual fan failure in an MAU.

Failure of the NIC/PROC module causes the total failure of the MAU that is annunciated by the 1(2) MAU caution message in the bottom left corner of both PFDs.
The failure messages, relevant to the Central Computer system, shown in the CAS messages window are as following:

CAS CAUTION MESSAGES			
CAS CAPTION	**FAILURE DESCRIPTION**	**PROCEDURE NAME**	**AW139-RFM-4D**
1(2) MAU OVHT	Associated MAU overheat	MODULAR AVIONICS UNIT OVER-HEAT / FAIL	Section 3 EMERGENCY AND MALFUNCTION PROCEDURES
SYS CONFIG FAIL	Hardware or software configuration problem (displayed on ground only)	SYSTEM CONFIGURATION FAILURE	
SYS CONFIG FAIL	Hardware or software changed, configuration validation operation required (displayed on ground only)	VALIDATE CONFIGURATION	

PFD CAUTION MESSAGES			
PFD CAPTION	**FAILURE DESCRIPTION**	**PROCEDURE NAME**	**AW139-RFM-4D**
1(2) MAU	Associated MAU failure	MODULAR AVIONICS UNIT OVERHEAT / FAIL	Section 3 EMERGENCY AND MALFUNCTION PRO-CEDURES

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SECTION 50 - CENTRAL WARNING SYSTEM

DESCRIPTION

The Central Warning System (CWS) supplies system alerts to the flight crew when unsatisfactory aircraft conditions occur. System alerts include Crew Alerting System (CAS) messages, visual indications, and aural warning messages. The system alerts are done by the Monitor Warning Function (MWF) installed in the both Modular Avionics Units (MAU).
The Central Warning System has the components that following:

- Network Interface Controller with Processor (NIC/ PROC) Modules supply the processing for the MWF. The NIC/PROC function supplies the interface between the Avionics Standard Communication Bus, version-D (ASCB-D) and the MAU backplane.
- Control Input/Output #2 (CIO 2) Module supplies the aural warnings output for the pilots Audio Panel that are generated by the AWG.
- Custom Input/Output (CSIO) Modules sense the discrete input warning/ caution malfunction that are processed by the MWF.

MONITOR WARNING FUNCTION (MWF)

The MWF is a software function in the NIC/PROC module. The MWF monitors the other helicopter systems continuously to send the pilot and copilot messages through the CAS display Primary Flight Display (PFD).
The MWF messages along with the CAS function supplies message logic and timing and lists the messages on the CAS display. The importance is identified by color codes. Each message shown with the colors that follow:

- Priority 1 – Warning Message (Red)
- Priority 2 – Caution Message (Amber)
- Priority 3 – Advisory Message (Green)
- Priority 4 – Maintenance Status Message (White)

The CAS message window has 12 lines with 18 characters for each line.
The warning messages (red) show at the top of the list. When a new warning message is active, it is displayed as a white text on red background (inverse video technique), until it is acknowledged. When acknowledged, via the master warning light pushbutton, it is displayed as red text on black background.
When a new caution message (amber) is active, it is displayed as a black text on amber background until it is acknowledged.
When acknowledged, via the master caution light pushbutton, it is displayed as amber text on black background.
When a new advisory message (green) is active, it is displayed as a black text on green background for the first 5 seconds. After this time, this message is automatically acknowledged and it is displayed steady-on green text on black background.
The MAINTENANCE status message is displayed steady-on white text on black background. Any unused message lines show blank spaces.
All messages show in the chronological sequence for each category of message. The newest message shows at the top, and the oldest message shows at the bottom of the display. The end of the message stack is indicated by a white message "END" being displayed.
The purpose is to indicate the existence of non-displayed caution or advisory or status messages.

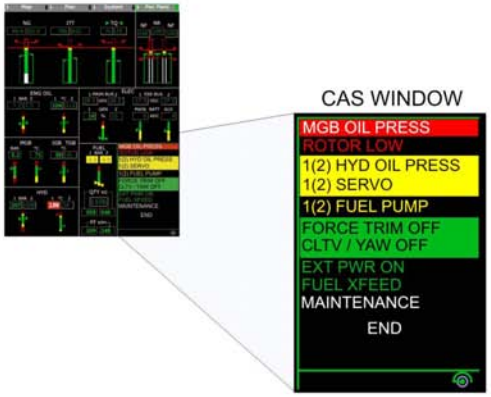
CREW ALERTING SYSTEM (CAS)

The CAS transmits warning, caution, advisory and status data that come in view on the Multi Function Display (MFD).
Thus the CAS lets the flight crew know about the conditions of the helicopter systems. If there is a warning, caution or condition, the CAS causes the master warning indicator on the master caution indicator to flash.

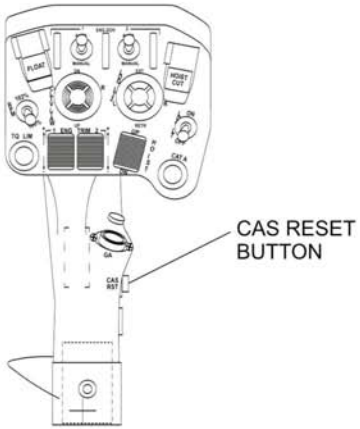
The CAS also sends the warning and caution signals to the audio control panels and the related aural tone must heard in the cockpit.

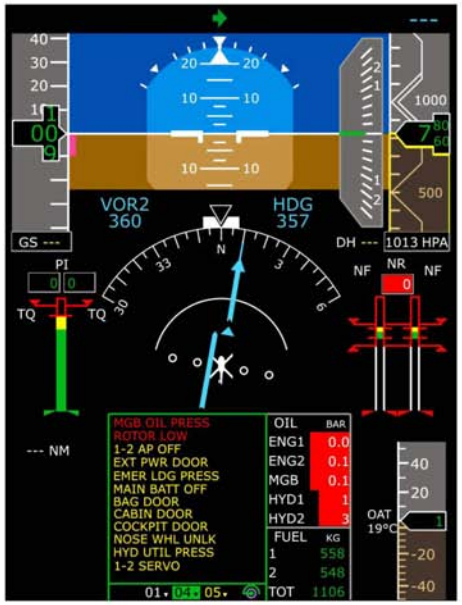
CAS MESSAGE SCROLL

The warning messages can not be scrolled out of view. The number of warning messages has a limit of the total number of display lines.
The CAS messages of lower criticality (caution, advisory and status) for which display space is not available can be scrolled if all displayed warning and caution message have been acknowledged.
The pilot and copilot can use the relative Cursor Control Device (CCD) to scroll up (CW) or down (CCW) the caution, advisory, and maintenance status messages out of view.
An out of view message display (at the bottom of the message list) shows how many messages are scrolled out of view.
All messages to be scrolled off must be acknowledged. When a new CAS message becomes active and CAS messages are scrolled off the display, the new message is placed at the top of its respective color stack and the entire CAS message list from that point down in priority is brought back into view automatically.
Thus, if all active messages are scrolled off the display and a new caution massage becomes active, the new caution message is placed at the top of all the active caution messages, and all active caution and advisory messages are brought into view, within the limits of the CAS display.
If the new message is an advisory message, it would be placed at the top of all active advisory messages and only all active advisory messages are brought into view within the limits of the CAS display.









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CENTRAL WARNING SYSTEM

CAS MESSAGES - PRIORITY AND FORMAT		
Priority **1 = WARNING** Messages	INITIAL DISPLAY WARNING NOTE A Warning messages are always in view (not scrol-lable)	AFTER ACKNOWLEDGEMENT WARNING NOTE B All Warning messages trigger an aural warning
Priority **2 = CAUTION** Messages	INITIAL DISPLAY CAUTION	AFTER ACKNOWLEDGEMENT CAUTION
Priority **3 = ADVISORY** Messages	INITIAL DISPLAY ADVISORY	AFTER FIVE SECONDS ADVISORY
Priority **2 = STATUS** Messages	MAINTENANCE	NOTE C On ground only
	END	

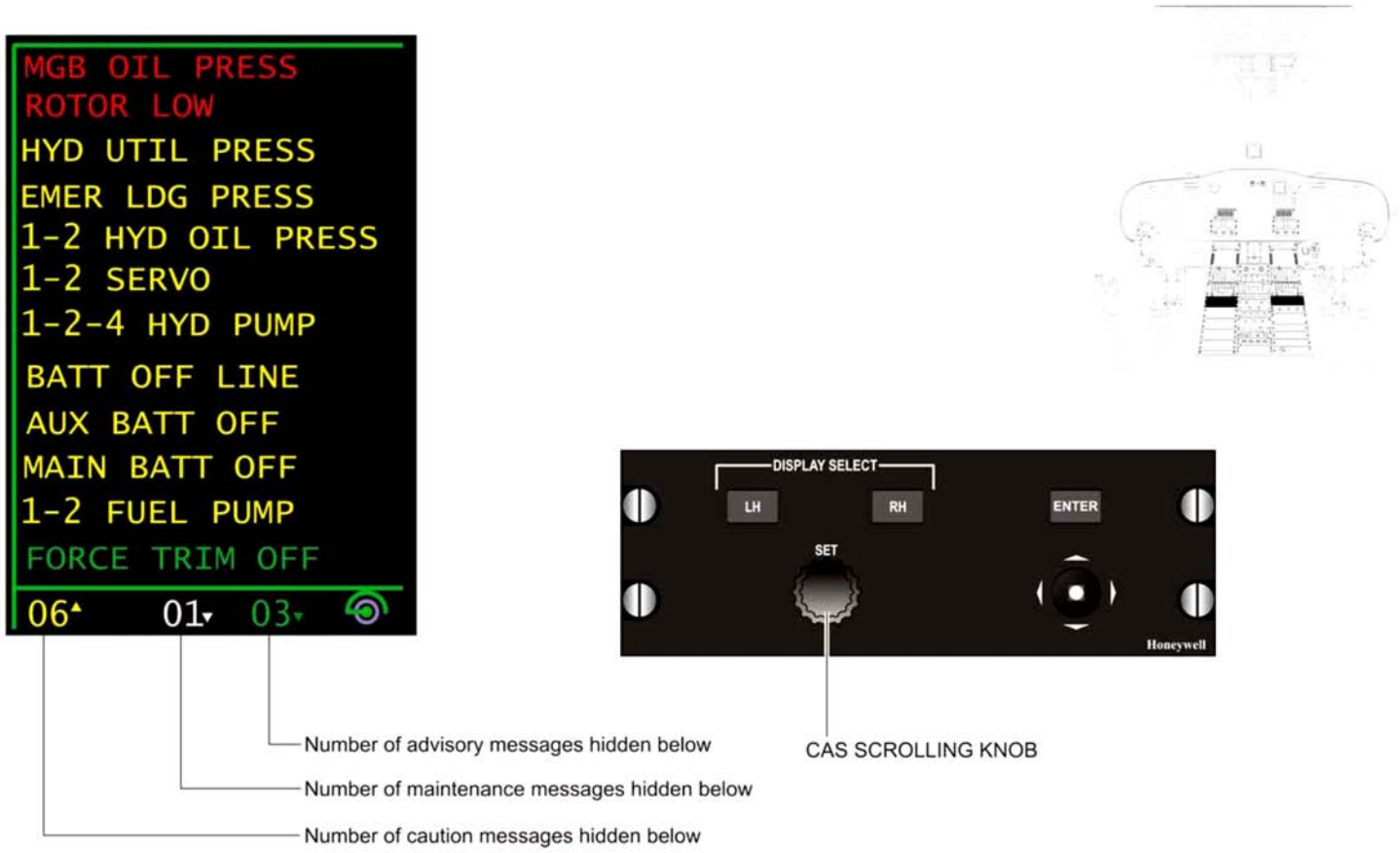
MASTER WARNING/CAUTION LIGHTS

When warning or caution alerts occur, the CAS function causes the master warning/caution lights to flash.
There are two master warning lights and two caution lights.
One set of master warning/caution lights are installed on the side of the pilot, and the other is installed on the side of the copilot. When a warning or caution alert occurs, the applicable light comes on to supply a visual indication to the pilot or copilot.
The master warning/caution lights flash until the pilot or copilot presses the light itself.
On the pilots collective sticks there is a CAS RST pushbutton that, when pressed, resets the warning or caution flashing lights.
The MWL and MCL are dimmable.

- The Master Warning Lights:.... when pressed, the pilot or copilot acknowledges the warning. The lights stop flashing and the aural message removed.
- The Master Caution Lights: … when pressed, the pilot or copilot acknowledges the caution and the light stop flashing.

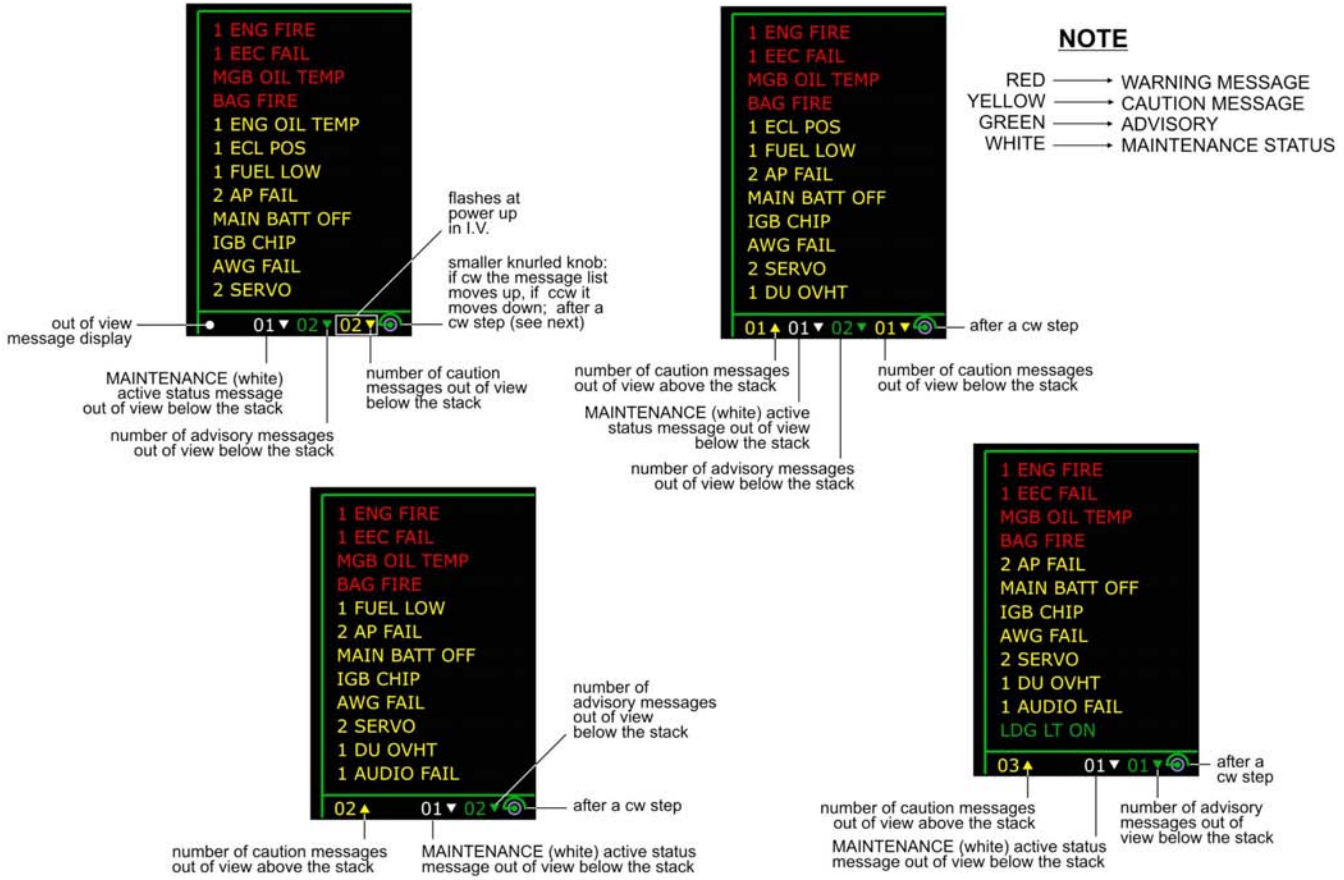
AURAL WARNING GENERATOR

The Aural Warning Generator (AWG) function supplies sounds (tones) and aural messages (voice) to alert the pilot and copilot about possible dangerous operations or flight conditions.
The aural warning generator hardware is inside the CIO 2 module in the MAU.
The necessary tones are digitized and kept in programmable memory in the system.
When a warning occurs, the MWF informs the AWG on the applicable tone and aural voice message to be generated. These ones are then routed to he pilots audio panel. The audio control panels have the volume controls that let the pilot or copilot adjust the volume of the incoming signal.



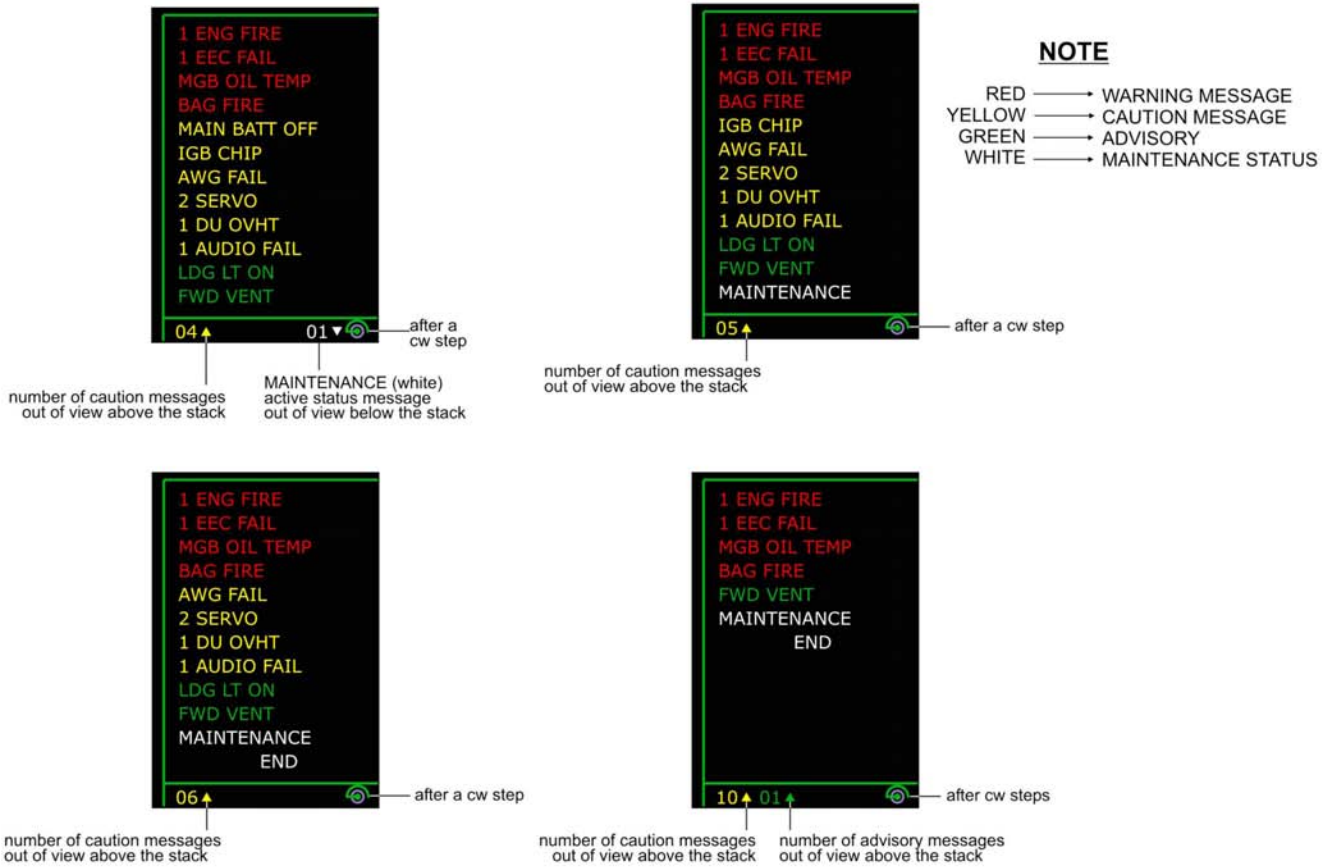
ICN-39-A-315000-G-A0126-02951-A-01-1

CAS WINDOW



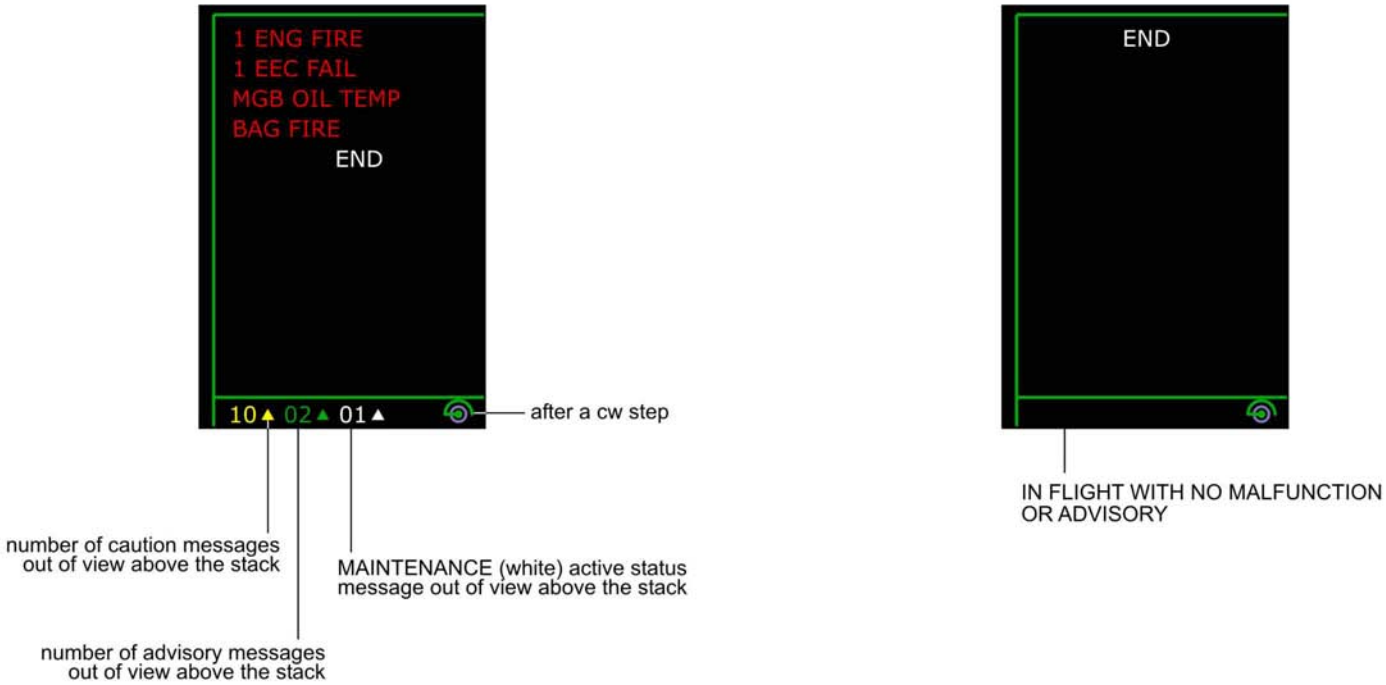
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CAS MESSAGE LIST PRESENTATION (SHEET 1 OF 3)



ICN-39-A-315000-G-A0126-01452-A-01-1

CAS MESSAGE LIST PRESENTATION (SHEET 2 OF 3)



ICN-39-A-315000-G-A0126-01453-A-01-1

CAS MESSAGE LIST PRESENTATION (SHEET 3 OF 3)

AURAL WARNING MESSAGES				
Priority	**Aural Message**	**No. of cycles**	**Cause**	
1	"ROTOR LOW - ROTOR LOW"	Continuous		
2	"ENGINE 1(2) OUT - ENGINE 1(2) OUT"	1		
3	"ENGINE 1(2) OUT - ENGINE 1(2) FIRE"	Continuous		
4	"ROTOR HIGH - ROTOR HIGH"	1		
5	"ENGINE 1(2) IDLE - ENGINE 1(2) IDLE"	1		
6	"WARNING - WARNING"	1	Any other warning message	
7	"AUTOPILOT - AUTOPILOT"	1	 	
8	"AIRSPEED - AIRSPEED"	1	VNE Exceeded	

AURAL WARNING MESSAGES				
Priority	**Aural Message**	**No. of cycles**	**Cause**	
9	"LANDING GEAR"	1		
10	"ONE-FIFTY FEET"	1	Descending below 150 ft AGL	

AURAL WARNING GENERATOR TONES AND MESSAGES LIST

AWG TONES AND MESSAGES				
PRIORITY	**TONE**		**AURAL MESSAGE**	**NO. OF CYCLES**
1	2700 Hz	3	ROTOR LOW – ROTOR LOW	CONT.
2	700 thru 1700 Hz	2	ENGINE (X) OUT – ENGINE (X) OUT	1
3	2700 thru 900 Hz	2	ENGINE (X) FIRE – ENGINE (X) FIRE	CONT.
4	900 Hz	2	ROTOR HIGH – ROTOR HIGH	1
5	NONE	NONE	WARNING – WARNING (see NOTE 1)	1
6	NONE	NONE	AUTOPILOT – AUTOPILOT	1
7	600 Hz	1	AIRSPEED – AIRSPEED	1

AWG TONES AND MESSAGES				
PRIORITY	**TONE**		**AURAL MESSAGE**	**NO. OF CYCLES**
8	554 thru 440 Hz	1	LANDING GEAR	1
9	494 Hz	1	150 FEET	1
10	494 Hz	1	ALTITUDE – ALTITUDE	1
11	NONE	NONE	AURAL SYSTEM TEST	2

*: even if the number of cycles is single, the MWLs continue to flash until the acknowledgment is done.

NOTE D

This message is generated when any of the following warning condition is detected: 1(2) EEC FAIL, 1(2) ENG OIL PRESS, MGB OIL PRESS, MGB OIL TEMP, MAIN (AUX) BATT HOT, 1-2 DC GEN, BAG FIRE, DE-ICE.

CONTROLS AND INDICATORS

The CAS message window houses the warning, caution advisory, Maintenance Status messages and the word “END”. The “END” message is not included in the count of messages scrolled off the bottom of the display queue.

1 On the Cursor Control Device

- The smaller knurled knob: ..... enables pilot/copilot to perform the scroll up (CW) or down (CCW) of the CAS messages.

2 On the pilots collective stick

- The CAS RST pushbutton: ..... when pressed, the pilot or copilot acknowledges the warning/caution. the MWL/MCL light stop flashing and the aural message removed.

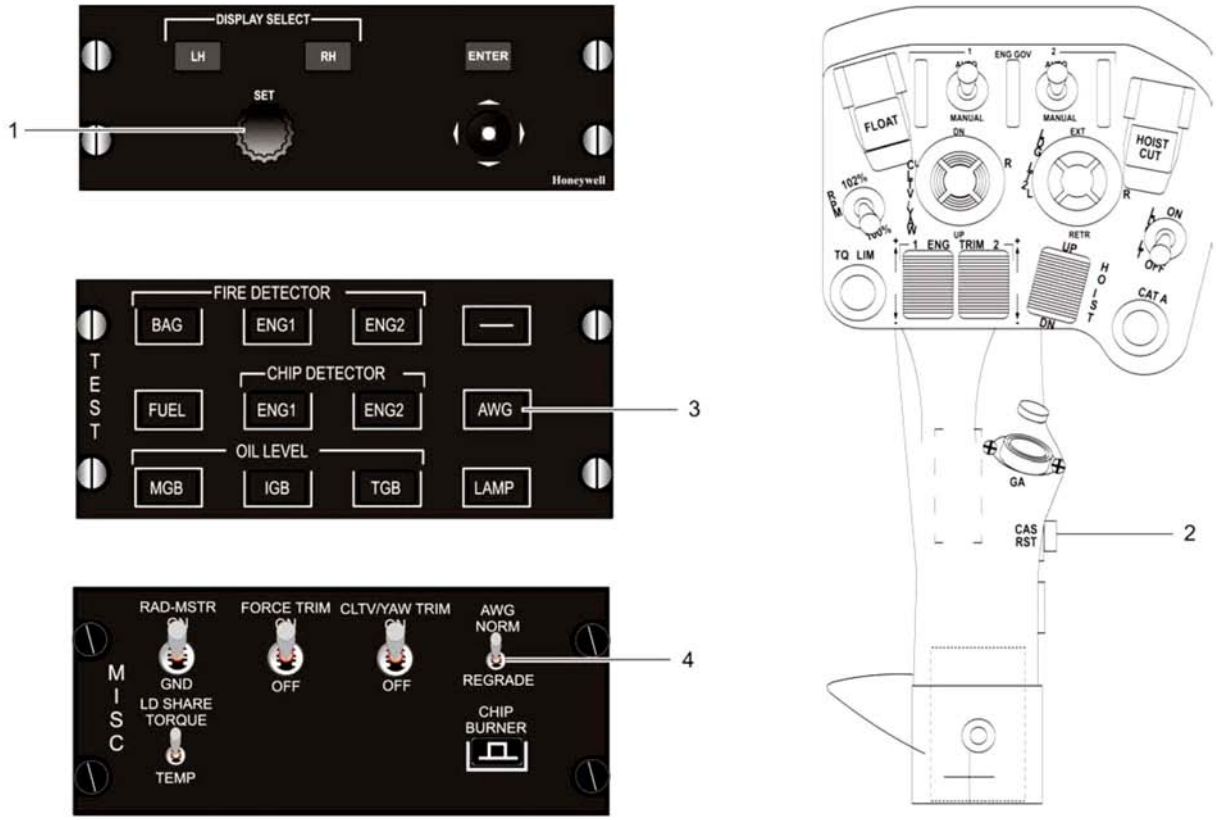
3 On the TEST CONTROL PANEL

- AWG pushbutton: ..... when momentarily pressed (both on ground and in flight) the “AURAL SYSTEM TEST” message is heard twice.
  When pressed for 6 sec (on ground only) all the tones and aural messages are heard in

the specific priority order. This test lasts 1 min about.

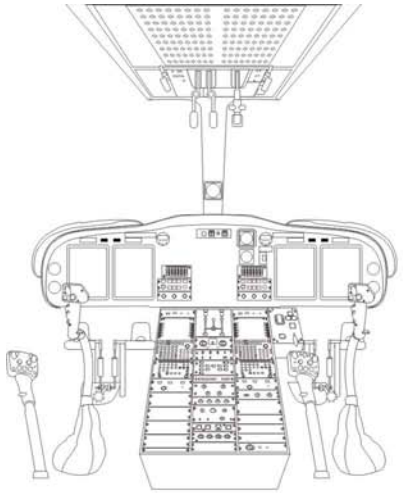
4 On the MISCELLANEOUS PANEL – AWG Switch

- NORMAL: ..... the tone and the “150 FEET” aural message is normally heard when the aircraft is descending below 150 feet.
- REGRADE: ..... the tone and the “150 FEET” aural message is suppressed.



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CENTRAL WARNING SYSTEM – CONTROLS





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MASTER ALARM PANEL

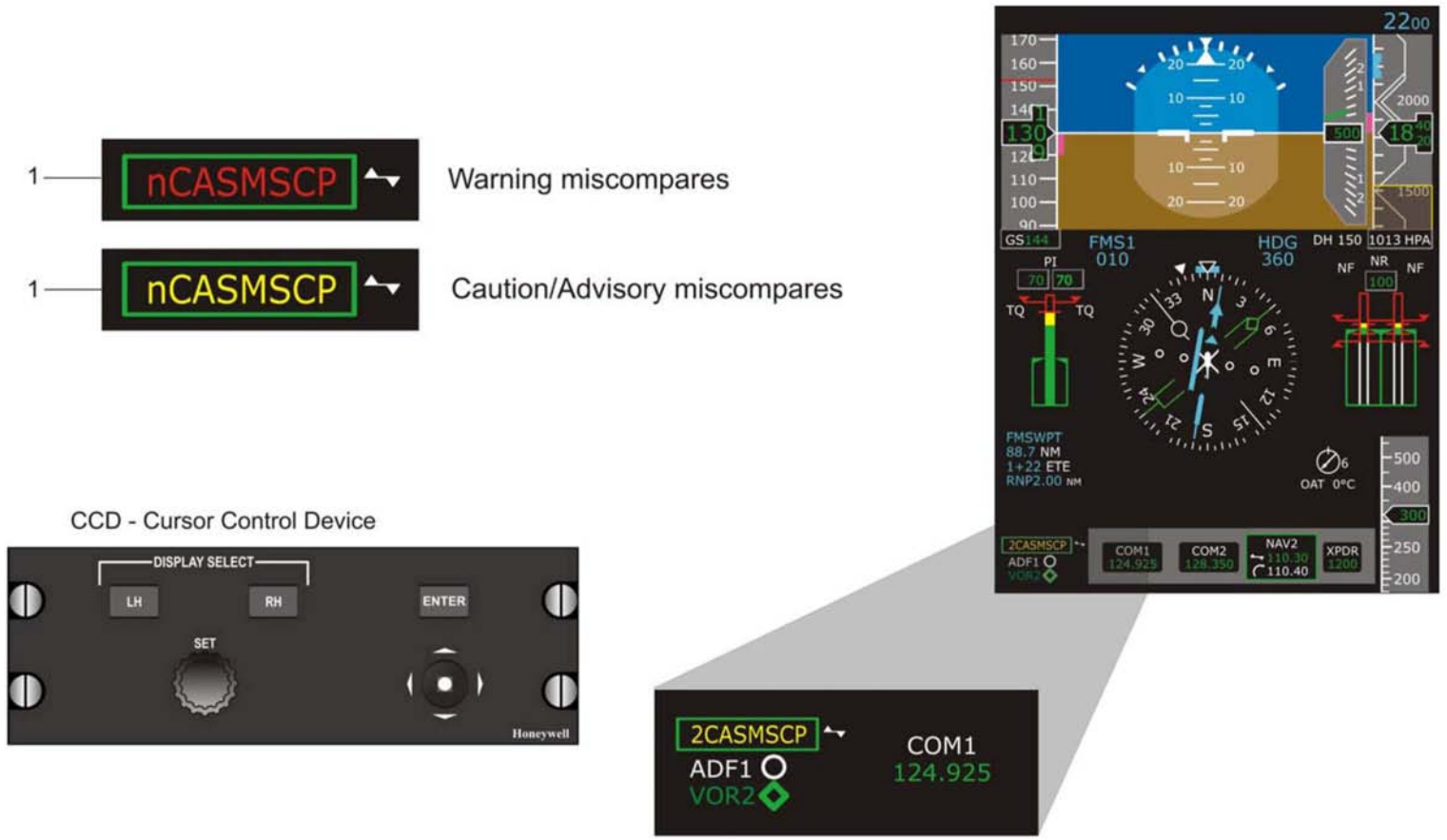
CAPTIONS

The CAS miscompare monitor verifies that both monitor warning functions are producing the same active text messages. The function calculates a checksum for the entire list of active CAS messages.
Additionally, a second checksum is calculated for the warning messages. A miscompare annunciation is displayed on the PFD if the warning message checksum fails to compare for 3 seconds or if the active list checksum miscompares for 7 seconds.
The miscompare is annunciated on the PFD.

1	"1(2)CASMSCP"	is the annunciations of the miscompare and depending on the MW function currently being displayed. When the miscompare is first detected, the CCD cursors is synchronized to the CASMSCP annunciation and a switch symbol () is displayed adjacent to the annunciation. While the CASMSCP annunciation is highlighted by the cursor, the pilot can toggle the CAS message list (pressing the “ENTER” pushbutton on the CCD) between the two monitor warning or caution message lists. This allows the pilot to ensure that he has read all messages to resolve the discrepancy. If the CAS miscompare involves a warning message the "1(2) CASMSCP" is displayed in red, otherwise it is displayed in amber. The annunciation is inside a green box.

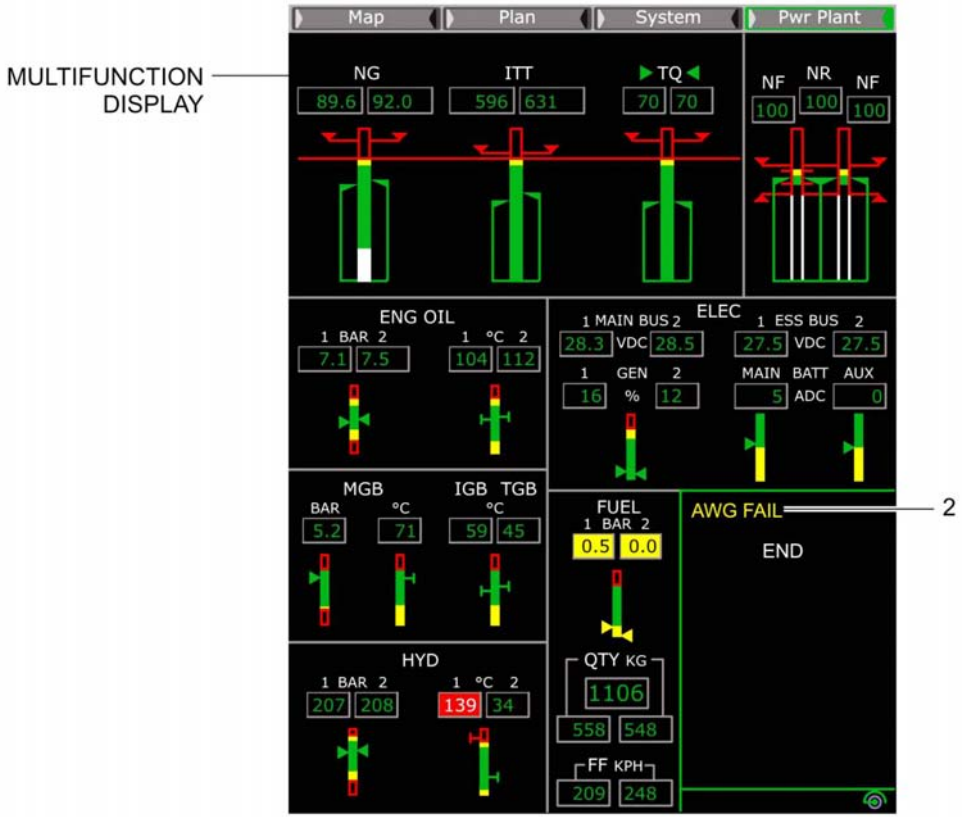
The caution provided in the CAS message window is as follows:

2	AWG FAIL:	when the Aural Warning Generator fails



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CENTRAL WARNING SYSTEM – INDICATING (SHEET 1 OF 2)



ICN-39-A-315000-G-A0126-01458-A-01-1

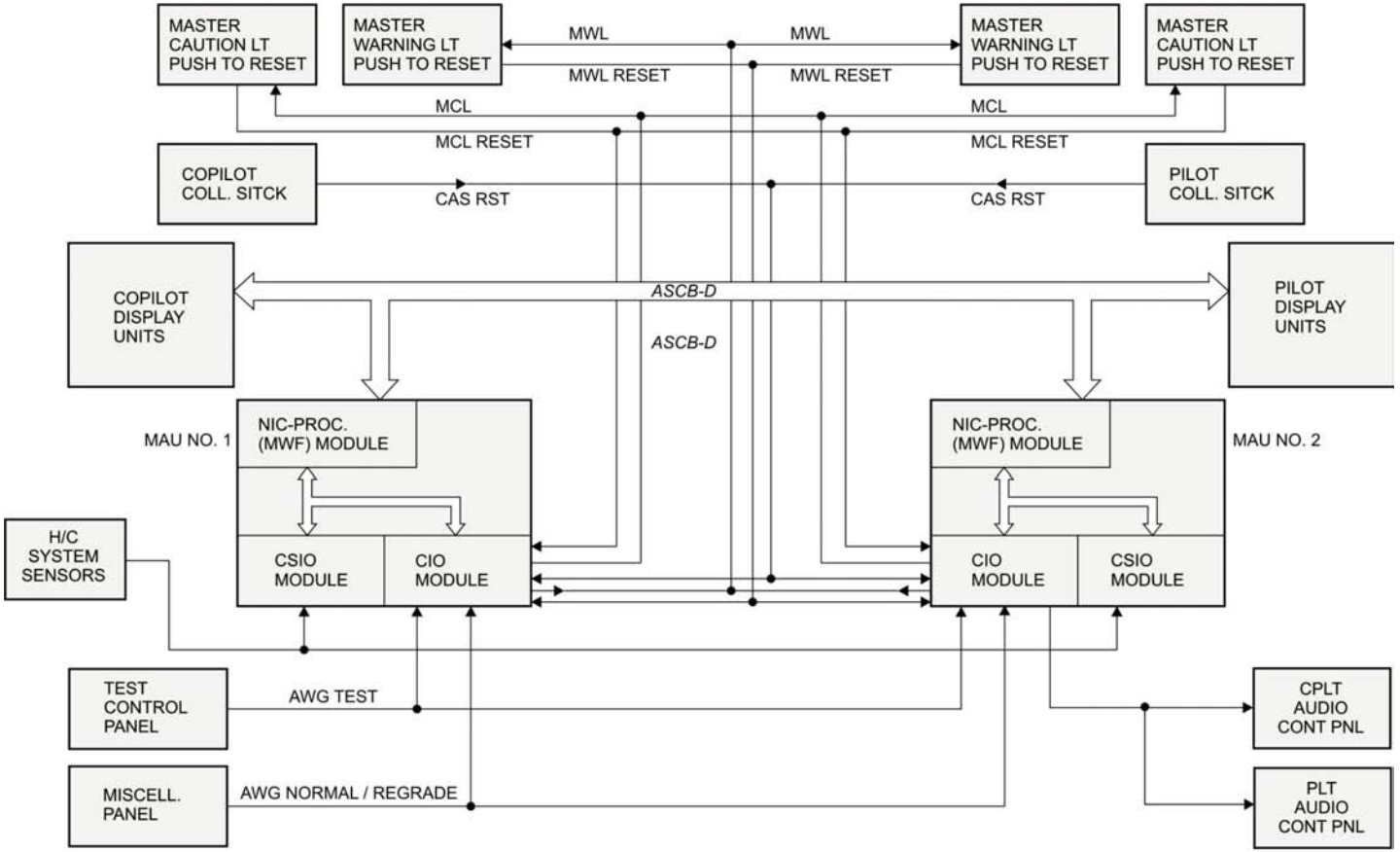
CENTRAL WARNING SYSTEM – INDICATING (SHEET 2 OF 2)

PRIORITY ACTIVATION

If a higher priority message is activated while a lower priority message is being generated, the two message are generated using the following interleaving method:



In any case, the aural message completes at least one intelligible cycle prior to termination.



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CENTRAL WARNING SYSTEM SCHEMATIC DIAGRAM

SECTION 60 - CENTRAL DISPLAY SYSTEMS

GENERAL DESCRIPTION

The Central Display System (CDS) is part of the Primus Epic® Integrated Avionics system and shows all the aircraft data to the pilots via four Display Units (DUs).
The most important data provided by the CDS are:

- attitude and heading (HDG) from the Attitude and Heading Reference System (AHRS)
- airspeed, vertical speed, and altitude data from the Air Data System (ADS)
- navigation data (such as deviations, bearing, distance, etc.) from the navigational radios and from the Flight Management System (FMS)
- guidance data from the Automatic Flight Control System (AFCS)
- active COM and NAV radio frequencies and ATC Transponder code
- engine and aircraft system parameters (such as NG, NF, NR, torque, temperatures, pressures, etc.)
- engine and aircraft system failure and status indications
- digital map data from the FMS and from Digital Map Generator (if installed)
- weather and ground-map data from the weather radar (WXR) system (if installed)
- terrain data from the Enhanced Ground Proximity Warning System (EGPWS) (if installed)
- traffic data from the Traffic Alert and Collision Avoidance System (TCAS) (if installed)
- video images from FLIR and/or cameras (if installed)

The DUs are controlled by the following equipment:

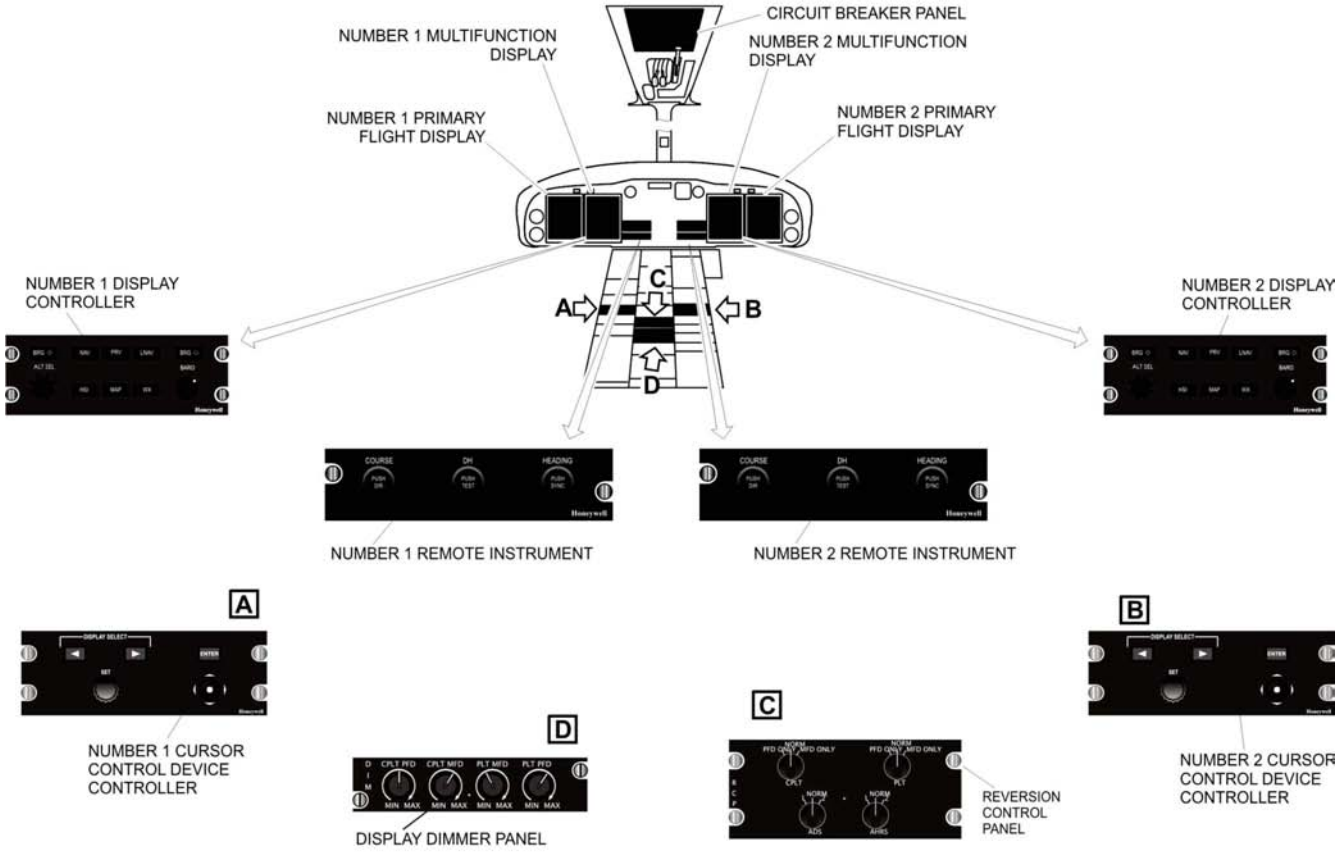
- The DU dimmer panel
- The Cursor Control Devices (CCDs)
- The Remote Instrument Controllers (RICs)
- The Display Controllers (DCs)
- The Multifunction Control Display Units (MCDUs)
- The Reversion Control Panel (RCP).

PRIMARY COMPONENTS

The central display system shows the primary flight data and navigation data to the flight crew. The primary components of the central display system are:

- The Number 1 primary flight display (A31)
- The Number 1 primary flight display mounting-tray
- The Number 2 primary flight display (A30)
- The Number 2 primary flight display mounting-tray
- The Number 1 cursor control device (PL7)
- The Number 2 cursor control device (PL23)
- The Number 1 remote instrument controller (A35)
- The Number 2 remote instrument controller (A34)
- The Number 1 display controller (A33)
- The Number 2 display controller (A32)
- The Number 1 multifunction display (A95)
- The Number 1 multifunction display mounting-tray

- The Number 2 multifunction display (A42)
- The Number 2 multifunction display mounting-tray
- The display dimmer panel (PL22)
- The display-reversion control panel (PL21).



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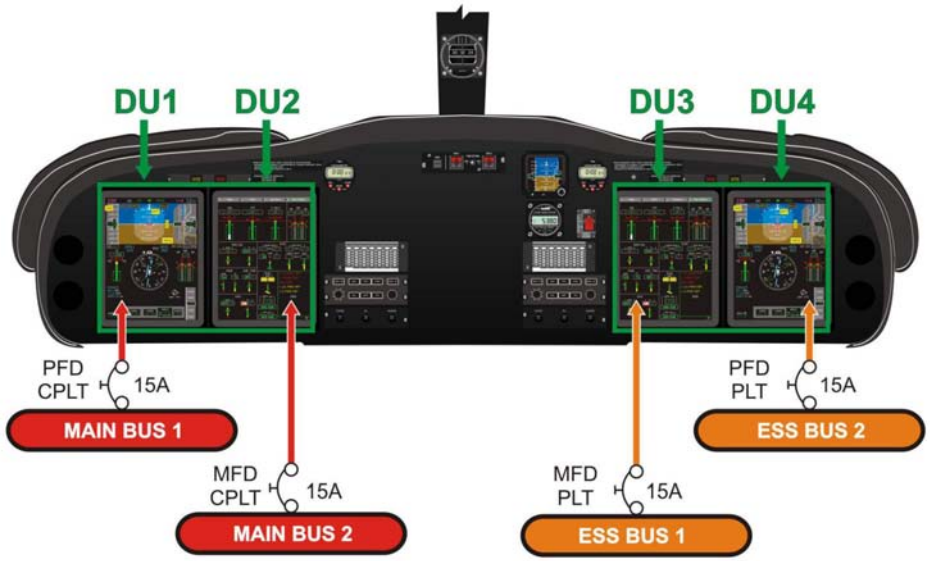
DISPLAY UNIT (DU)

The DU is a Line Replaceable Unit (LRU) provided with an 8”×5” color flat-panel Active Matrix Liquid-Crystal Display (AMLCD).
Each DU is connected directly to the ASCB-D and the Local Area Network (LAN). Each DU receives digital data from the Avionics-Standard Communication-Bus version-D (ASCB-D), generates the graphic information and displays them.
Four DU’s are installed on the instrument panel, two in front of the pilot and two in front of the co-pilot and are numbered DU 1 to DU 4 from left to right.
The outboard DU’s (DU 1 and DU 4) automatically operate as Primary Flight Displays (PFD).
The inboard DU’s (DU 2 and DU 3) automatically operate as Multi-Function Displays (MFD).
The DUs are electrically supplied by different buses:

- DU 1 (PFD1) from MAIN 1 bus via the DISPLAY – PFD CPLT circuit breaker
- DU 2 (MFD1) from MAIN 2 bus via the DISPLAY – MFD CPLT circuit breaker
- DU 3 (MFD2) from ESS 1 bus via the DISPLAY – MFD PLT circuit breaker
- DU 4 (PFD2) from ESS 2 bus via the DISPLAY – PFD PLT circuit breaker

NOTE A

Copilot’s DUs power inputs are connected to ESS 1 and ESS 2 buses when the aircraft is on the ground or when starting an engine.
Each unit has an internal cooling fan that supplies the necessary cooling.



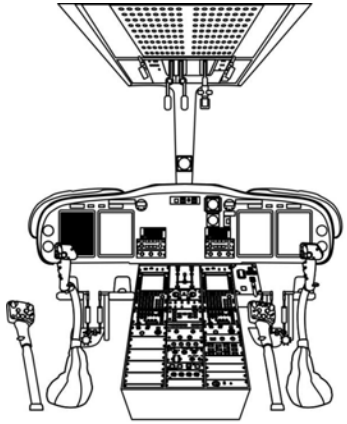


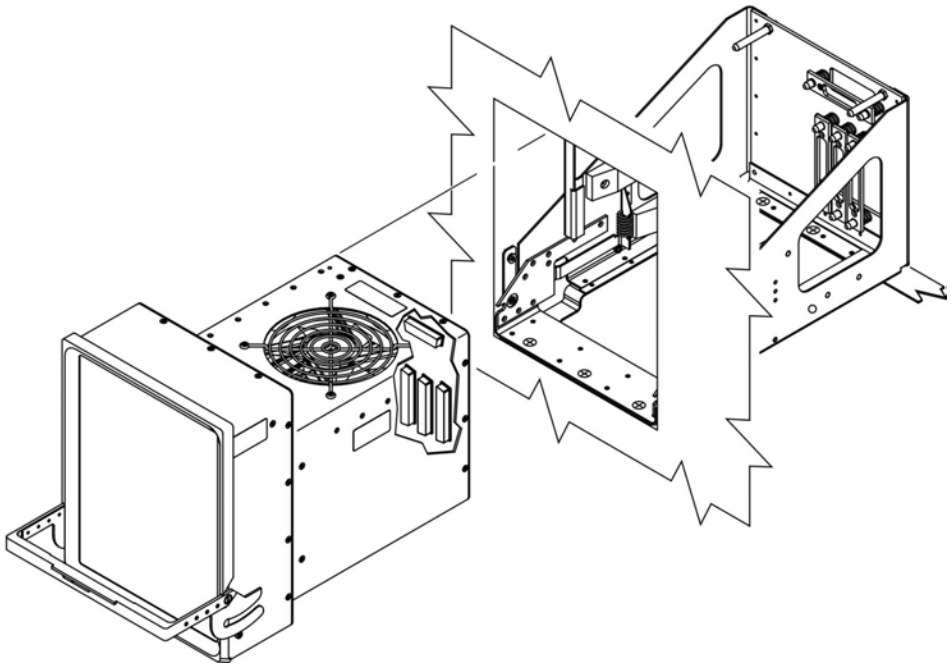
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NUMBER 1 PRIMARY FLIGHT DISPLAY

The Number 1 Primary Flight Display (PFD1) is a Line Replaceable Unit (LRU). It is installed in the external left side of the instrument panel. The display unit is the front of the PFD1. It is a flat panel of the LCD type. The PFD1 is installed on a mounting tray and is attached to it with a latch. The PFD1 has a handle that operates the latch. A latch release button is installed in the handle for easy removal of the PFD1 from its mounting tray. Three electrical connectors (A31P1, A31P3 and A31P4) in the rear of the PFD1 connect the PFD1 to the components that follow through the connector interface on the mounting tray:

- The Number 1 modular avionic unit
- The Number 1 cursor control device
- The Number 1 remote instrument controller
- The Number 1 display controller
- The Number 1 multifunction control display unit.



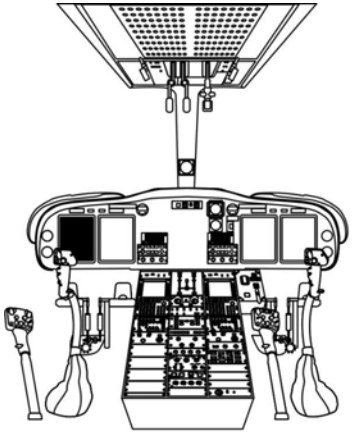


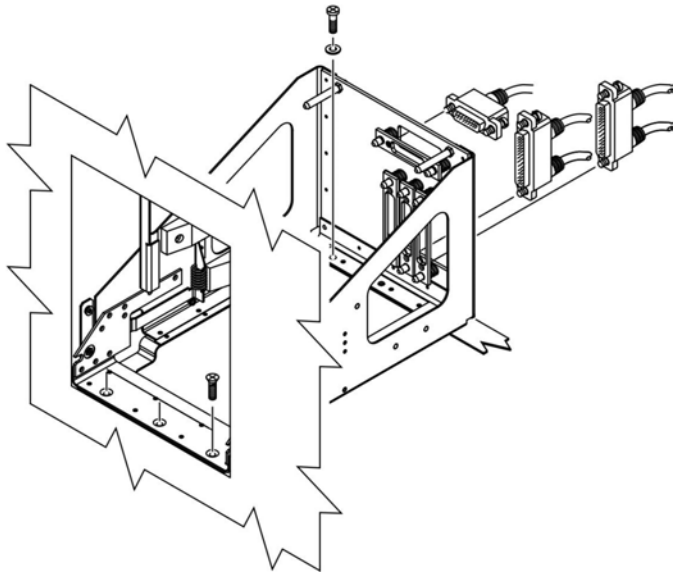
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N.1 PRIMARY FLIGHT DISPLAY

NUMBER 1 PRIMARY FLIGHT DISPLAY MOUNTING TRAY

The PFD1 mounting-tray is installed behind the instrument panel. Three screws attach it to the instrument panel, and three screws and washers attach its rear part to the structure.
The mounting tray holds the PFD1 attached to the instrument panel. On the connector interface, three electrical connectors let the connection of the three PFD1 electrical connectors A31P1, A31P3 and A31P4.





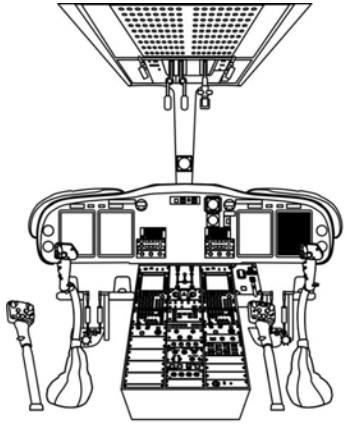
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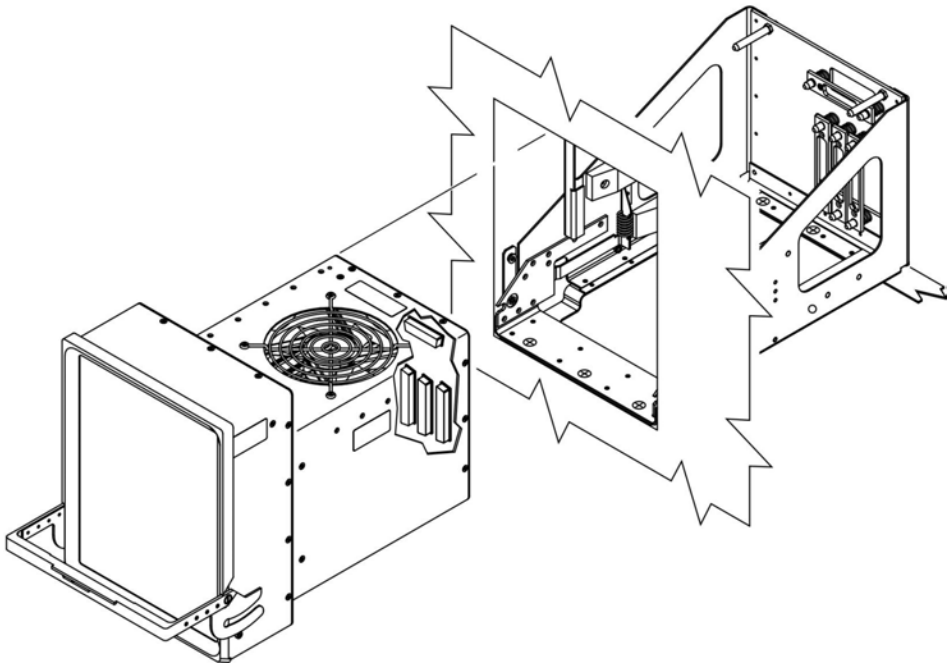
N.1 PRIMARY FLIGHT DISPLAY MOUNTING TRAY

NUMBER 2 PRIMARY FLIGHT DISPLAY

The Number 2 Primary Flight Display (PFD2) is a Line Replaceable Unit (LRU). It is installed in the external right side of the instrument panel. The display unit is the front of the PFD2. It is a flat panel of the LCD type. The PFD2 is installed on a mounting tray and is attached to it with a latch. The PFD2 has a handle that operates the latch. A latch release button is installed in the handle for easy removal of the PFD2 from its mounting tray. Three electrical connectors A30P1, A30P3 and A30P4 in the rear of the PFD2 connect the PFD2 to the components that follow through the connector interface on the mounting tray:

- The Number 2 modular avionic unit
- The Number 2 cursor control device
- The Number 2 remote instrument controller
- The Number 2 display controller
- The Number 2 multifunction control display unit.



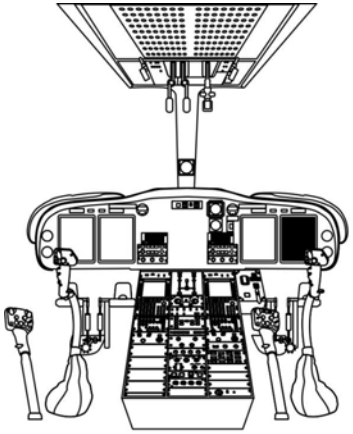


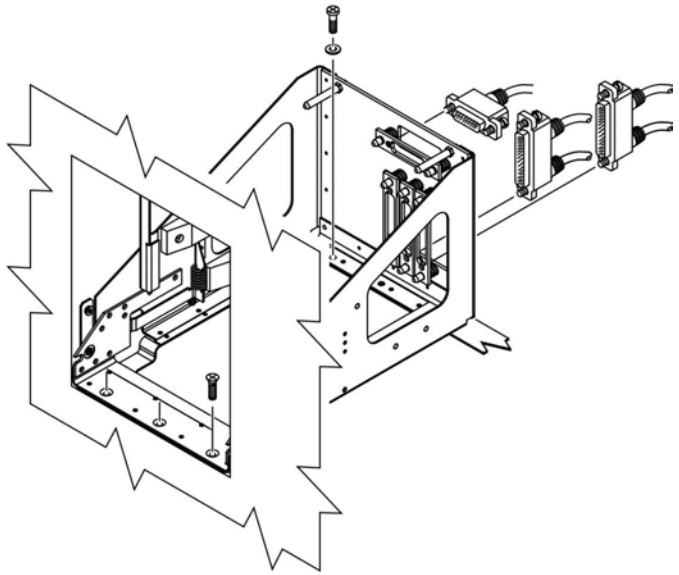
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N.2 PRIMARY FLIGHT DISPLAY

NUMBER 2 PRIMARY FLIGHT DISPLAY MOUNTING TRAY

The PFD2 mounting tray is installed behind the instrument panel. Three screws attach it to the instrument panel, and three washers and three screws attach its rear part of the structure. The PFD2 mounting tray holds the PFD2 attached to the instrument panel.
On the connector interface is installed in the rear of the PFD2 tray. On the connector interface, three electrical connectors let the connection of the three PFD2 electrical connectors A30P1, A30P3 and A30P4.





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N.2 PRIMARY FLIGHT DISPLAY MOUNTING TRAY

PRIMARY FLIGHT DISPLAY (PFD)

The PFD is connected directly to the ASCB-D and the Local Area Network (LAN). The PFD receives digital data from the Avionics-Standard Communication-Bus version-D (ASCB-D), generates the graphic information and displays them.
The PFD has a single format and its main function is to provide the following:

- ADI (Attitude Director Indicator) display
- HSI (Horizontal Situation Indicator) display
- Airspeed Indicator
- Vertical Speed Indicator
- Barometric Altimeter
- Power Index Indicator (PI)
- Triple Tachometer (NR, NF)
- Navigational data (Source, Ident, Course, Distance, etc)
- Radio Altimeter (RADALT)
- OAT and Wind Vector Indicators
- COMM / NAV / XPDR Frequencies and Code



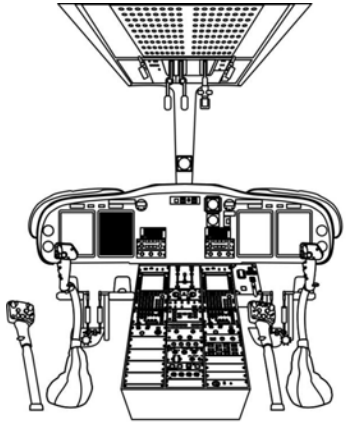
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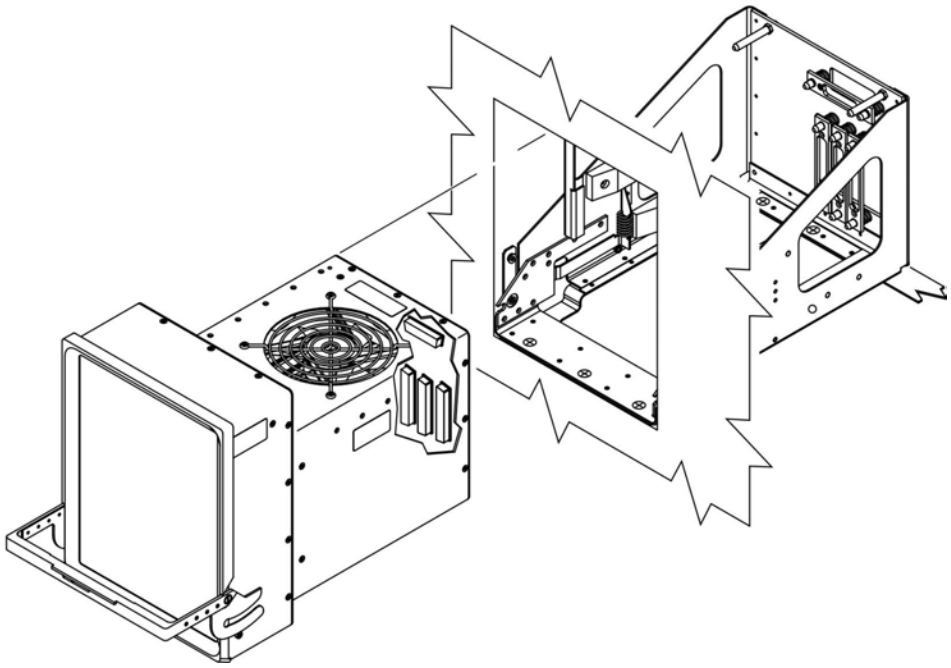
PRIMARY FLIGHT DISPLAY

NUMBER 1 MULTIFUNCTION DISPLAY

The Number 1 Multi-Function Display (MFD1) is installed in the left side of the instrument panel adjacent to the PFD1. The display unit is the front of the MFD1. It is a flat panel of the LCD type. The MFD1 is installed on a mounting tray and is attached to it with a latch. The MFD1 has a handle that operates the latch. A latch release button is installed in the handle for easy removal of the MFD1 from its mounting tray. Three electrical connectors A95P1, A95P3 and A95P4 in the rear of the MFD1 connect the MFD1 to the components that follow through the connector interface on the mounting tray:

- The Number 2 modular radio cabinet (A7)
- The Number 1 modular radio cabinet (A8)
- The MFD2 (A42)
- The PFD1 (A31).



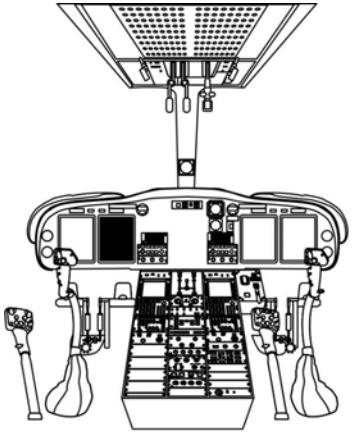


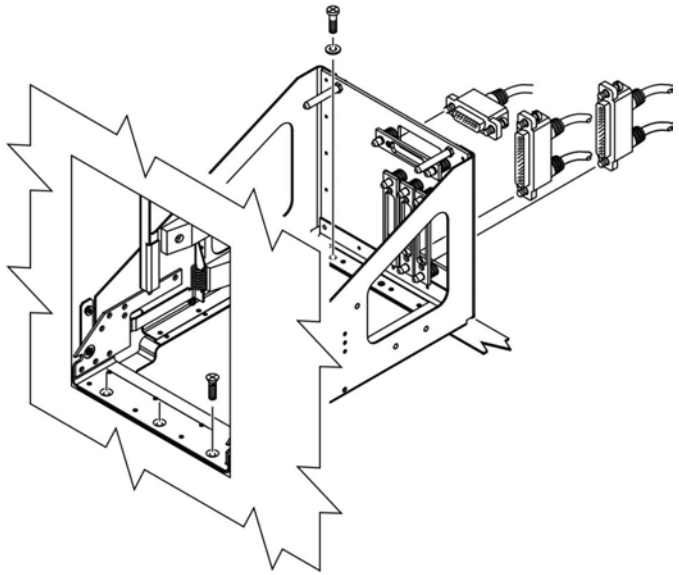
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N.1 MULTIFUNCTION DISPLAY

NUMBER 1 MULTIFUNCTION DISPLAY MOUNTING TRAY

The MFD1 mounting tray is installed behind the instrument panel. Three screws attach it to the instrument panel, and three screws and three washers attach its rear part to the structure. The mounting tray holds the MFD1 attached to the instrument panel.
A connector interface is installed in the rear of the MFD1 tray. On the connector interface, three electrical connectors let the connection of the three MDF1 electrical connectors A95P1, A95P3 and A95P4.





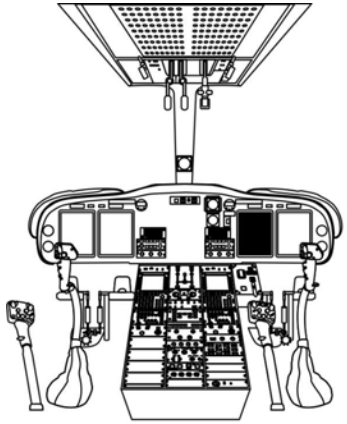
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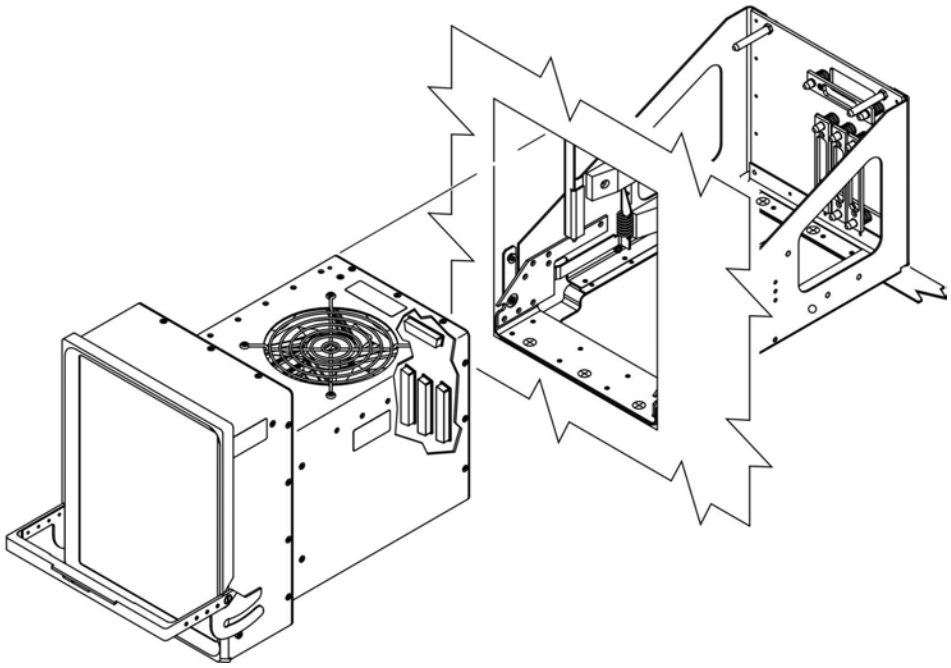
N.1 MULTIFUNCTION DISPLAY MOUNTING TRAY

NUMBER 2 MULTIFUNCTION DISPLAY

The Number 2 Multi-Function Display (MFD2) is installed in the right side of the instrument panel adjacent to the PFD2. The display unit is the front of the MFD2. It is a flat panel of the LCD type. The MFD2 is installed on a mounting tray and is attached to it with a latch. The MFD2 has a handle that operates the latch. A latch release button is installed in the handle for easy removal of the MFD1 from its mounting tray. Three electrical connectors A42P1, A42P3 and A42P4 in the rear of the MFD2 connect the MFD2 to the components that follow through the connector interface on the mounting tray:

- The Number 2 modular radio cabinet (A7)
- The Number 1 Modular radio cabinet (A8)
- The MFD1 (A92)
- The PFD2 (A30).



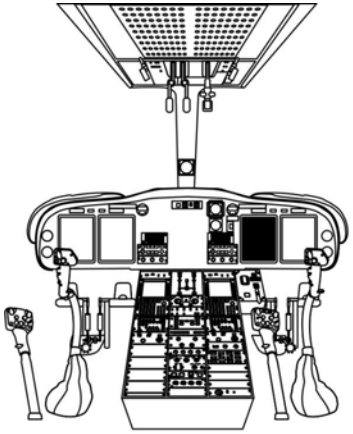


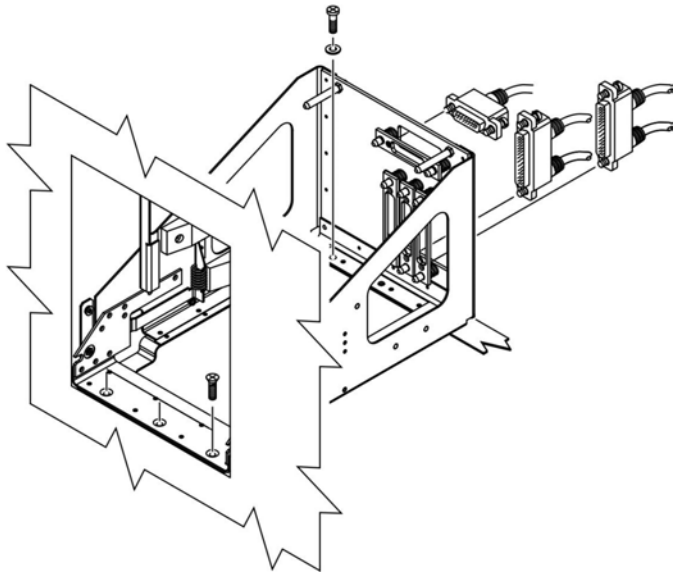
ICN-39-A-316000-G-A0126-02961-A-01-1

N.2 MULTIFUNCTION DISPLAY

NUMBER 2 MULTIFUNCTION DISPLAY MOUNTING TRAY

The MFD2 mounting-tray is installed behind the instrument panel. Three screws attach it to the instrument panel, and three screws and three washers attach its rear part to the structure.
The mounting tray holds the MFD2 attached to the instrument panel.
A connector interface is installed in the rear of the MFD2 tray. On the connector interface, three electrical connectors let the connection of the three MDF2 electrical connectors A42P1, A42P3 and A42P4.





ICN-39-A-316000-G-A0126-02962-A-01-1

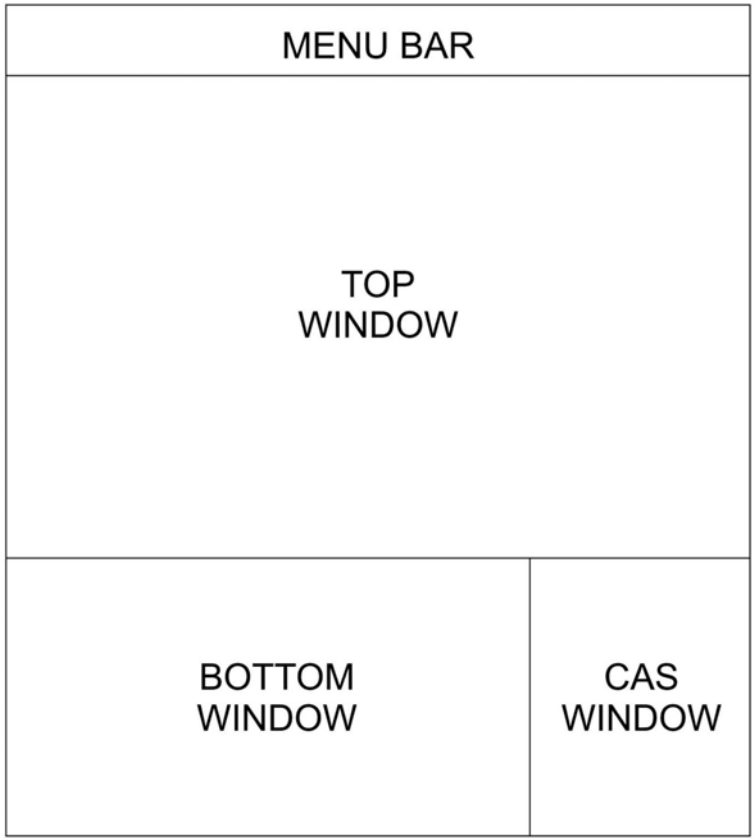
N.2 MULTIFUNCTION DISPLAY MOUNTING TRAY

MULTIFUNCTION DISPLAY

The Multifunction Display (MFD) is connected directly to the ASCB-D and the Local Area Network (LAN). Refer to 39-A-00-00-00-00A-044B-A. The MFD receives digital data from the Avionics-Standard Communication-Bus version-D (ASCB-D), generates the graphic information and displays them.
The MFD shows navigation map (NAV) displays, other system (synoptic) pages, engine instrument and Crew Alerting System (CAS) pages. The MFD show the primary flight display (PFD) in a composite format for automatic and manual reversion modes. The MFD uses the data from the primary aircraft systems to show NAV, WX, air traffic, and other displays.
The windows of the MFD are:

- The menu bar
- The top window
- The bottom window
- The CAS window.

The Power Plant format is the default for the MFD. When the Power Plant format is set the top and lower windows become one window.
The CAS window shows indications from the CAS. This window always shows when the aircraft is in flight. Only in ground If you set the Maintenance or Configuration formats, all three MFD windows become one full-screen window.



ICN-39-A-316000-G-A0126-01488-A-01-1

MULTIFUNCTION DISPLAY PRESENTATION

MENU BAR

The menu bar is at the top of the MFD. It is used to set the display format of the MFD. The menu bar has four virtual pushbuttons. A green box cursor shows that the set menu is the one related to it. Each menu has a fold-down sub-menu that lets system modes be set and format details be shown. To do the selections in the menu bar, the pilots must move the green box cursor with the joystick of the cursor control device (CCD) to the applicable virtual pushbutton, then push the enter button on the CCD.
The menu bar has the virtual pushbuttons that follow:

- Pwr Plant
- System
- Map
- Plan.

If a menu virtual pushbutton is selected, the related sub-menu comes in view when the CCD joystick is moved in the down direction.
The CCD joystick must be moved up or down to do the selection of the items on the menu list.

TOP WINDOW

The top window shows the formats that follow:

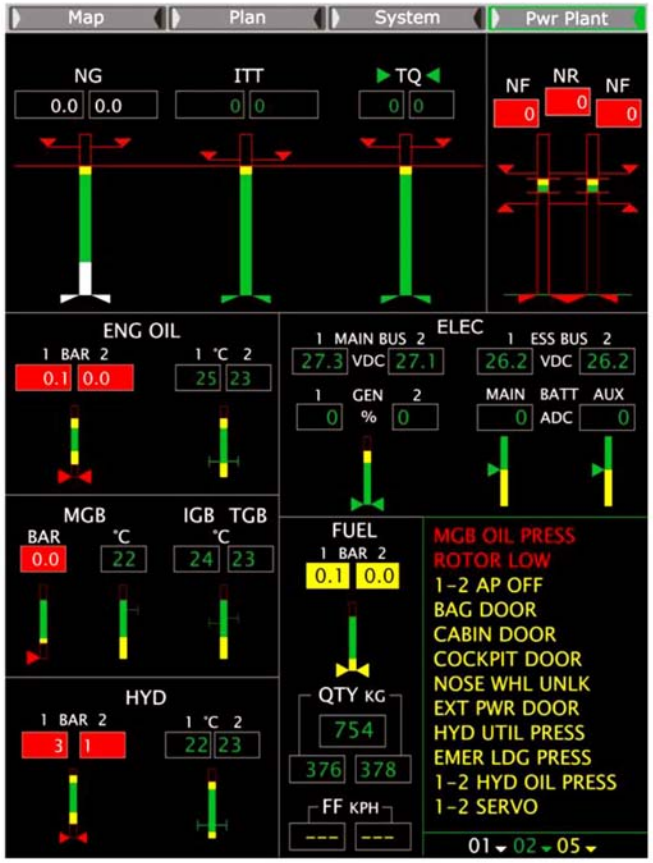
- Pwr Plant (default at power up)
- System
- Map
- Plan.

The different formats are selectable via Curse Control Device (CCD).

POWER PLANT FORMAT

The Pwr Plant format is the standard format for MFD. The MFD shows the primary, secondary engine and system data. When the Main format is set the top and bottom windows become one window.
The main function of the MFD Power Plant format is to provide the following:

- Engine primary data (NG, ITT, TQ)
- Triple Tachometer (NR, NF)
- Engine secondary indications (Oil Press and Temp)
- Gearboxes indications (Oil Press and Temp)
- Electrical power system indications (Voltage, Load)
- Hydraulic power system indications (Oil Press and Temp)
- Fuel system indications (Fuel Press, Quantity and Flow)
- Crew Alerting System (CAS) messages



ICN-39-A-316000-G-A0126-02963-A-01-1

MULTIFUNCTION DISPLAY (POWER PLANT FORMAT)

SYSTEM FORMAT

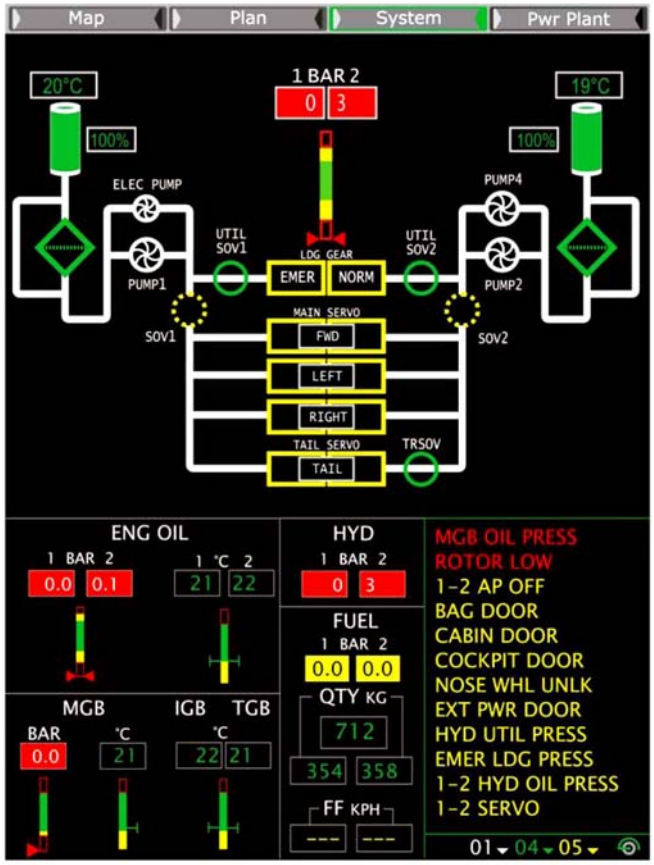
Different sub-formats are available to display different system data; selection is performed via the CCD.
Standard System sub-formats are:

- “Electrical” (Electrical power system synoptic diagram)
- “Hydraulic” (Hydraulic power system synoptic diagram)
- “Flt Contr” (AFCS indications)
- “Maintenance” (Central Maintenance Computer data)
- “Sys Config” (Configuration management data)
- Time/Date
- Config

NOTE B

“Maintenance” and “Sys Config” formats are designed for maintenance purposes and can only be selected on the ground.

In case the helicopter is fitted with optional systems that output a video signal, additional sub-formats will be available (Eg: Cameras, FLIR, Digital Map, etc.). All the sub-formats that are selectable during flight show the relevant system data in the top window of the MFD; the bottom window shows reduced Power Plant format data.

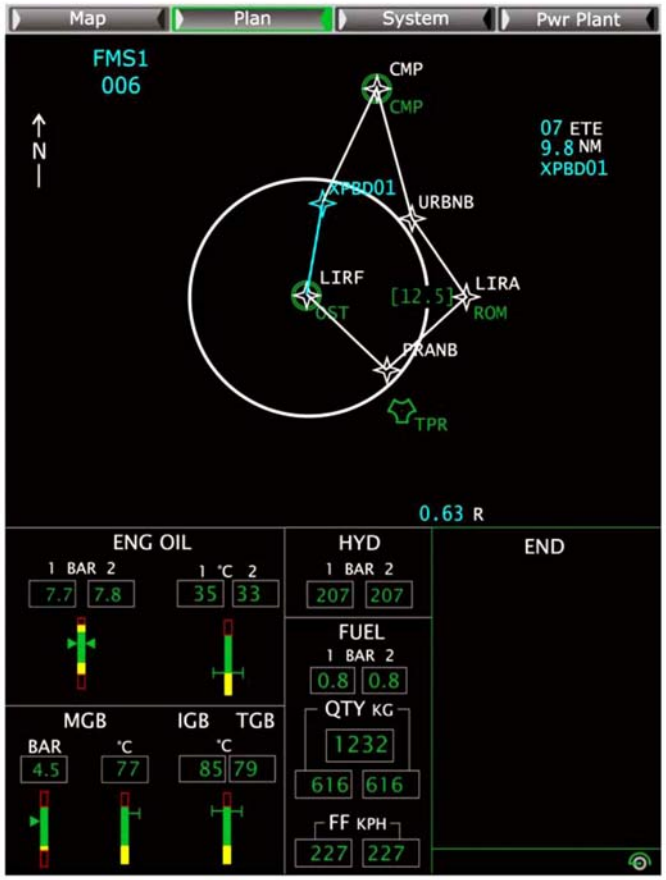


ICN-39-A-316000-G-A0126-02964-A-01-1

MULTIFUNCTION DISPLAY (SYSTEM FORMAT)

PLAN FORMAT

The main function of the MFD Plan format is to display a North-Up synthetic map based on FMS data in the top window of the MFD. The bottom window of the MFD shows reduced Power Plant format data.



ICN-39-A-316000-G-A0126-02965-A-01-1

MULTIFUNCTION DISPLAY (PLAN FORMAT)

MAP FORMAT

The main function of the MFD Plan format is to display a Heading-Up synthetic map based on FMS data in the top window of the MFD. The bottom window of the MFD shows reduced Power Plant format data.

BOTTOM WINDOW

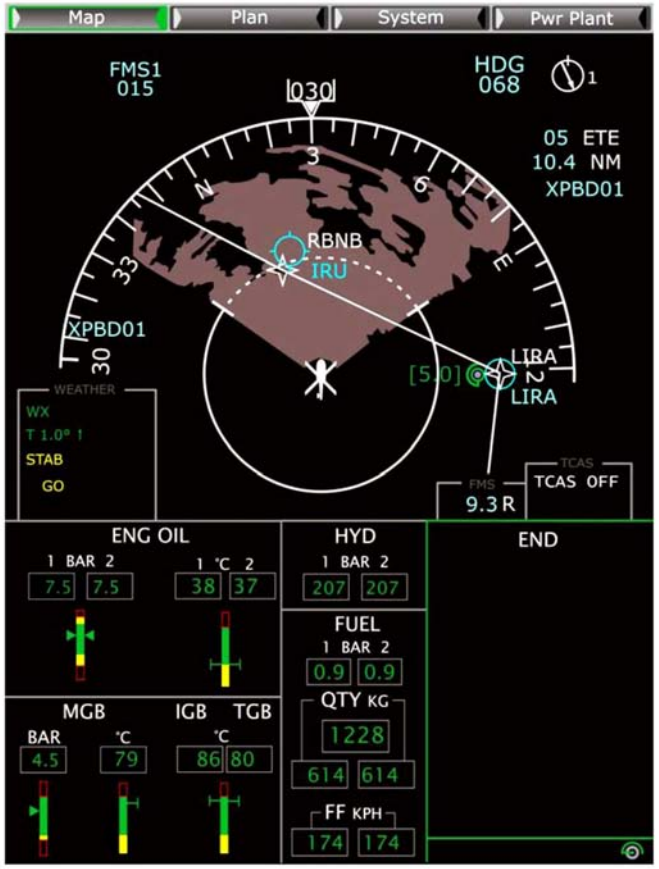
The bottom window in the bottom left corner of the MFD. The bottom window shows the parameters that follow:

- Engines Oil Pressure
- Engines Oil Temperature
- MGB Oil Pressure
- MGB Oil Temperature
- IGB Oil Temperature
- TGB Oil Temperature.
- HYD systems Pressure
- Fuel systems Pressure
- Tanks and total Fuel quantity
- Fuel Flow

The format of the scales for these parameters is the same format used when on the Pwr plante page.

CAS WINDOW

The CAS window shows messages from the CAS in the bottom right corner of the MFD. This window always shows when the aircraft is in flight. When the MFD is in the composite display mode, the CAS message window is in view at the bottom center of the composite PFD format. If the Maintenance or Configuration formats is set, the three MFD windows become one full-screen window.



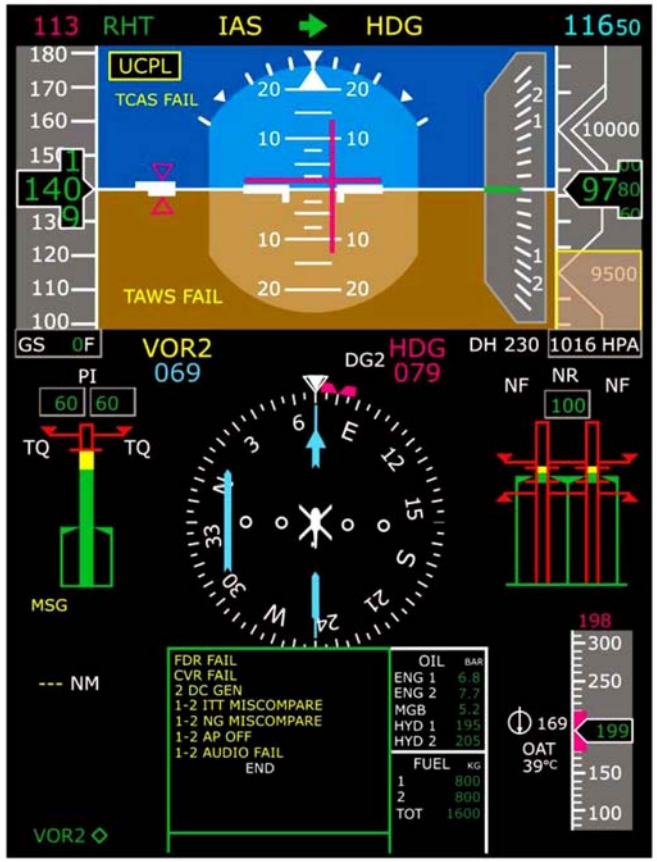
ICN-39-A-316000-G-A0126-02966-A-01-1

MULTIFUNCTION DISPLAY (MAP FORMAT)

COMPOSITE FORMAT

Any DU shows the Composite format whenever the paired DU is off, so as to provide pilot or co-pilot with minimum essential information for safe flight with a single DU.
The Composite format can either be automatically or manually selected.
Automatic reversion to Composite format occurs when the paired DU becomes inactive on the ASCB D and latches to prevent flashing due to intermittent failures.
Manual reversion can be selected via the Reversion Control Panel (RCP) PLT or CPLT selectors: this causes the non-selected paired DU to turn off.
When the PLT or CPLT selectors are returned to NORM, the automatic reversion operation is reset.
Composite format is a reduced PFD format with the addition of the following:

- Crew Alerting System (CAS) messages
- Pressure readouts (Engines, Main Gearbox, Hydraulic power systems)
- Fuel Quantity indicator

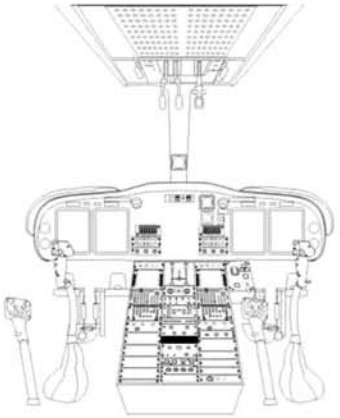


ICN-39-A-316000-G-A0126-02967-A-01-1

MULTIFUNCTION DISPLAY (COMPOSITE FORMAT)

DISPLAY DIMMER PANEL

The display dimmer panel is installed on the interseat console. Two quick-release fasteners attach it to the interseat console. The display dimmer panel has four dimming potentiometers. Two electrical connectors connect the display dimmer panel to the PFD1, PFD2, MFD1 and MFD2.
DU Dimming control panel allows adjusting the LCD brightness of the two PFD and two MFD individually by means of four potentiometers. The backlighting system uses a fluorescent lamp that has a long life and a stable colour range. Each display monitor the ambient light on the front of the unit and automatically adjusts the unit brightness and contrast to be compatible with the levels set by the pilot through the display dimmer panel.



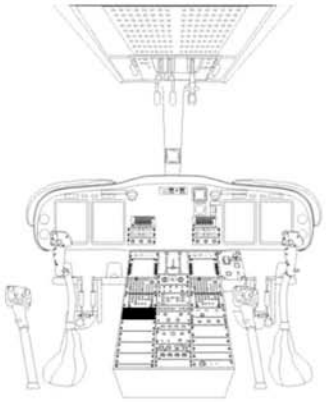


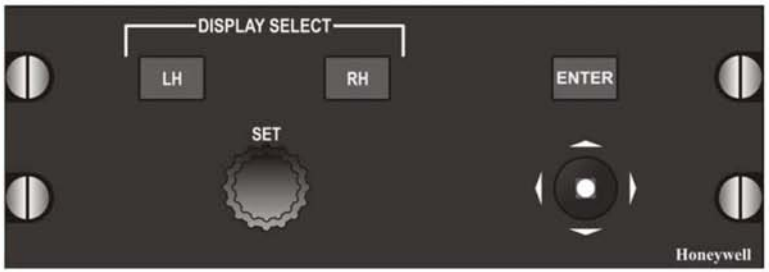
ICN-39-A-316000-G-A0126-02968-A-01-1

DISPLAY DIMMER PANEL

NUMBER 1 CURSOR CONTROL DEVICE

The Number 1 Cursor Control Device (CCD1) is installed in the left side of the interseat console. Four quick-release fasteners attach the CCD1 to the interseat console. The CCD1 has three buttons and two knobs. The electrical connector PL7P1 on the rear side of the device connects the CCD1 to the Number 1 display controller and Number 2 display controller.



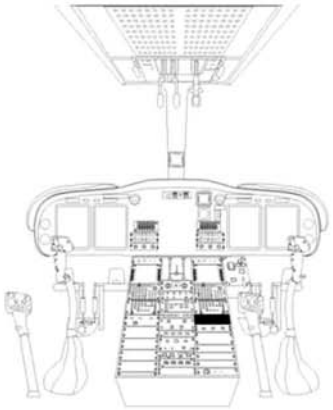


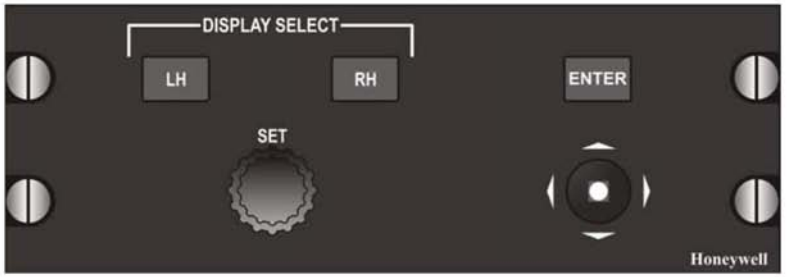
ICN-39-A-316000-G-A0126-02969-A-01-1

N.1 CURSOR CONTROL DEVICE

NUMBER 2 CURSOR CONTROL DEVICE

The Number 2 Cursor Control Device (CCD2) is installed in the right side of the interseat console. Four quick-release fasteners attach the CCD2 to the interseat console. The CCD2 has three buttons, and two knobs. The electrical connector PL23P1 connects the CCD2 to the Number 1 display controller and Number 2 display controller.
CCD is used to manage data and control the DU's. The control panel has two DISPLAY SELECT buttons, the SET control (dual concentric knobs), a joystick and an ENTER button.





ICN-39-A-316000-G-A0126-02970-A-01-1

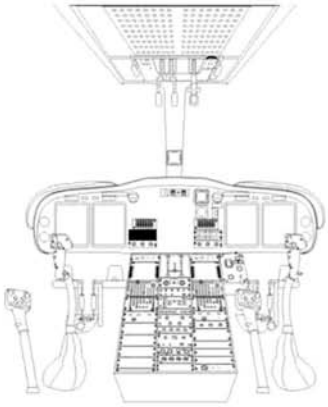
N.2 CURSOR CONTROL DEVICE

NUMBER 1 DISPLAY CONTROLLER

The Number 1 display controller is installed in the left side of the instrument panel. Four quick-release fasteners attach the Number 1 display controller to the instrument panel. Two electrical connectors A33P1 and A33P2 connect the Number 1 display controller to the components that follow:

- The display reversion control panel
- The Number 2 display controller
- The CCD1
- The CCD2
- The circuit breaker CB82 (28 V DC).

The Number 1 display controller uses 0 to 5 V AC or 0 to 5 V DC to supply power for the lights. The controller is night vision goggle (NVG) compatible.





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N.1 DISPLAY CONTROLLER

NUMBER 2 DISPLAY CONTROLLER

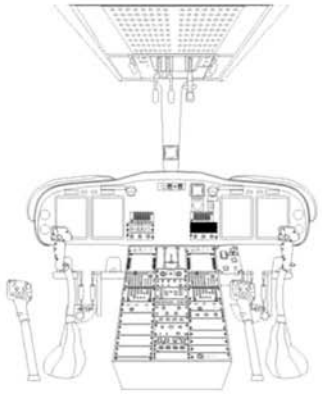
The Number 2 display controller is installed in the center-right side of the instrument panel. Four quick-release fasteners attach the Number 2 display controller to the instrument panel. Two electrical connectors A32P1 and A32P2 connect the Number 2 display controller to the components that follow:

- The display reversion control panel
- The Number 1 display controller
- The CCD2
- The CCD1
- The circuit breaker CB79 (28 V DC).

All buttons on the controller are back-lighted to see better at night. The knobs on the controller use halos to help find them in the dark. The controller uses 0 to 5 V AC or to 5 V DC to supply power for the lights.
The controller is night vision goggle (NVG) compatible. The back-lighted panel and buttons are internally filtered.
The two DC provide pilot/co-pilot independent selection of:

- Barometric pressure reference
- Selected Altitude
- Short-range and long-range navigation sources
- On-side/cross-side bearing sources
- HSI format for display on the associated PFD

The DC also provides for interface of the on-side Cursor Control Device (CCD) and Remote Instrument Controller (RIC) to the MAU's.



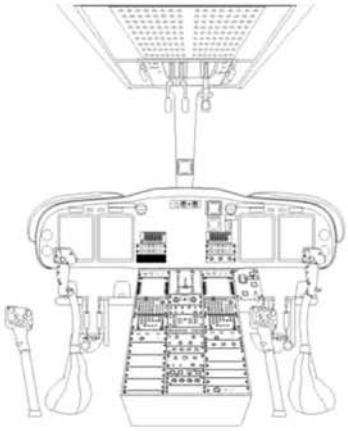


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N.2 DISPLAY CONTROLLER

NUMBER 1 REMOTE INSTRUMENT CONTROLLER

The Number 1 Remote Instrument Controller (RIC1) is installed in the left side of the instrument panel. Two quick-release fasteners attach the RIC1 to the instrument panel. The RIC1 has three knobs, each of which has a button at center. The electrical connector A35P1 connects the RIC1 to the Number 1 display controller.



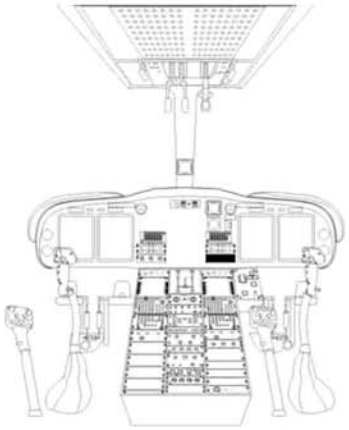


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N.1 REMOTE INSTRUMENT CONTROLLER

NUMBER 2 REMOTE INSTRUMENT CONTROLLER

The Number 2 Remote Instrument Controller (RIC2) is installed in the right side of the instrument panel. Two quick-release fasteners attach the RIC2 to the instrument panel. The RIC2 has three knobs, each of which has a button at center. The electrical connector A34P1 on the rear of the controllers connects the RIC2 to the Number 2 display controller.
The two RIC are used to set Course, Decision Height (DH) and Heading and to perform Radio Altimeter self-test by means of three control knobs with associated buttons.
All RIC selections except Heading Set are independent for pilot and co-pilot.
CCD, RIC and DC panels constitute a set of control panels for the DUs.
CCD and RIC are interfaced with the DC panel; the DC interfaces the set of panels to the MAUs.
The co-pilot set of panels is supplied by MAIN 1 bus and protected by DISPLAY – PFD CPLT CONTR circuit breaker.
The pilot set of panels is supplied by ESS 1 bus and protected by DISPLAY – PFD PLT CONTR circuit breaker.





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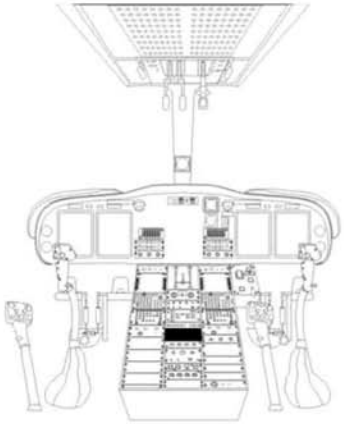
N.2 REMOTE INSTRUMENT CONTROLLER

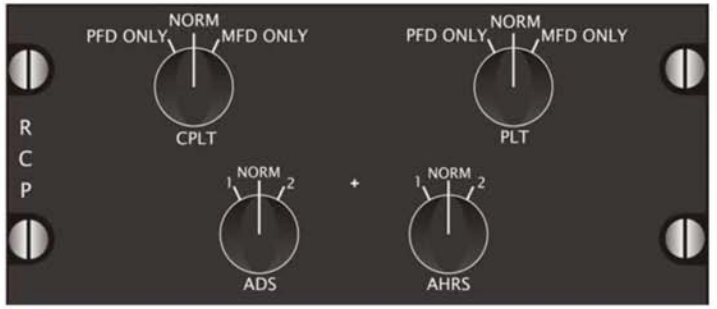
DISPLAY REVERSION CONTROL PANEL

The display-reversion control panel is installed on the interseat console. Four quick-release fasteners attach it to the interseat console. The display-reversion control panel has four switches.
Two electrical connectors PL21P1 and PL21P2 connect the display-reversion control panel to the components that follow:

- The Number 2 multifunction display (A42)
- The Number 1 primary-flight display (A31)
- The Number 2 primary-flight display (A30)
- The Number 2 display controller (A32)
- The Number 1 display controller (A33).

RCP permits manual reversion to single-unit operation for DU's in pilot's and co-pilot's stations, and manual reversion to single-source display for the Air Data System (ADS) and the Attitude and Heading Reference System (AHRS). Such manual reversion actions permit recovering from single unit/system failure where loss of important data on displays occurs.





ICN-39-A-316000-G-A0126-02975-A-01-1

DISPLAY REVERSION CONTROL PANEL

OPERATION

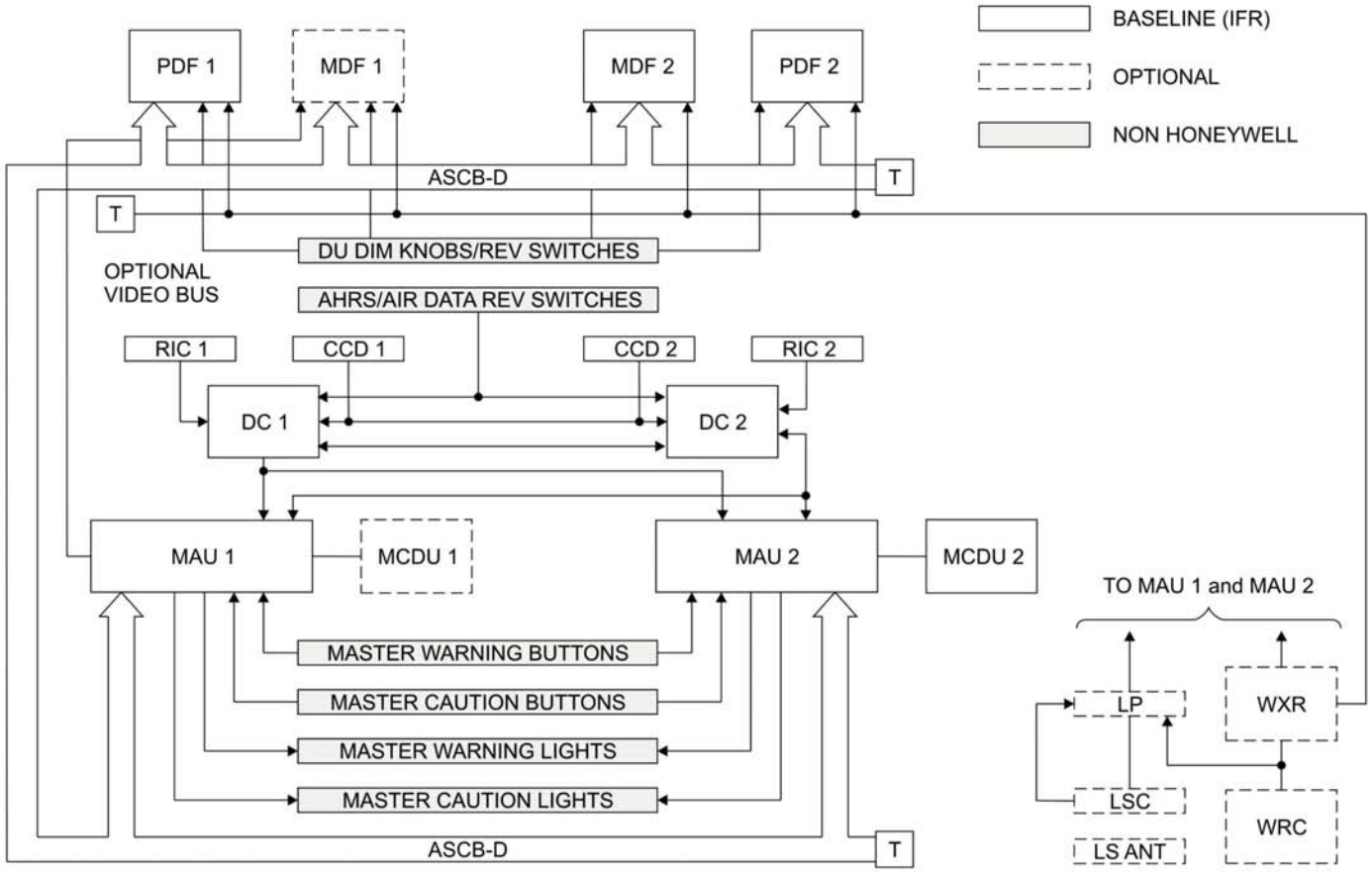
CDS

The central display system (CDS) is the aircraft integrated display system. It gives the flight crew a visual indication of data supplied by many different aircraft systems.
The central display system shows the flight crew data for aircraft control, navigation, and system condition monitoring. The central display system includes the engine indicators and the crew alerting system.
The central display system shows the data that follow:

- attitude and heading (HDG) from the Attitude and Heading Reference System (AHRS)
- airspeed, vertical speed, and altitude data from the Air Data System (ADS)
- navigation data (such as deviations, bearing, distance, etc.) from the navigational radios and from the Flight Management System (FMS)
- guidance data from the Automatic Flight Control System (AFCS)
- active COM and NAV radio frequencies and ATC Transponder code
- engine and aircraft system parameters (such as NG, NF, NR, torque, temperatures, pressures, etc.)
- engine and aircraft system failure and status indications
- digital map data from the FMS and from Digital Map Generator (if installed)
- weather and ground-map data from the weather radar (WXR) system (if installed)
- terrain data from the Enhanced Ground Proximity Warning System (EGPWS) (if installed)
- traffic data from the Traffic Alert and Collision Avoidance System (TCAS) (if installed)
- video images from FLIR and/or cameras (if installed)

The CDS uses the DUs to show all data. The Display Unit is a flat panel screen that uses an active matrix liquid crystal display (LCD) technology. The Display Unit (DU) is a line replaceable unit (LRU) is located in the aircraft cockpit supplying flight, navigation (NAV) and airframe subsystem data. On board are installed four Display Units that receive digital display data from the avionics standard communications bus version-D (ASCB-D).
Each DU is connected directly to the ASCB-D and the Local Area Network (LAN). Each DU receives digital data from the Avionics-Standard Communication-Bus version-D (ASCB-D), generates the graphic information and displays them.
The DU has no user controls or buttons. Functions of the four Display Units are controlled by the following equipment:

- The DU dimmer panel
- The Cursor Control Devices (CCDs)
- The Remote Instrument Controllers (RICs)
- The Display Controllers (DCs)
- The Multifunction Control Display Units (MCDUs)
- The Reversion Control Panel (RCP).



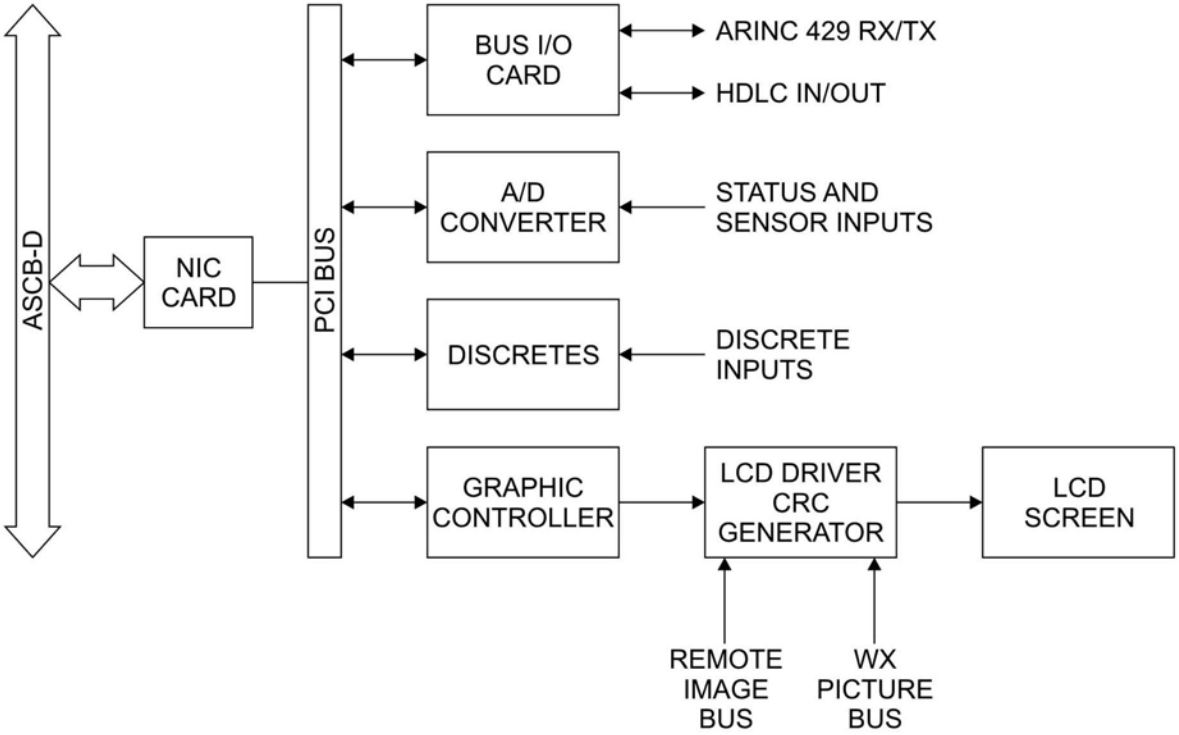
ICN-39-A-316000-G-A0126-01502-A-01-1

SYSTEM OPERATION

DU

The main component of the Display Unit is the LCD module. The LCD module mixes red, green, and blue pixels to show full colour images on the display. A thin film transistor (TFT) controls each element. The TFT gives maximum control and performance for the transmission of each element.
Each display contains the software to show all formats and to connect it to the ASCB-D.
The graphics generator causes a fully anti-aliased image to supply a smooth, step less, painted-on display format.
The template shown on the Display Unit can have one of the following formats:

- PFD. The PFD is supplied to show data as an attitude director indicator (ADI) and a horizontal situation indicator (HSI) at the same time. The ADI and HSI show the heading, attitude and other primary flight data.
- MFD. The MFD is supplied to show data from the primary aircraft systems, NAV, WX, air traffic and other displays.
- Composite. Composite mode is supplied to make sure that essential flight data is available after a failure of one of the aircraft DUs. The composite mode uses the normal PFD layout where possible.



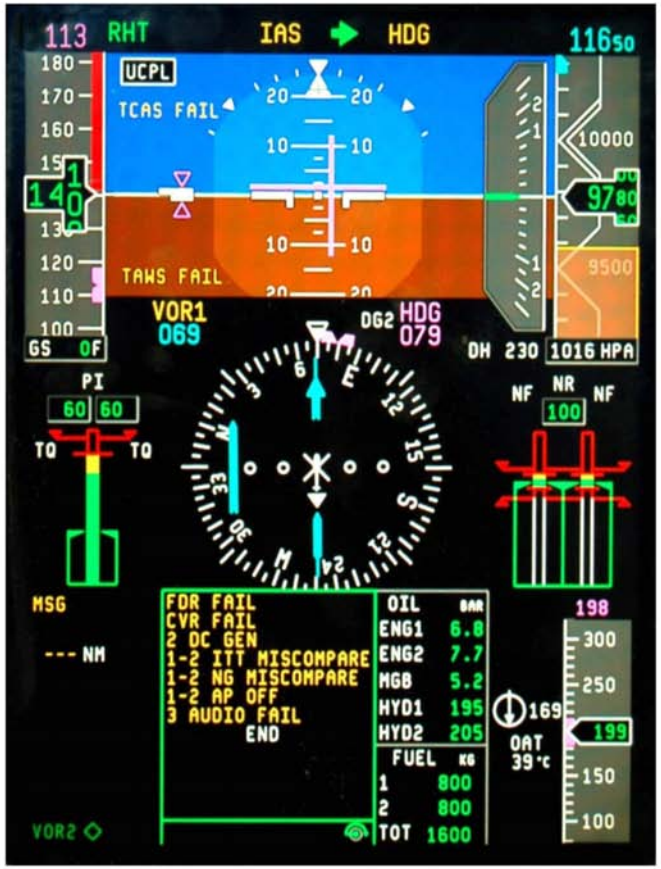
ICN-39-A-316000-G-A0126-01503-A-01-1

DU SYSTEM OPERATION

DU COLOURS

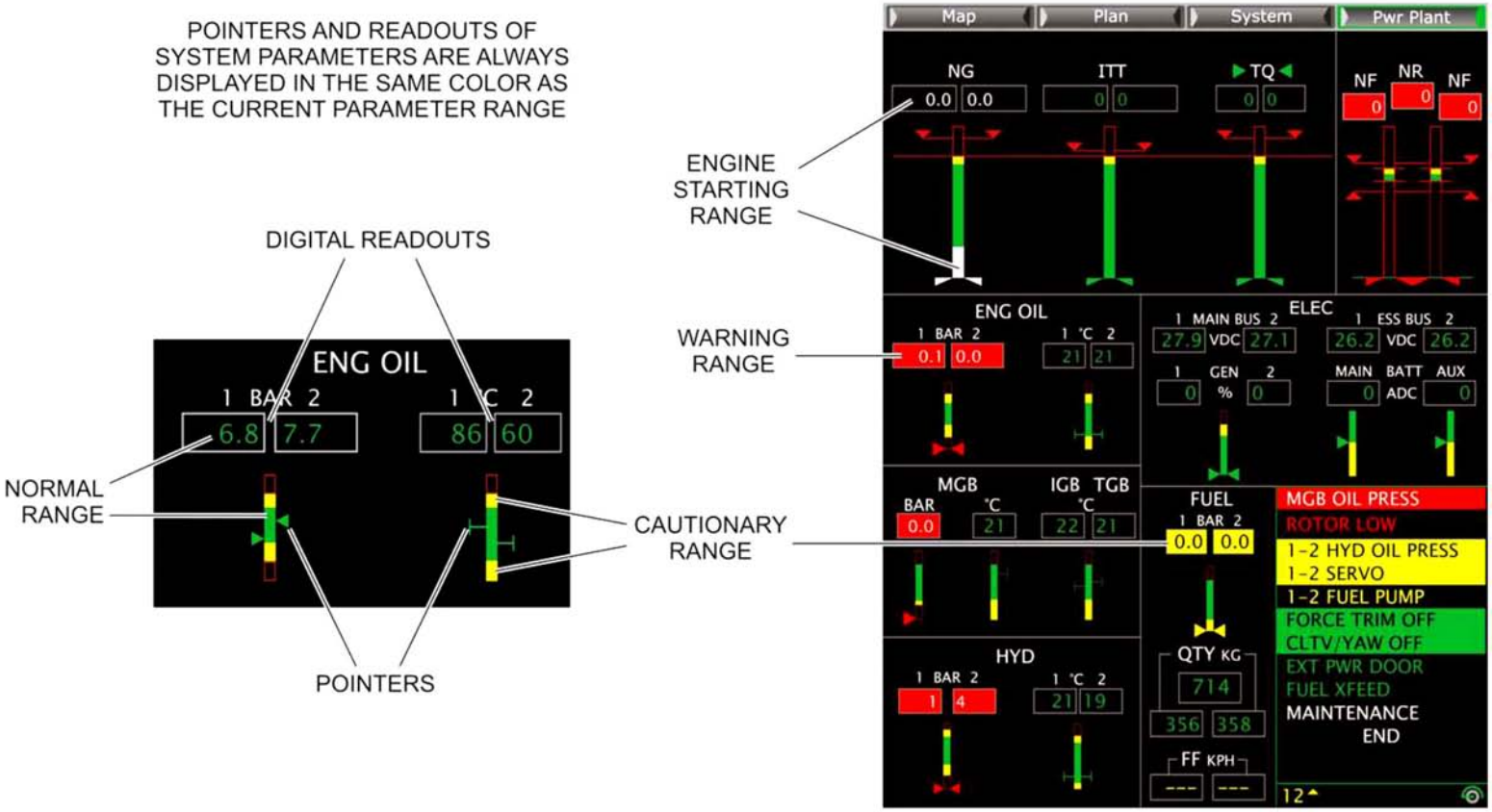
Normally many of the graphic and digital displays change colour when limits are reached or exceeded (usually from white or green to amber or red). Colours used in the PFD and MFD displays follow the rules listed in the table.

NORMAL VIDEO	REVERSE VIDEO	
RED	RED	Warning conditions & messages Flight envelope and system limits Invalid primary instruments
AMBER	AMBER	Caution condition & messages Invalid data Miscompare annunciators
GREEN	GREEN	Normal operating conditions & data Advisory messages Engaged (Active) FD modes ON conditions (synoptic displays)
CYAN	CYAN	Crew selected reference data and cues Primary Navaid data and cues
MAGENTA	MAGENTA	Crew selected reference data and cues when active with Flight Director Primary Navaid data and cues when active with Flight Director
WHITE		Labels, scales and reference symbols Status messages & data Armed FD modes OFF conditions (synoptic displays)



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DU SYSTEM COLORS



ICN-39-A-316000-G-A0126-02976-A-01-1

DU SYSTEM COLORS

CURSOR

Pilot and co-pilot can independently control cursors using their own CCD.
At power up cursors are automatically selected on the two MFD's, to scroll the CAS. Cursors can be toggled between PFD and MFD by using the DISPLAY SELECT buttons.
Cursors are typically shown as green boxes and respond either to the joystick and the associated ENTER button or to any of the SET concentric knobs.



ICN-39-A-316000-G-A0126-02977-A-01-1

CURSOR CONTROL DEVICE

MENU BAR

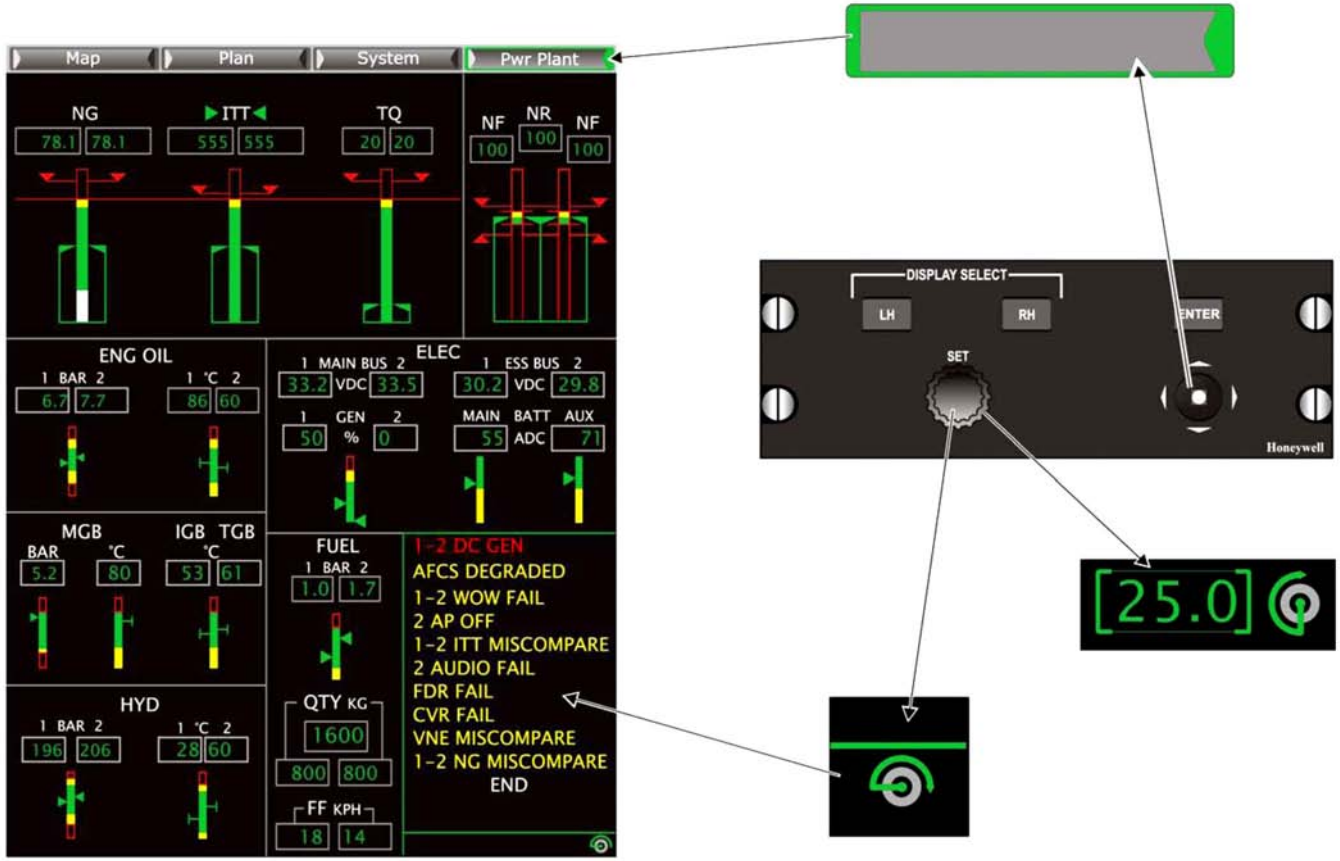
The menu bar is at the top of the MFD. It is used to set the display format of the MFD. The menu bar has four virtual pushbuttons for each menu.
The menu bar has the virtual pushbuttons that follow:

- Pwr Plant
- System
- Map
- Plan.

Each menu has a fold-down sub-menu that lets system modes be set and format details be shown.
To do the selections in the menu bar, the pilots must move the green box cursor with the joystick of the cursor control device (CCD) to the applicable virtual pushbutton, then push the enter button on the CCD.
If a menu virtual pushbutton is selected, the related sub-menu comes in view when the CCD joystick is moved in the down direction. The CCD joystick must be moved up or down to do the selection of the items on the sub-menu list.
PWR PLANT
At power-up, the Pwr Plant format is automatically selected and there is also a cursor on the CAS window to scroll the CAS list.
The Pwr Plant format is the standard format for MFD. The top window shows the primary or secondary engine and system data. When the Pwr Plant format is set for the top window, the top and bottom windows become one window.
The Pwr Plant sets and controls the display of the power plant pages on the MFD. The Pwr Plant has two menu items that follow:

- Main Page
- Analog input/output.

The Main Page shows the digital primary engine data by Electronic Engine Controllers (EEC).
The Analog menu item lets the pilot manually set the display of analog engine data on the MFDs and PFDs. A check mark shows in the check box when the display of analog engine data is set to ON. When the Analog button on the Pwr Plant menu is set, the display system shows analog primary engine data as alternative to the EEC data.



ICN-39-A-316000-G-A0126-02978-A-01-1

MENU BAR

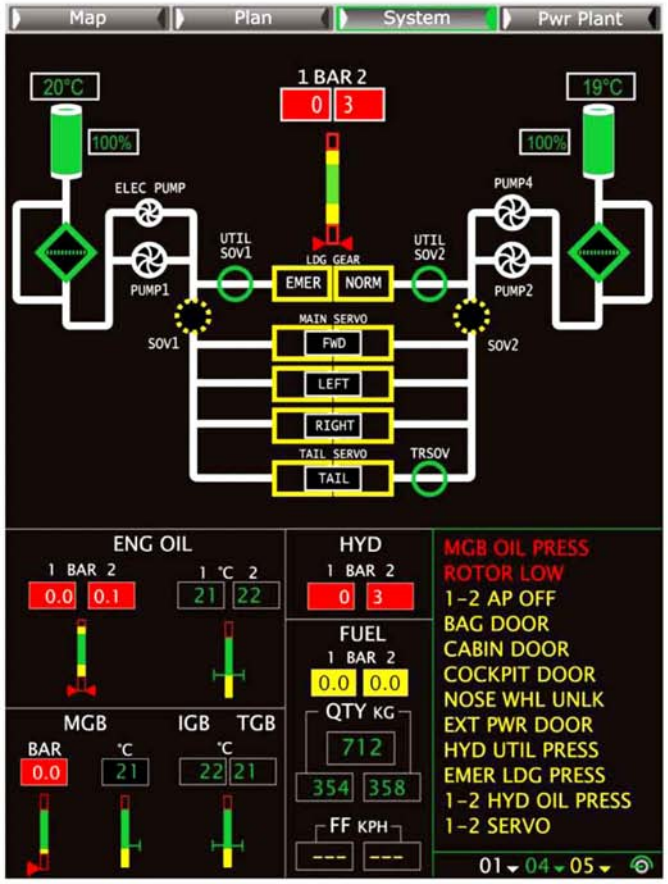
SYSTEM

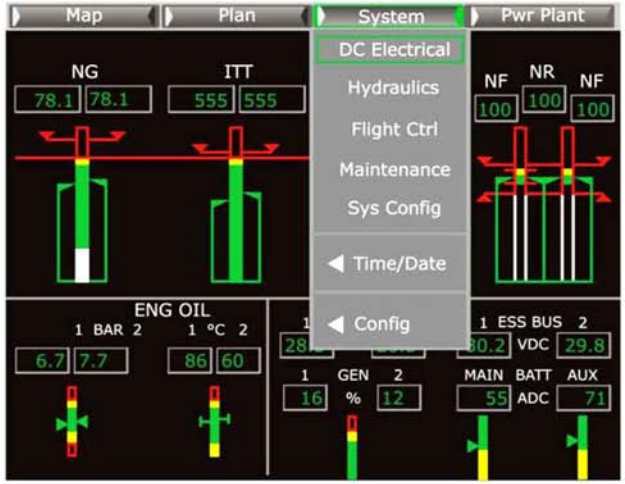
The System menu on the menu bar sets and controls the display of the different aircraft system pages on the MFD.

To see the most recent system page, use the CCD to move the cursor to the System menu button, then move down the joystick to see the sub menu and the synoptic page available. Move the course on the system page required and after press the enter button on the CCD, to see the sys page.

The System pages that follow shows:

- The DC Electrical-menu item to show the DC electrical synoptic-page in the top window of the MFD.
- The Hydraulic menu-item to show the hydraulic synoptic page in the top window of the MFD.
- The Flight Ctrl-menu item to show the flight-control synoptic-page in the top window of the MFD.
- The Maintenance menu item to show the output of the CMC on the MFD in a full-screen format. A selection of the Maintenance menu-item is possible when the Central Maintenance Computer (CMC) is applicable and the aircraft is on the ground.
- The Sys Config menu-item shows the output of the configuration monitoring function in full-screen format. The selection of the Sys Config menu-item is possible only when the aircraft is on the ground.
- The Time/Date menu-item shows the real-time clock. If the global positioning system (GPS) receiver is operating, the system clock is automatically synchronized with the time and date from the GPS receiver automatically. If the GPS receiver is unserviceable, or is not installed, the Time/Date menu-item must be used to set the system clock manually.
- The Config submenu lets the selection of the units that show the parameters of the fuel parameters, barometric pressure, and interturbine temperature (ITT).





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SYS CONFIG

The Software Configuration Monitoring System (SCMS) is a function of the MAU that performs automatic monitoring of the hardware and software configuration items that are connected to the Avionics Standard Communication Bus (ASCB).
In case new software or hardware has been loaded in the system and the new configuration has not been validated, the



caution message is displayed in the CAS window.

Configuration validation is then required.
Pilot or copilot selects the Configuration window via the System / Sys Config drop-down menu to display the current configuration for verification.
When a pilot selects the Configuration submenu, the submenu is grayed out on the other MFD.



ICN-39-A-316000-G-A0126-01511-A-01-1

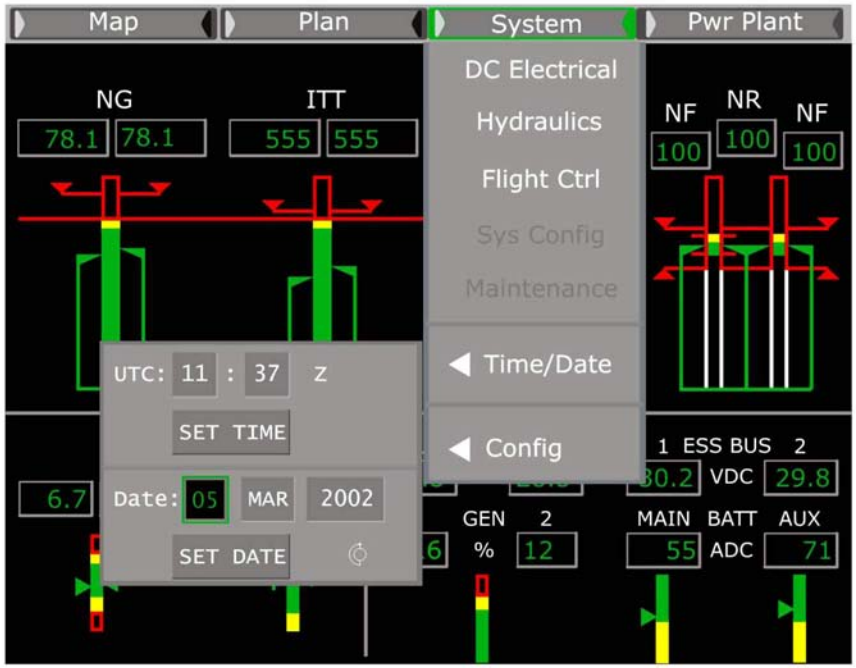
SYSTEM CONFIGURATION PAGE

TIME / DATE SETTING

Moving the joystick in the direction of the arrow when cursor is on the Time/ Date option of the System drop-down menu opens a sub-menu that shows the Primus EPIC® real time clock; the Time/Date submenu also permits manual setting of time and date.
When the GPS receiver is valid and is receiving satellite data, the system clock automatically synchronizes with the time and date from the GPS receiver.
When the system time is synchronized with the GPS, both SET TIME and SET DATE prompts are displayed in gray to indicate that pilots cannot manually update the time and date.
When the system time and date are not synchronized with the GPS, the pilot (or co-pilot) can manually set them by:

- moving the cursor to the parameter that needs modification
- rotating the SET knob on the CCD to change values and
- pressing the ENTER on the CCD with cursor on SET TIME or SET DATE, as appropriate, to save the value

When a pilot selects the Time/Date submenu, the submenu is grayed out on the other MFD.



ICN-39-A-316000-G-A0126-01512-A-01-1

TIME/DATE SETTING

CONFIGURATION SETTING

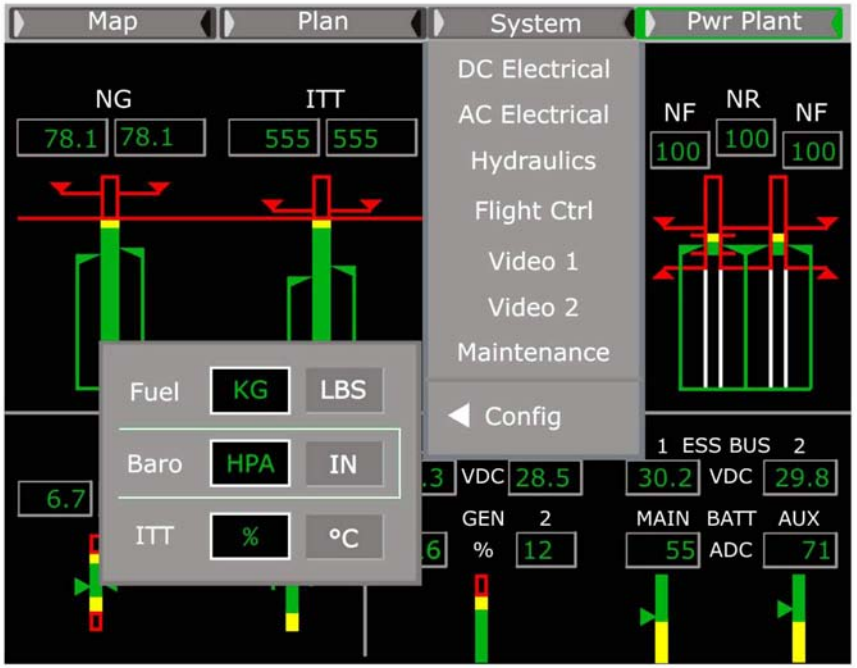
Moving the joystick in the direction of the arrow when cursor is on the Config option of the System drop-down menu opens a sub-menu that permits setting of the following:
Metric Altimeter checkbox

- Fuel..... Unit for Weight data (KG or LBS)
- Baro..... Unit for Barometric Pressure Reference (IN or HPA)
- ITT..... Unit for ITT Indicator (°C or %)

The Metric Altimeter is displayed as an additional digital readout in the top portion of the Barometric Altimeter when the option box is checked.
The selection of Weight data unit affects the following:

- Fuel Quantity Indicator
- Fuel Flow Indicator: unit will be KPH (kg per hour) or PPH (pounds per hour)
- Weight data readouts on the MCDU (see FMS in chapter 34-00)

Changes to the selections are made by using the cursor to highlight the parameter to be configured, and by using the ENTER button on the CCD to toggle between the available states.
When a pilot selects the Config submenu, the submenu is grayed out on the other MFD.
The selections made by a pilot affect all DUs and are memorized (i.e. the configuration selected before removing power is shown again at the next power-up).



ICN-39-A-316000-G-A0126-01513-A-01-1

SYSTEM CONFIGURATION

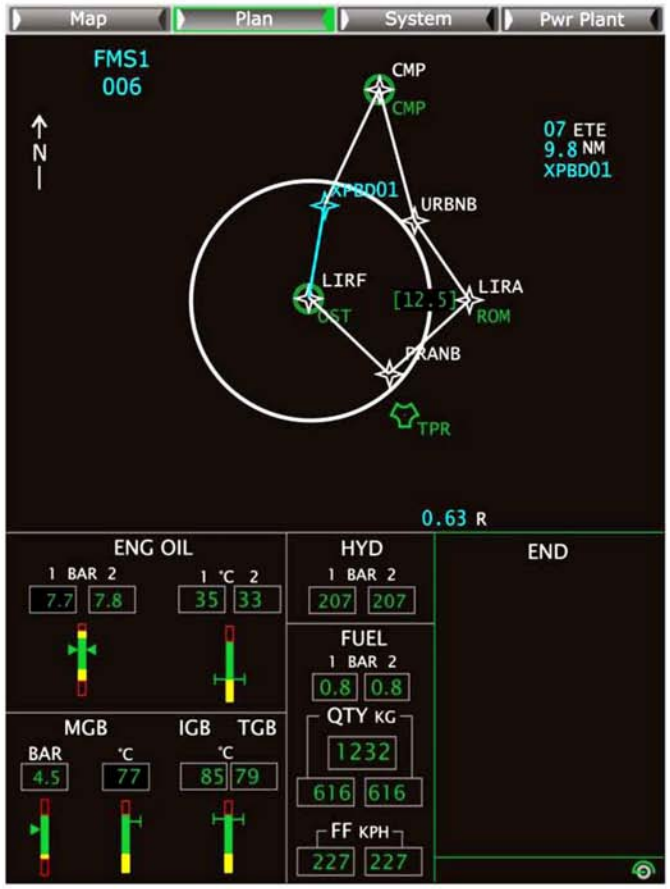
PLAN

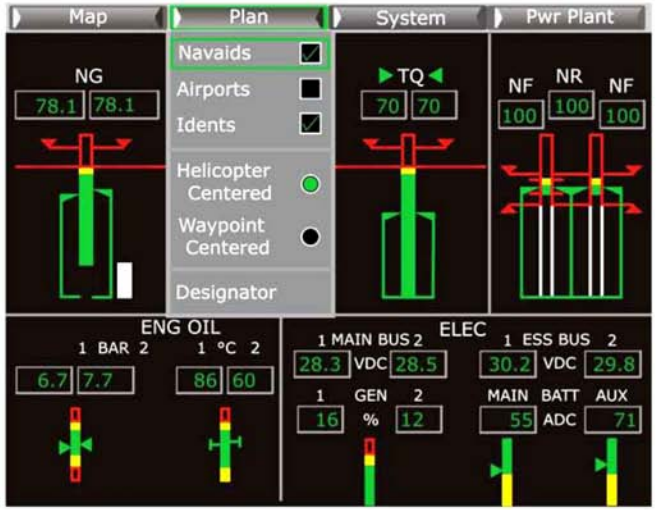
The Plan menu on the MFD menu bar sets and controls the Plan page. The selection of the Plan menu gives the pilots the data that follow:

- The heading refer to the magnetic north
- The navigation Source
- The WPT current
- The WPT Distance
- The ETE (estimate time en-route).
- The Holding Pattern
- The Lateral Deviation

When the Plan menu is open, the menu-items that follow become available for selection:

- Navaids: shows the NAVAIDs on the map
- Airports: shows the airports on the map
- Idents: shows the symbol identifiers on the map
- Helicopter centered: shows the helicopter at the center of the range rings
- WPT centered: shows the WPT at the center of the range rings.
- Designator: starts the designator mode on the Map page.





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PLAN CONFIGURATION

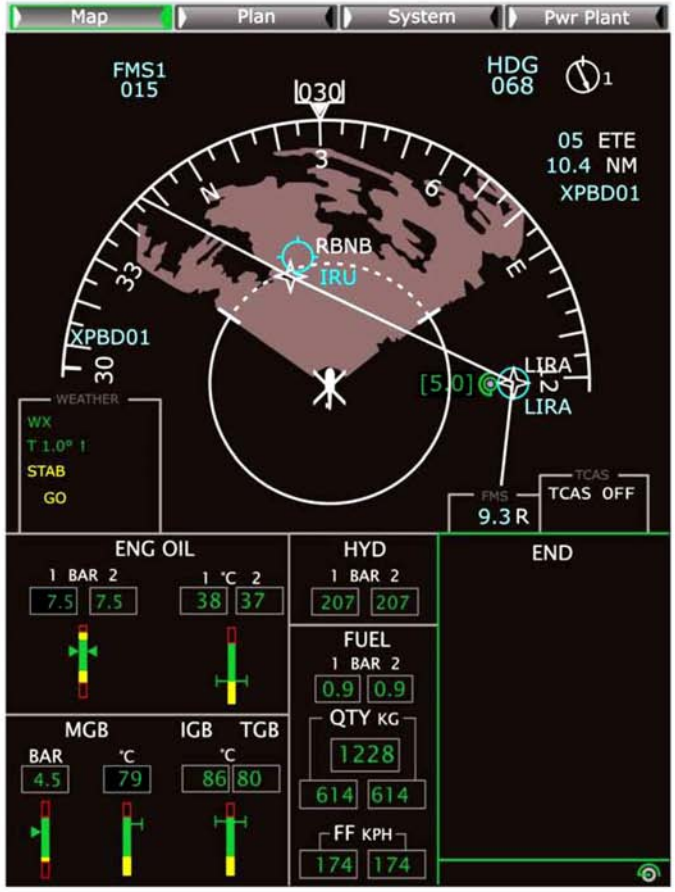
MAP

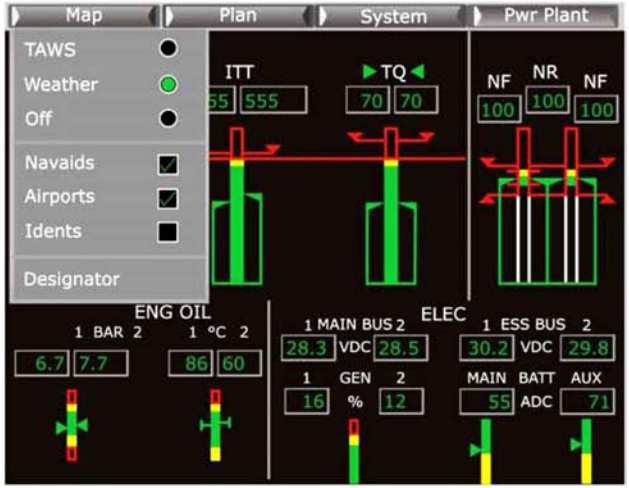
The Map menu on the MFD menu bar sets and controls the Map page. The selection of the Map menu gives the pilots the data that follow:

- The heading (HDG)
- The navigation source
- The current to-WPT
- The to-WPT Distance
- ETE (estimate time en-route).
- The WPT (waypoint), which includes Radials and Non-display Features
- The Holding Pattern
- The Lateral Deviation

When the Map sub-menu is open the menu-items that follow become available for selection:

- Traffic: shows Traffic alert and collision avoidance system (TCAS) data and the configuration menu
- TAWS: shows the data of the terrain-avoidance warning-system (TAWS) which are received from the Enhanced ground-proximity warning-system (EGPWS). The Flight Plan (FP) is shown ON the terrain display
- Weather: shows the WX image on the Map page
- OFF: the TAWS and WX displays are OFF
- Navaids: shows the position and identifiers of navigational aids (NAVAID) found in the map range
- Airports: shows the airports on the map
- Idents: shows an identifier adjacent to the FP WPT symbols shown on the page
- Designator: starts the designator mode on the Map page.





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MAP CONFIGURATION

CENTRAL DISPLAY SYSTEM - CONTROLS AND INDICATORS

CURSOR CONTROL DEVICE (CCD)

1 DISPLAY SELECT pushbutton

- LH pressed ..... activates the cursor(s) on the LH display of the relevant pilot's station (PLT = MFD; CPLT = PFD)
- RH pressed ..... activates the cursor(s) on the RH display of the relevant pilot's station (PLT = PFD; CPLT = MFD)

2 ENTER pushbutton

- MFD selected ..... executes the highlighted menu selection or toggles between highlighted menu options; deactivates the designator (Map or Plan formats)
- PFD selected ..... toggles the highlighted active/ standby values

3 JOYSTICK

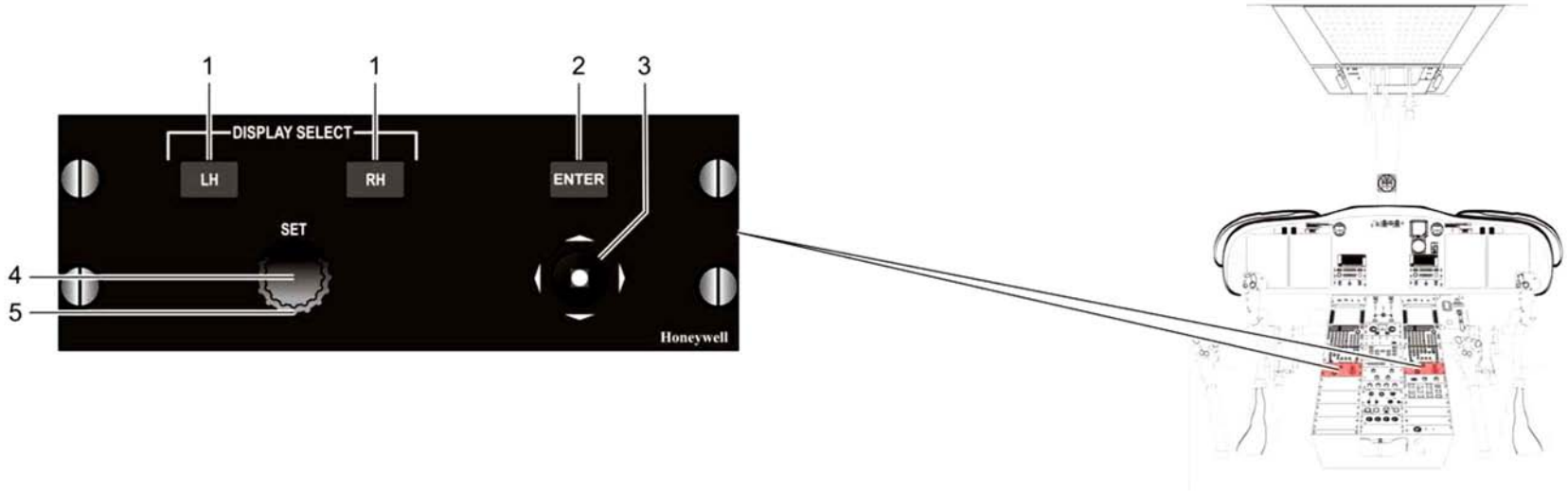
- MFD selected ..... controls cursor position in the drop-down menu or the designator position (when active)
- PFD selected ..... controls cursor position in the COMM/IDENT window

4 SET inner knob

- MFD selected ..... scrolls the CAS window messages
- PFD selected ..... adjusts decimals of the highlight standby frequency or the last two digits of XPDR code

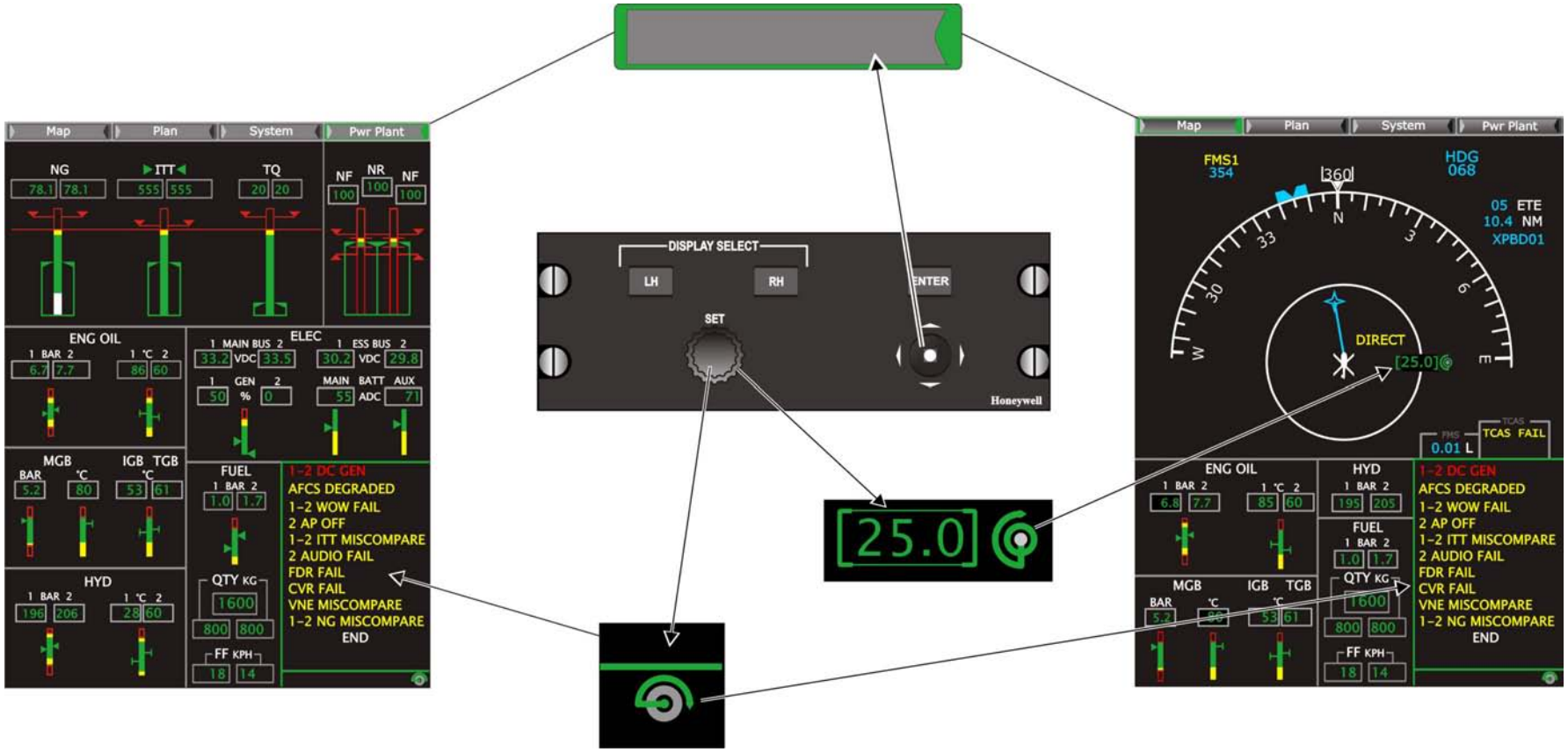
5 SET outer knob

- MFD selected ..... sets the scale range (Map or Plan formats) or the highlighted value in the drop-down menu
- PFD selected ..... adjusts units of the highlight standby frequency or the first two digits of XPDR code

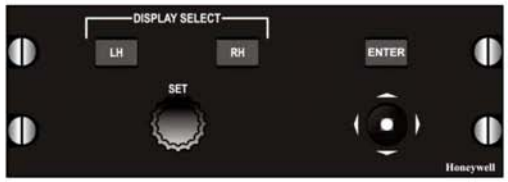


ICN-39-A-316000-G-A0126-02982-A-01-1

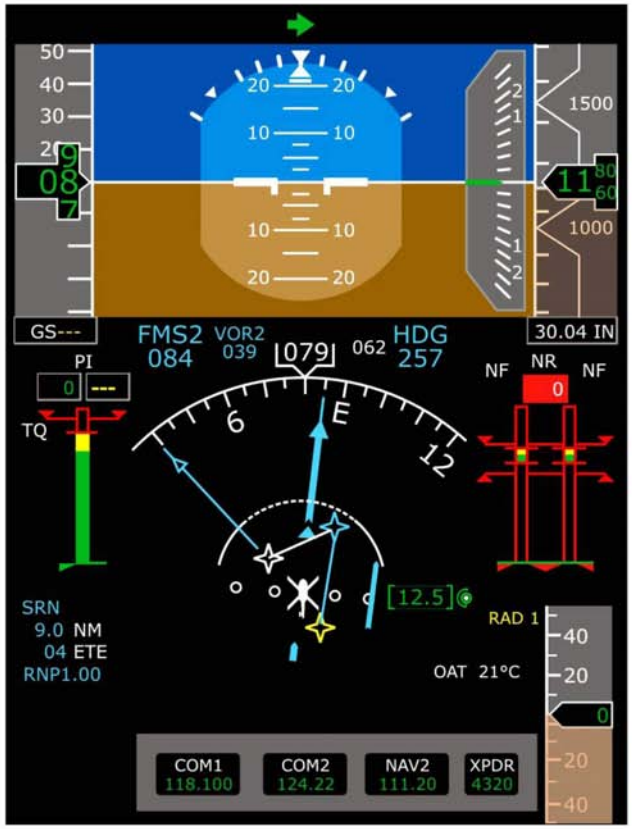
CURSOR CONTROL DEVICE (CCD) – CONTROLS AND INDICATORS



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ICN-39-A-316000-G-A0126-01518-A-01-1

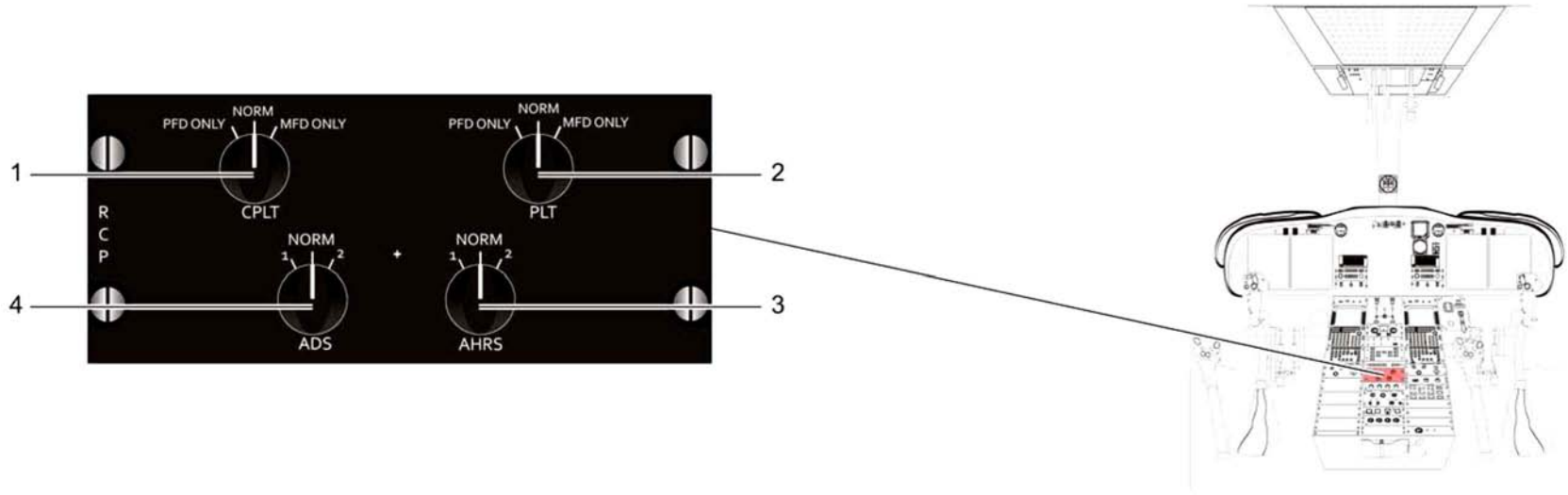
REVERSION CONTROL PANEL (RCP)

1 CPLT selector
- NORM ..... automatic reversion selected
- PFD ONLY ..... the copilot MFD is turned off; the copilot PFD is set to composite format
- MFD ONLY ..... the copilot PFD is turned off; the copilot MFD is set to composite format

2 PLT selector
- NORM ..... automatic reversion selected
- PFD ONLY ..... the pilot MFD is turned off; the pilot PFD is set to composite format
- MFD ONLY ..... the pilot PFD is turned off; the pilot MFD is set to composite format

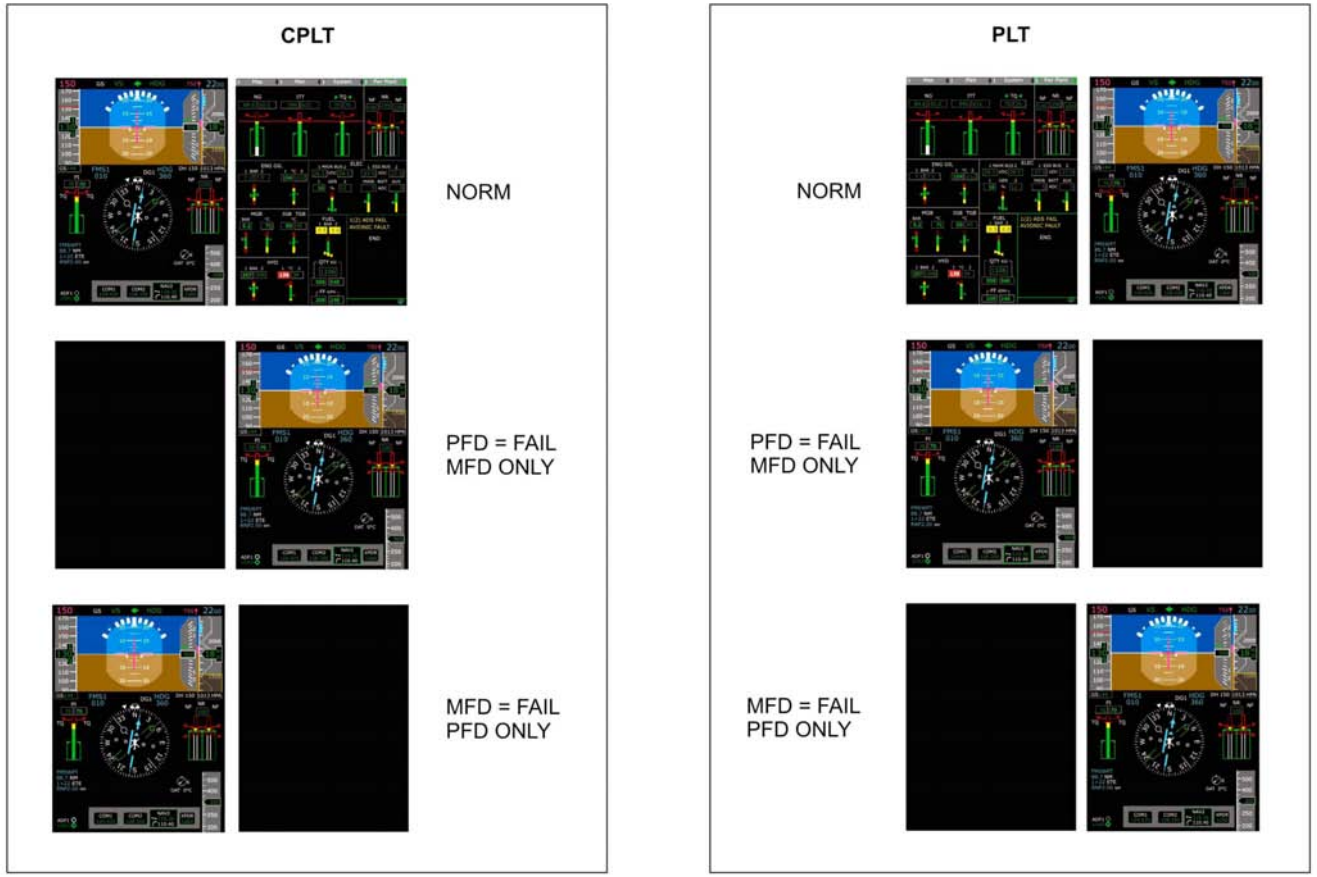
3 AHRS selector
- NORM ..... each pilot's station displays the on-side AHRS data
- 1 ..... both pilot's stations display the AHRS 1 data
- 2 ..... both pilot's stations display the AHRS 2 data

4 ADS selector
- NORM ..... each pilot's station displays the on-side ADS data
- 1 ..... both pilot's stations display the ADS 1 data
- 2 ..... both pilot's stations display the ADS 2 data



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REVISION CONTROL PANEL (RCP) – CONTROLS AND INDICATORS



ICN-39-A-316000-G-A0126-01520-A-01-1

DU DIMMING CONTROL PANEL

1 CPLT PFD potentiometer

- MIN ..... set the copilot PFD to the minimum brightness value
- MAX ..... set the copilot PFD to the maximum brightness value

2 CPLT MFD potentiometer

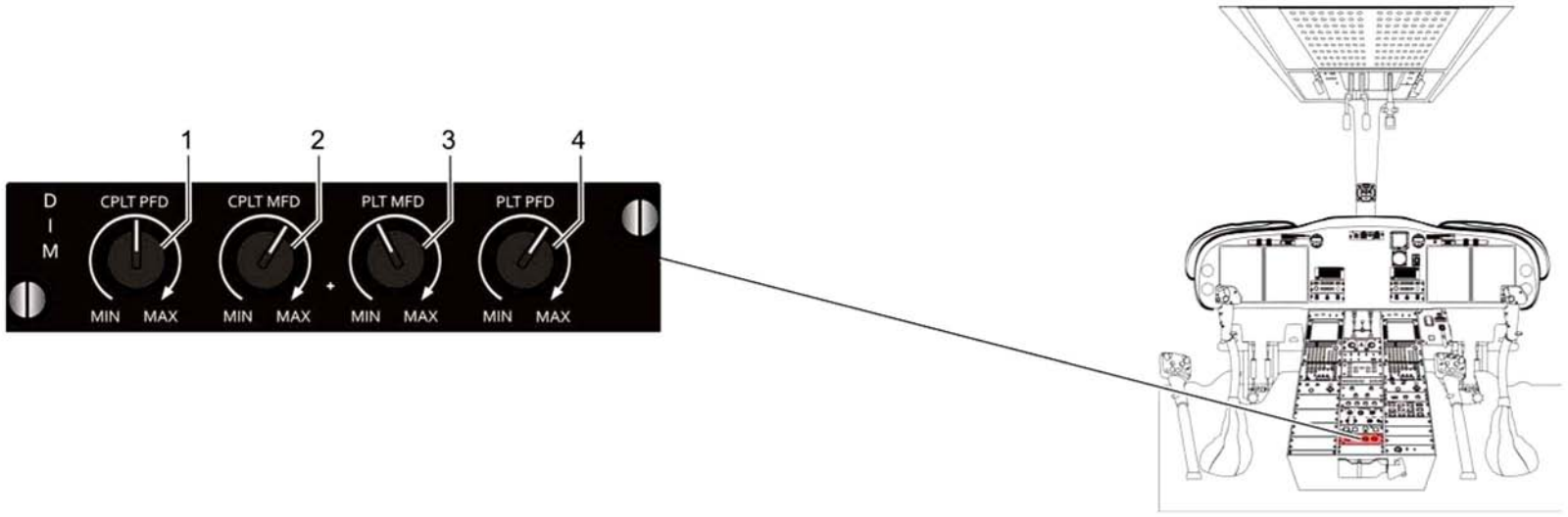
- MIN ..... set the copilot MFD to the minimum brightness value
- MAX ..... set the copilot MFD to the maximum brightness value

3 PLT MFD potentiometer

- MIN ..... set the pilot MFD to the minimum brightness value
- MAX ..... set the pilot MFD to the maximum brightness value

4 PLT PFD potentiometer

- MIN ..... set the pilot PFD to the minimum brightness value
- MAX ..... set the pilot PFD to the maximum brightness value



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DU DIMMING CONTROL PANEL (DCP) – CONTROLS AND INDICATORS

REMOTE INSTRUMENT CONTROLLER (RIC)

1 COURSE select knob

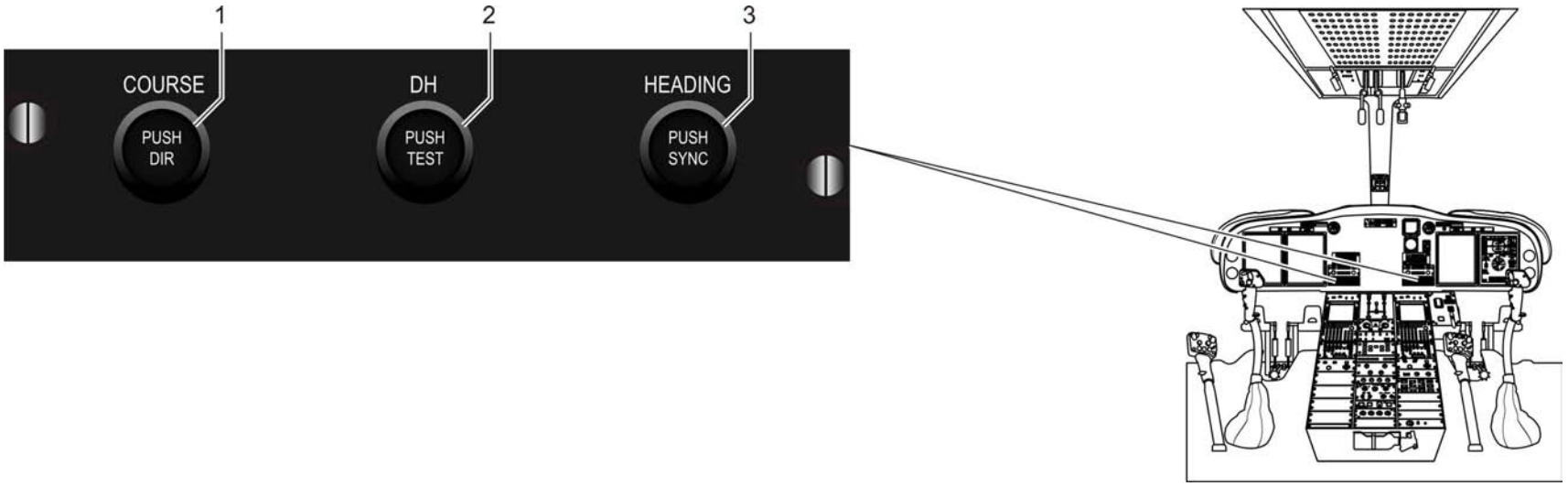
- rotated ..... selects the desired course when VOR/LOC is selected as the Primary Navaid or as the PRV (Preview) on the on-side HSI
- PUSH DIR button pressed ..... selects the Direct Course, i.e. synchronizes the selected course to the VOR bearing, when VOR is selected as the Primary Navaid or as the PRV (Preview) on the on-side HSI

2 DH select knob

- rotated ..... selects the desired course when VOR/LOC is selected as the Primary Navaid or as the PRV (Preview) on the on-side HSI
- PUSH TEST button pressed ..... selects the Direct Course, i.e. synchronizes the selected course to the VOR bearing, when VOR is selected as the Primary Navaid or as the PRV (Preview) on the on-side HSI

3 HEADING select knob

- rotated ..... controls the Heading Set bug on both HSI's.
- PUSH SYNC button pressed ..... synchronizes the selected Heading Set bug on both HSI's to the current aircraft heading of the coupled attitude and heading reference system (AHRS) or to the heading of the on-side AHRS when the flight director is not coupled



ICN-39-A-316000-G-A0126-02985-A-01-1

REMOTE INSTRUMENT CONTROLLER (RIC) – CONTROLS AND INDICATORS

DISPLAY CONTROLLER (DC)

1 Circle BRG (LH) pushbutton

- pressed ..... selects VOR1 for display on circle (white) bearing pointer on the on-side HSI. Subsequent presses toggle between the different sources following the sequence: OFF → VOR1 → ADF1 → DF (if installed) → FMS1 → OFF

2 NAV pushbutton

- pressed ..... selects the on-side short-range navigation source (VOR/LOC) as the Primary Navaid on the on-side PFD. Subsequent presses toggle between the on-side and cross-side VOR/LOC.

3 PRV (Preview) pushbutton

- pressed ..... when LNAV is selected as the Primary Navaid allows to preview the on-side VOR/LOC deviation to be displayed on a secondary CDI on the on-side HSI. Subsequent presses toggle between the different sources following the sequence: PRV OFF → on-side VOR/LOC → cross-side VOR/LOC → PRV OFF. Note: ineffective if pressed when short-range navigation source (VOR/LOC) is selected as the Primary Navaid

4 LNAV pushbutton

- pressed ..... selects the on-side long-range navigation source (FMS) as the Primary Navaid on the on-side PFD. Subsequent presses toggle between the on-side and cross-side FMS.

5 Diamond BRG (RH) pushbutton

- pressed ..... selects VOR2 for display on the diamond (green) bearing pointer on the on-side HSI. Subsequent presses toggle between the different sources following the sequence: OFF → VOR2 → ADF2 → DF (if installed) → FMS2 → OFF

6 HSI pushbutton

- pressed ..... toggles between FULL Compass and ARC modes on the on-side HSI.
  If FD HOV mode is available, it toggles between HSI modes in the following sequence:
  FULL Compass → ARC → HOV → FULL Compass

NOTE C

if HOV mode is engaged, the HSI button is ineffective

7 MAP pushbutton

- pressed ..... selects ARC+MAP mode on the on-side HSI.
  Subsequent presses toggle MAP on and off.

8 WX or WX/TERR pushbutton

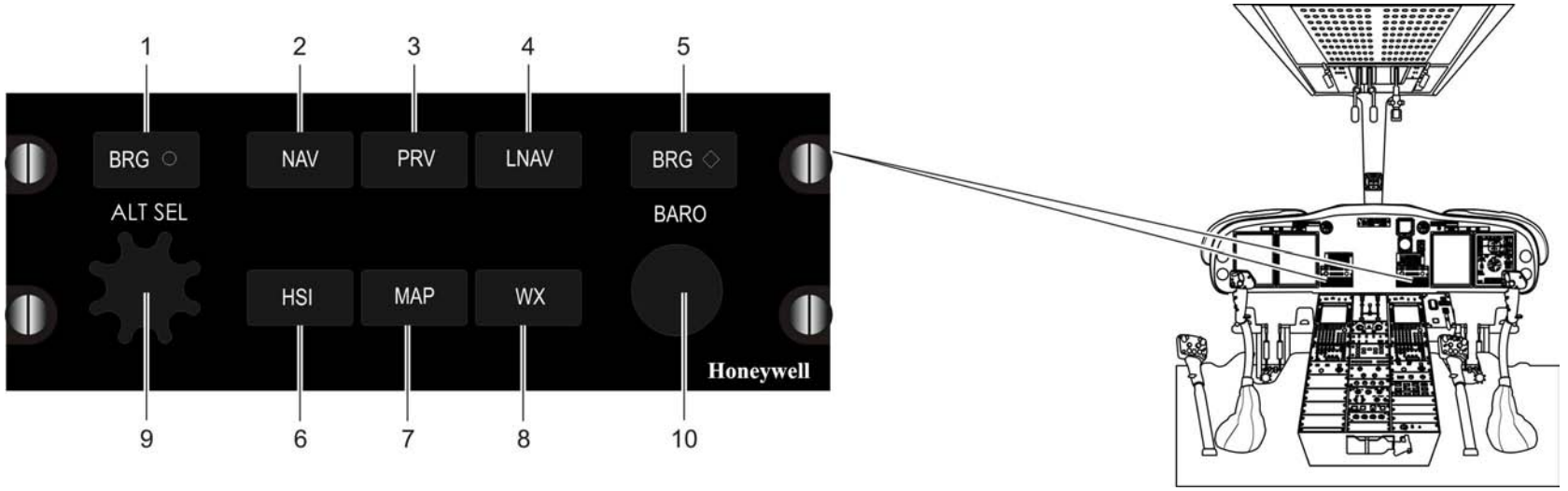
- pressed ..... (if Weather Radar is installed) selects ARC+WX mode on the on-side HSI.
  Subsequent presses toggle WX on and off, or between HSI modes in the following sequence:
  ARC → ARC+WX → ARC +TERR (is EGPWS is installed) → ARC

9 ALT SEL knob

- rotated ..... sets the Selected Altitude reference (bug + digital readout) on both barometric altimeters

10 BARO knob

- rotated ..... sets barometric pressure reference for the on-side barometric altimeter



ICN-39-A-316000-G-A0126-02986-A-01-1

DISPLAY CONTROLLER (DC) – CONTROLS AND INDIATORS

Three buttons labelled HSI, MAP, and WX (or WX/TERR) operate together to control the format of the HSI.
The HSI formats that can be selected are the following:

FULL (Default format)	Displays a conventional HSI with a full 360° compass rose
ARC	Displays an expanded arc of the compass rose extending to ±45° across heading
HOV (*)	Displays the HSI HOV mode with a velocity vector
HOV+360° MAP (*)	Displays HOV mode with velocity vector with 360° FMS map overlay
ARC+MAP	Displays the ARC mode with an FMS map overlay
ARC+WX (**)	Displays the arc format with a WX overlay
ARC+TERR (***)	Displays the arc format with EGPWS terrain overlay
ARC+MAP+WX (**)	Displays the arc mode with FMS map and WX overlays
ARC+MAP+TERR (***)	Displays the arc format with FMS map and EGPWS terrain overlays

NOTE D

(*)
Available only if Enhanced AFCS is installed.
(**)
Available only if Weather Radar (WX) is installed.
(***)
Available only if Enhanced Ground Proximity Warning System (EGPWS) is installed.



ICN-39-A-316000-G-A0126-01524-A-01-1

PFD ATTITUDE INDICATIONS

The ADI is an artificial horizon, pitch tape, flight-path vector, roll pointer/scale and aircraft symbol. The ADI extends from the airspeed tape to the altitude tape with a cut sphere outline. The PFD has the attitude indications that follow:

- The Artificial Horizon
- The helicopter Symbol
- The Pitch Tape
- The Excessive Pitch Indications
- The Roll Scale/Roll Pointer
- The Slip/Skid Display
- The Attitude Source Indication
- The Excessive Attitude Declutter
- The Attitude Miscompare Indication.

The artificial horizon has a line on it that simulates the horizon. The blue area above the horizon line simulates the sky, and the brown area below the horizon line simulates the ground. The horizon line always agrees with the earth's horizon, while the pitch and roll movements of the helicopter show around it.
The attitude-reference helicopter-symbol is at the centre of the artificial horizon and does not move. The pitch tape shows the pitch attitude of the helicopter.
The attitude roll pointer and scale show the aircraft roll attitude. The scale has thick marks at ±10, ±20, ±30, ±60 degrees and a triangle at 0 and ± 45 degrees.
The bottom half of the roll pointer is a slip/skid (lateral acceleration) annunciator. The skid pointer changes colour to amber when the lateral acceleration is more than 0.1 g.
The attitude indication source (AHRS1 or AHRS2) shows at the 11 o'clock-position in the related ADI. If the pilot and copilot set the same AHRS source, the indication colour is amber. If the pilot or copilot does not use the default source for their side, the indication colour is white.

When too much attitude occurs, the display does not show the less important items. Too much attitude occurs with the conditions that follow:

- The bank angle is more than ±65 degrees
- The pitch angle is more than 30 degrees up
- The pitch angle is more than 20 degrees down.
- The display does not show the items that follow:
- The Flight Director (FD) mode indications and command bars
- The marker beacons
- The vertical deviation scale, printer, and indication
- The speed bugs and readout
- The groundspeed
- The FD fail flag, lateral/vertical deviation flags, speed/altitude target flags, and radio altitude flags
- All the comparators and indicator, Attitude, Airspeed, and Altitude stay in view.
- The Monitor Warning System (MWS) compares the attitude data from the two AHRS.

When an attitude miscompare is sensed, the attitude display shows the ATT indication in its top right area. The indication flashes for the first 5 seconds, then its color changes to amber on a black background.
The indication goes off when the attitude miscompare is no more sensed. PITCH comes in view when the pitch attitudes of the two AHRS are different. ROLL comes in view when the roll attitudes of the two AHRS are different. ATT comes in view if the pitch and the roll attitudes of the two AHRS are different.

PFD AUTOPILOT AND FLIGHT DIRECTOR INDICATIONS

The PFD supplies the Autopilot (AP) and Flight Director (FD) items that follow:

- The FD command bars

- The FD mode indication
- The coupled FD indication
- The vertical deviation display
- The Glideslope (GS) display
- The GS miscompare indication
- The collective cue display
- The radio altitude tape
- The radio-altitude source selection
- The Decision Height (DH)
- The marker beacons.

The FD bars comes in view in the middle of the attitude display when the FD is engaged. One bar is vertical and the other bar is horizontal.
The FD mode indication comes into view near the attitude display. The change from the armed to locked condition causes the locked indication mode to flash for 5 seconds.
The related indication comes in view by an arrow above the attitude sphere.
The vertical deviation scale comes in view on the right attitude display when localizer (LOC) or the flight management system (FMS) is the primary NAV source.
The radio altitude tape is in the lower right corner of the PFD. It shows altitudes less then 1000 feet. The radio-altitude digital-readout below the triple-tach shows altitudes less then 2500 feet.
The resolution of the digital readout is 1 foot from -20 to +250 feet and 10 feet from +250 to +2500 feet. The PFD does the automatic source selection for the radio altitude data.
The red RAD indication comes in view when the radio altitude becomes not applicable.
When the EDS software senses a radio altitude miscompare, a RAD indication comes in view in the amber reverse video on top of the radio altitude scale.

The DH digital readout comes in view below and on the right of the attitude display. The display control function sets the DH. The data range is the same as the radio altitude range (<2500 feet). The first turn of the control-knob sets the display to 200 feet. The DH display is not into view when the decision height data is less than the 20 feet. It is also not in view when the radio altitude is equal to DH +100 feet climb.
During the descent, when radio altitude is equal to DH +100 feet, an empty box comes in view on the top right part of the attitude display.
The indications related to the inner, middle, and outer marker beacons are also available on the attitude display.
The marker beacon indication is a letter. The I is for the inner marker, M for the middle marker and O for the outer marker. The NAV (very high frequency omnidirectional radio range localizer [VOR/LOC]) receiver sets the marker beacon data. The marker-beacon visual indication is on the lower left corner of the attitude display.

ALTITUDE INDICATIONS

The PFD supplies the indications that follow:

- The altitude display (includes the altitude tape and altitude digital readout)
- The altitude trend vector
- The altitude barometric-correction indication
- The low altitude indication
- The metric altitude
- The altitude preselect display (includes altitude preselect bug)
- The low altitude awareness
- The FMS Vertical Navigation (VNAV) Altitude Target and Bug
- The Air Data System (ADS) Source Indication
- The Altitude Miscompare Indication.

Read the aircraft altitude on a tape and a scale on the right side of the attitude display. The tape shows the data in intervals in the range of ±550 ft, with the highest values at the top of the tape. It also has a digital readout at its center, which shows the altitude from -2000 to +60000 ft. The NEG legend is in view above the digital readout if the altitude is negative.
The altitude "trend" vector, on the left side of the altitude tape, gives a visual indication of the helicopter possible altitude after the subsequent 6 seconds.
The barometric correction indication comes in view below the attitude tape. This indication can show inches of mercury (inHg) or hecto pascals (hPa). A label (IN or HP) tells you which unit of measure the system uses. The indication resolution is 0.01 inHg or 1.0 hPa.
The altitude (given in meters) comes in view above the applicable value in the altitude window with a M on the right of the digital readout. The resolution is 1 meter.
The selection of the altitude functions occurs through the MFD menu.
The digital readout of the altitude and the altitude bug come in view above the altitude tape. Its range is from of -2000 to +51000 feet.
The usual colour of the digital readout and bug is the cyan.
When the bug is off the scale, it becomes an arrow which points to the direction in which the off scale condition occurs. The shape of an arrow is the off scale indication for the bug.
The radio altitude gives the low altitude display on the altitude tape. When the radio altitude is less than 550 feet, the lower part of the altitude tape starts to show a brown colour. The low altitude indication is shown linearly between 550 and 0 feet. The lower part of the altitude tape becomes fully brown when the radio altitude is 0 feet.
The altitude-target digital readout comes in view when:

- The VNAV Altitude Alert is enable
- The VNAV Altitude Target is valid.

When one of the VNAV modes that follow is engaged:

- The flight level change
- The altitude select
- The altitude hold, or vertical path.

The ADS indications come in view on top of the attitude display and at the right of the roll-zero index marker. When the two pilots use the same ADS, (ADS1 or ADS2) amber indications come into view on the two PFDs.
When an altitude miscompare is sensed, an amber reverse-video ALT-indication comes in view on the top half of the altitude tape. This indication is above all scale marks. The indication changes to the amber reverse video for the first 5 seconds. The indication goes out of the view when the altitude miscompare condition is not not longer available.
When the EDS software senses a BAROALTITUDE miscompare, the ALT indication changes to the cyan reverse video for the first 5 seconds. If applicable altitude data are not available from the ADS function, this causes the removal of the ALT indication from the display.
With the GS mode engaged, the altitude indication is not shown.

AIRSPEED AND VERTICAL SPEED INDICATORS (VSI)

The VSI indications are as follows:

- The airspeed tape
- The airspeed rolling drum display
- The overspeed/autorotation speed warning indication
- The airspeed trend vector
- The airspeed reference bug and readout
- The VNE (Never-exceed velocity)
- The airspeed miscompare indication
- The vertical speed dial display
- The vertical speed target bug and readout.

The airspeed tape comes in view on the left of the attitude display. The airspeed tape window shows the airspeed in ±40 knots.

The airspeed tape has the higher speeds on top and the lower ones at the bottom, and thick marks at intervals of 10 knots.
The tape has a label each 10 knots below 200 knots and each 20 knots at more then 200 knots.
The airspeed digital readout is a window at the centre of the airspeed tape. The last digit changes in 1 knot increments. The right edge of the digital display has a notch for the location of the airspeed reference bug.
The airspeed trend vector-display is an indication of acceleration direction. This indication comes into view as a thermometer type scale to the right side of the airspeed tape.
The trend vector is not in view when the helicopter acceleration is less than 0.1 knots for each second.
The airspeed reference-digital readout comes in view on the airspeed tape when the FD airspeed held mode is set.
A red barber pole line gives the indication of the VNE on the airspeed tape when All Engines Operate (AEO) correctly. The VNE comes into view as a red/white barber pole line in "One Engine Inoperate" (OEI) condition or helicopter "auto-rotation". The OEI/auto-rotation VNE is the offset from the VNE indication at AEO with a parameter that is in memory on the APM. A thermometer, in a colour that agrees with the VNE line, is on the airspeed tape. The thermometer comes in view when the airspeed is more than VNE less 5 knots.

HEADING INDICATIONS

The AHRS supplies the data for the heading indications.
The heading display has four formats as set through the control function. The full compass shows a 360 degree heading display that moves counterclockwise for positive heading angles that increase. The arc mode comes in view in the centre and a 90 degree section of the full compass on the heading.
The annunciators are as follow:

- The heading scale
- The heading source selection and indication
- The heading select display
- The heading bug out of view
- The heading miscompare indication
- The drift bug display
- The course pointer
- The course select/desired track pointer
- The course select/desired track readout
- The lateral deviation display
- The localizer deviation scaling
- The VOR deviation
- The FMS lateral deviation
- The to/from display
- The VOR NAV source
- The FMS NAV source
- The NAV source indications
- The LOC Miscompare Indication.

The heading scale comes in view on a compass that is below the attitude display.
When the full compass format is set, the compass shows a 360 degrees card. In the arc mode, the heading scale shows 45 degrees to the left and right the applicable heading.
The two AHRSs supply the heading data. The heading source comes in view adjacent to the set heading indication when their source is the same or the DG mode is set.
When the flight director operates in the heading select mode, it supplies signals to align the helicopter to the heading set by the heading bug.

The heading select bug has the shape of a rectangle open on the lower side with a triangle on top. It moves around the external of the compass card. A digital readout shows the heading set with the heading bug.
The heading bug goes out of the view in the arc mode. At the point the bug goes out of the view, an arrow shows the compass card that identifies the shortest direction to the bug. When the bug comes into view, the arrow goes out of the view.
After the supply of data from a SRN source (VOR/LOC) the set NAV source comes in view above the digital CRS/DTK readout (course set/desired track).
The VOR1 or VOR2 indication comes in view when the NAV source is tuned to a VOR frequency. When the NAV source is tuned to a localizer frequency, the LOC1 or LOC2 indication comes into view.
The VOR1 or LOC1 indication comes in view if the source is the copilot's system. The VOR2 or LOC2 indication if the source is the pilot's system. If the data are from the FMS, the
FMS1 or FMS2 come in view. If only one FMS is installed, the indication is FMS.
The VOR indication is in the view when VOR/LOC is set and the NAV source is not tuned to a localizer frequency. Differently the LOC indication comes in view when the NAV source is tuned to a localizer frequency.
The NAV sources are as follows:

- The VOR1
- The VOR2
- The LOC1
- The LOC2
- The FMS1
- The FMS2 (optional).

When a localizer miscompare is sensed, a LOC indication comes in view in the lower half of the compass card, below the helicopter symbol.

The LOC indication comes in view in the reverse video for 1 second and for 0.5 seconds is OFF for the first 5 seconds. When a localizer miscompare is no more sensed, the indication goes out of the view.

BEARING AND FMS INDICATIONS

The central display system receives the bearing data from the VOR, automatic direction finder (ADF) and FMS through the ASCB-D data bus. The VOR uses magnetic (MAG) headings. The FMS uses true headings. The ADF data shows the bearing of the signal from the aircraft.
If the FMS is in the view against a MAG compass card, the MAG variation it is applied to the true heading to get the MAG heading. The FMS bearing pointer then shows the MAG heading.
The bearing and FMS indications are as follows:

- The bearing pointers
- The bearing-pointer source indication
- The TO waypoint identifier
- The TO waypoint-distance readout
- The Estimated Time Enroute (ETE) readout
- The groundspeed (if available).

The bearing pointers come in view when applicable and set by the control function. The circle bearing pointer is used for the left side sources. The diamond bearing pointer is used for the right side sources.
The bearing-pointer source indications come in view near the lower right corner of the PFD. The indications are:

- The VOR1
- The VOR2
- The ADF1 (optional)
- The ADF2

- The FMS
- The FMS1 (if second optional FMS is installed)
- The FMS2 (optional).

If a VOR/LOC receiver is set and the equipment (DME) station is available, the station of the DME identifier comes in view. When the FMS is set as the primary NAV source, the identifier of the subsequent FMS waypoint comes in view.
The DME distance comes in view when the VOR/LOC receiver is in operation and the primary NAV source and a DME station is available. The DME H comes in view above.
the DME distance readout when the DME is tuned to an applicable VOR station.
When the DME is set in the hold mode and is not synchronized with the VOR, the H indication comes in view.
The full scale of the distance digital readout is 999 NM. From 0 to 99.9 NM, the resolution of the shown data is 0.1 NM.
If the distance is more than 99.9 NM, the resolution is 1 NM. The letters NM, identify the digital readout.
When DME distance comes in view, the DME groundspeed calculates the ETE to the station.
When FMS distance comes in view, the FMS calculates the ETE to the waypoint.
If the ETE is less than 1 hour, two digits (in minutes) gives its visual indication. If ETE is one hour or more (hours + minutes), the format becomes H + MM. The maximum ETE that is possible to give the visual indication is 8 hours and 31 minutes (8+31).
The groundspeed indication is digital. It comes in view at the bottom of the airspeed tape, and the DME or the FMS supplies the related data. When VOR is the primary NAV source, the groundspeed transmitted from the related DME channel is in view. When FMS is the primary NAV source, the groundspeed transmitted from the FMS is in view. The groundspeed digital readout shows the GS indication to the left and the value, from 0 thru 999 knots, to the right.
The used FMS mode comes in view in the lower left corner of the PFD display. The FMS mode indication can show:

- The APP: Approach
- The XTK: Offset
- The MSG: CDU Message
- The DR: Dead Reckoning
- The DGR: Degrade.

The XTK and APP indications use the same location above the TO waypoint Identifier. The MSG indication location is adjacent to the XTK/APP indications. The XTK/APP indications come in view in the reverse video for the first 5 seconds. The DR and DGR indications use the same location above the FMS distance field.

WIND WINDOW

The FMS can show the speed and direction of the wind on the PDF.
The indication of the wind direction occurs through an arrow included in a circle on the left side of the HSI compass card. Wind speed comes in view adjacent to the arrow, and the scale resolution is 1 knot. The wind value is not in view in zero wind conditions.

PFD WXR DISPLAY

The Weather Radar (WXR) display gives the indications that follow:

- The range display
- The WX mode indication
- The transmit indication
- The target alert indication
- The sector scan
- The lightning display (optional).

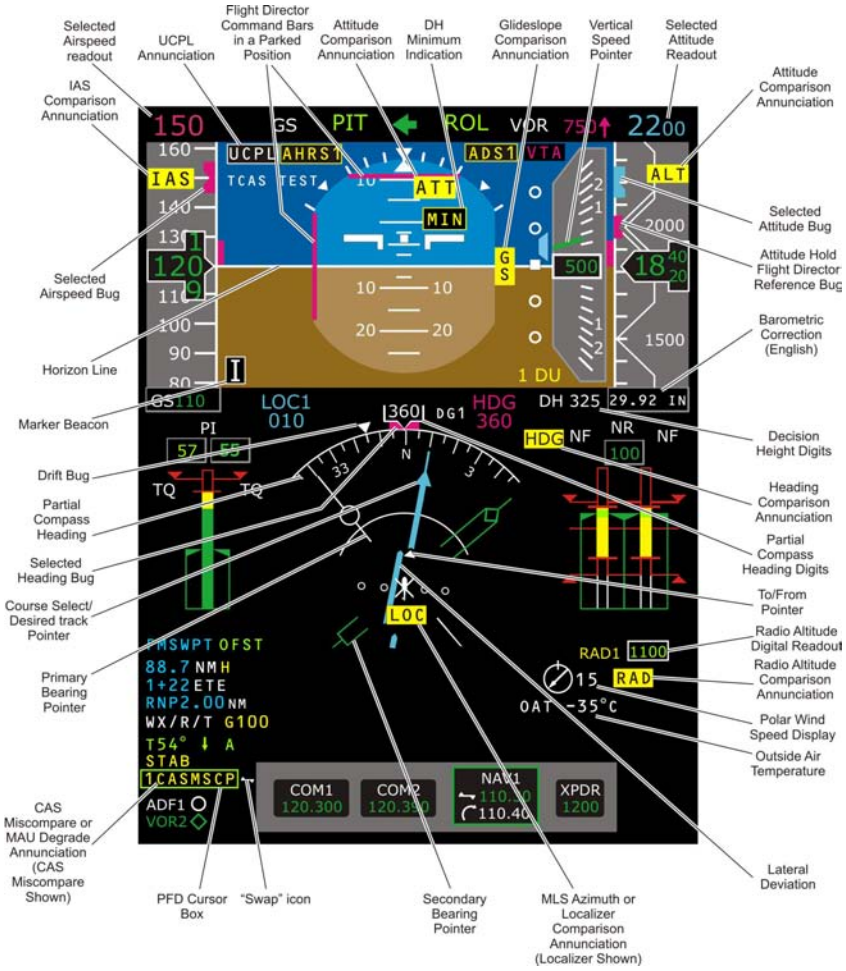
A half-range ring comes in view at the one-half of the arc-mode compass radius. A range indication comes in view at the end of the half-range ring. The range indication is equal to the one-half of the set WXR range.
The WXR indication comes in view adjacent to the heading display. This occurs when the radar signals are set to come in view on PFD or, when weight is on wheels and transmit mode is on. The WX mode annunciation flashes 1 second on and 0.5 seconds off if you do not have the conditions that follow:
The WX mode is WAIT
The transmit mode is on and the aircraft is on the ground.
With the selection one transmit modes (with the aircraft on the ground), an amber TX indication flashes 1 second on and 0.5 seconds off independently from the display selection.
When the target alert mode is set and a radar target is sensed, a TGT indication comes in view. The alert flashes for 1 second ON and 0.5 seconds OFF for 5 seconds when a target is first sensed.
The TGT indication stays in view until the alert condition continues. The indication is on the two full or the arc mode.
The PFD WX format colours are the same of the standard WX colour.
The PFD ARC mode only shows ±45 degrees around the applicable heading. Also the WX display limits have this range.
When the WX system operates in the sector-scan zone scan-mode, the left and right thick marks come in view at ±30 degrees on the scale arc.

NORMAL MODE ENGINE INDICATIONS

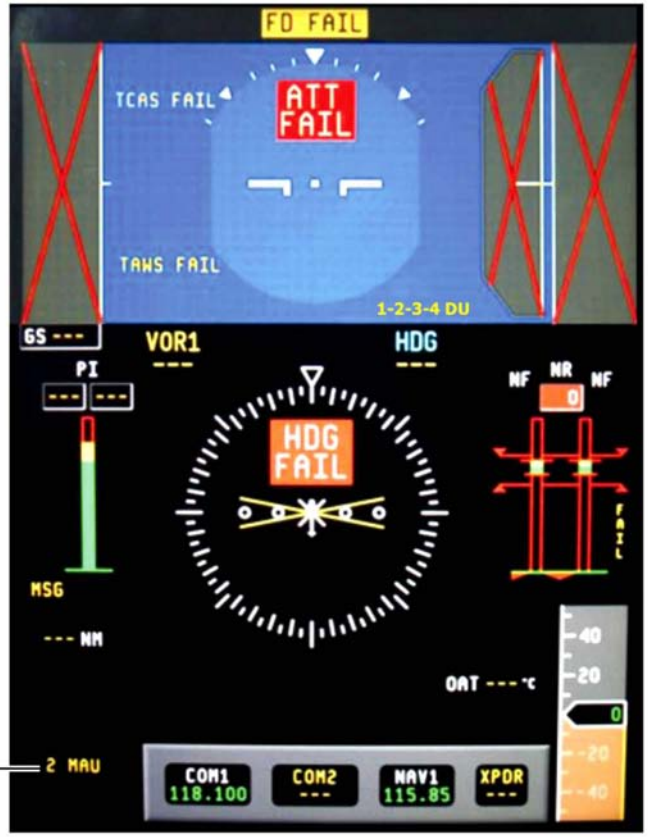
The primary engine annunciates are:

- The Power Index (PI)
- The triplex tach display.

The PI annunciator is at the left of the HSI. This annunciator shows the speed of the compressor-gas-generator (Ng), the inter-turbine temperature (ITT), or the torque (TQ) indications from 0 to 200 percent.
The triplex tachometer display is at the right of the HSI. It shows the fan speed (Nf) for each engine and the rotor speed (NR).



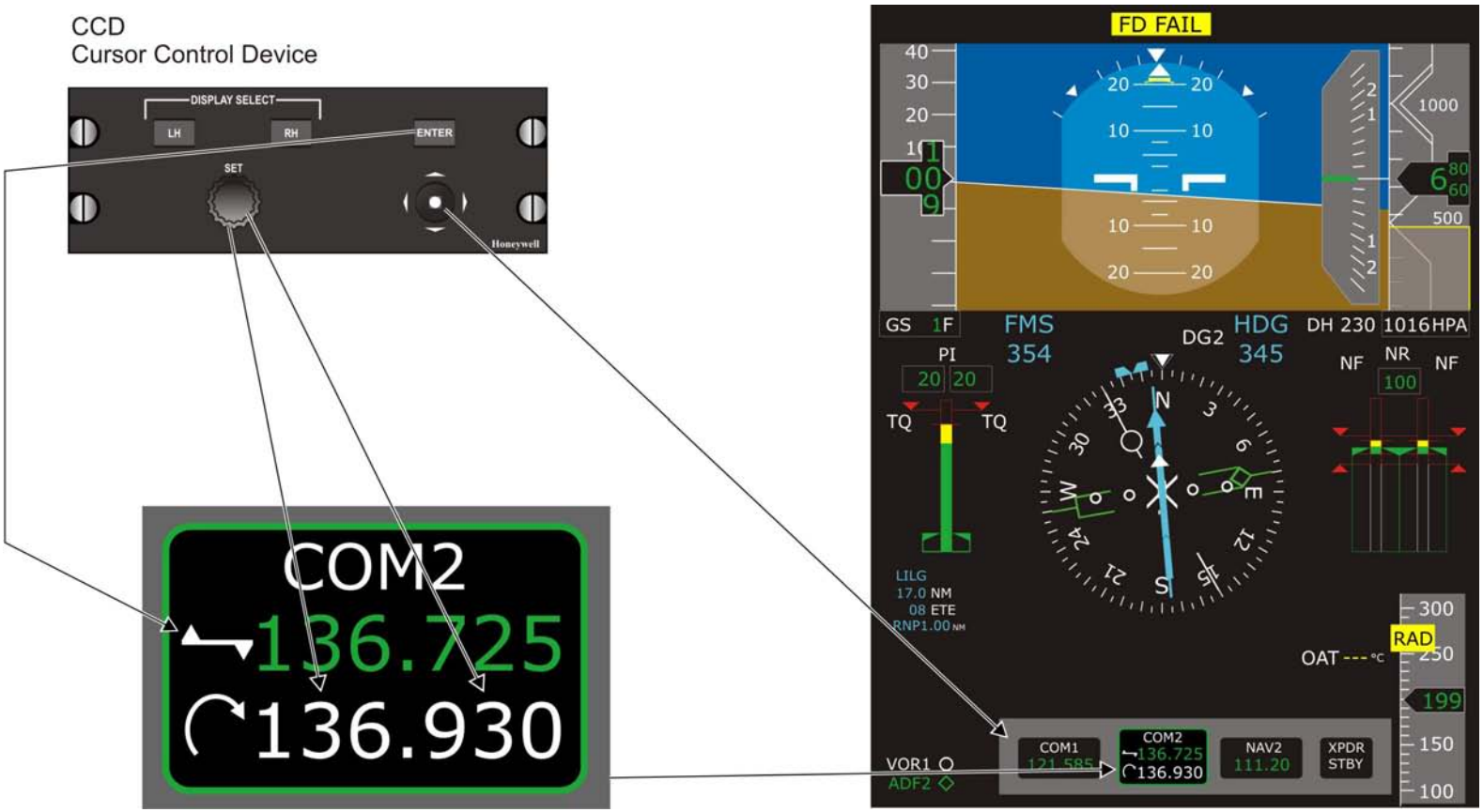
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PFD RADIO TUNING

The PFD tuned the radios of the helicopter. The PFD supplies tuned control of the COM1, COM2, the NAV, and the transponder codes. The PFD shows only the immediately tuned frequency for each radio.



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PFD RADIO TUNING

COMPOSITE MODE

The composite mode supplies the flight data available after one of the two Display Units (DUs) of the helicopter goes off. The composite format shows the data that follow:

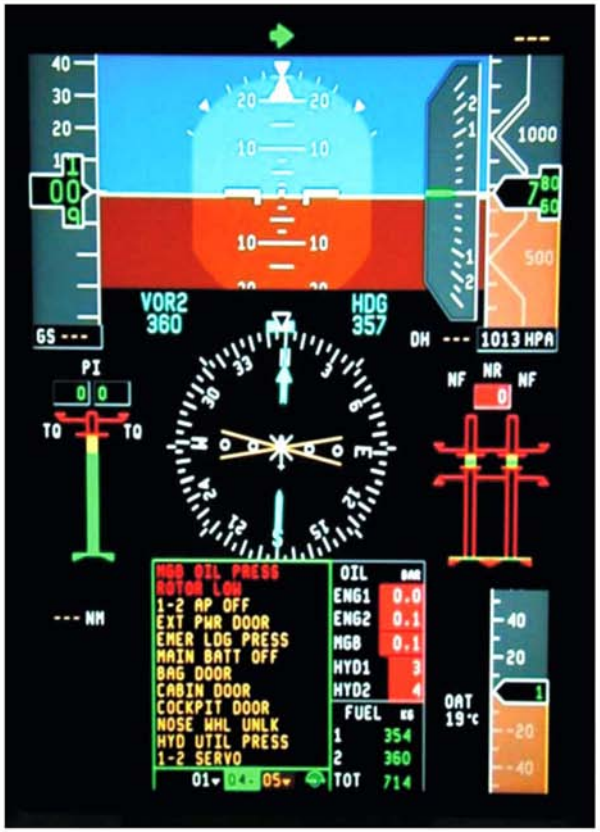
- The attitude display
- The slip/skip indicator
- The FD command bars
- The FD mode indications
- The AP mode indications
- The vertical deviation pointer and scale
- The radio altitude
- The DH
- The marker beacons
- The altimeter scale and digital displays
- The airspeed scale and digital displays
- The vertical speed scale and digital displays
- The heading
- The drift bug
- The lateral deviation scale
- The bearing ½
- The distance
- The FMS messages
- The preview mode
- The ITT, TQ, and Ng in the form of a PI
- The NR
- The NF
- The fuel quantity
- The source indications
- The source miscompare.

COMPOSITE MODE ENGINE INDICATIONS

The composite mode engine indications are:

- CAS Messages
- Engine Oil Pressure Indication
- Main Gearbox Oil Pressure Indication
- Fuel Quantity Indication
- Hydraulic Pressure.

The lower side of the PFD shows the CAS indications window. The window shows 12 CAS indications. The maximum number of characters that can come into view on the display is 18.
A digital indication of the oil pressure of Number 1 and the Number 2 engine comes in view in the bottom part of the PFD.
A digital indication of the oil pressure in the main gearbox comes in view in the bottom part of the PFD.
In the bottom part PFD there is a digital indication of the quantity of the fuel in the Number 1 and Number 2 tanks. The same part of the PFD also shows a digital indication of the Number 1 and Number 2 hydraulic system-pressure.





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COMPOSITE MODE - COMPOSITE MODE ENGINE INDICATIONS

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CHAPTER
32
LANDING GEAR

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SECTION 00 - LANDING GEAR

LANDING GEAR – GENERAL

The landing gear is a “fore and aft” retractable tricycle type.
The landing gear:

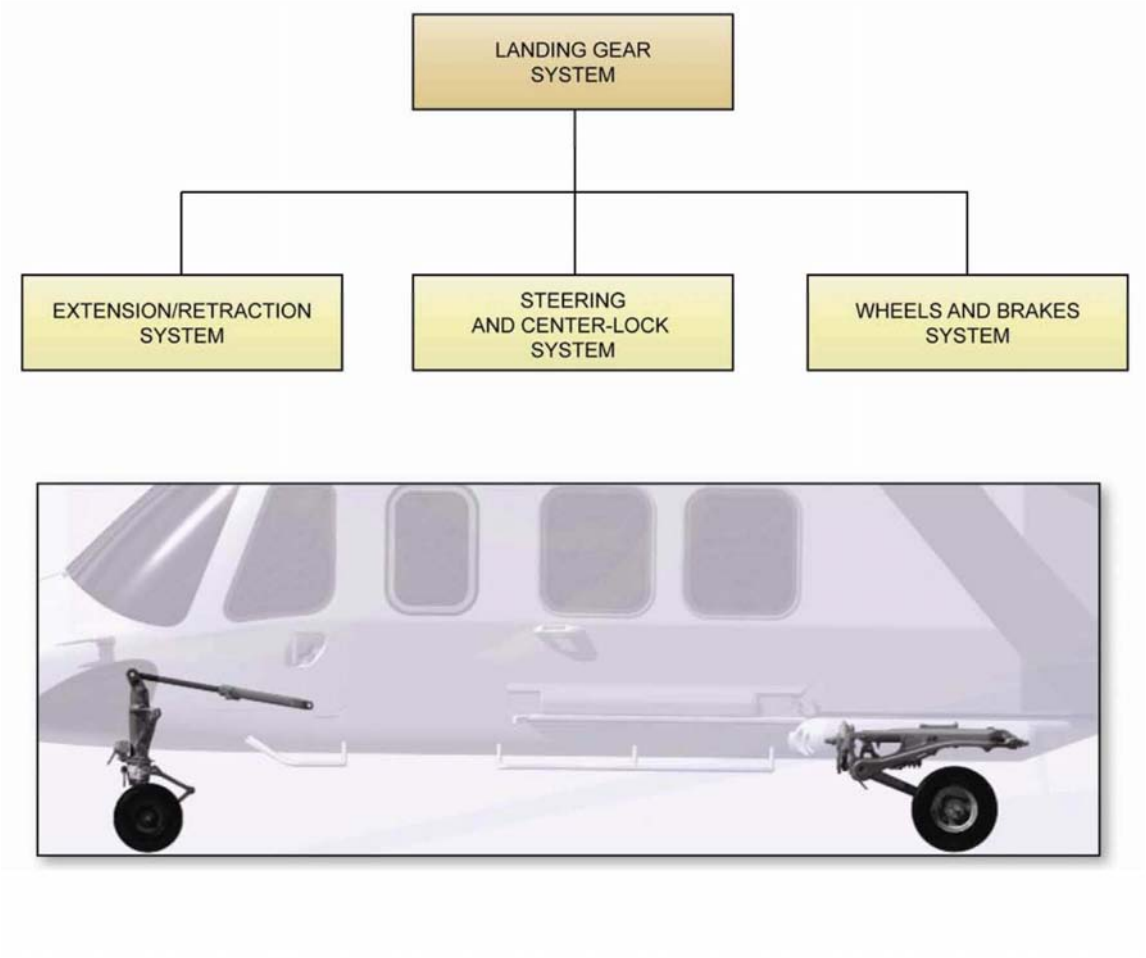
- supports the helicopter on the ground giving the correct ground clearance
- gives shock absorption and rebound control during landing, taxiing and manoeuvres on the ground
- decelerates and stops the helicopter after landing and during taxiing
- allows to park the helicopter

The landing gear includes:

- the Main Landing Gear (MLG) located on the left and on the right side of the fuselage under the sponsons
- the Nose Landing Gear (NLG) located under the cockpit
- the extension / retraction system of the MLG and NLG
- the braking system
- the automatic centering and lock system (or steering system) located on the NLG

The landing gear is operated by the pilot to perform:

- normal retraction and extension by means of the Landing Gear Control Lever (LGCL)
- emergency extension by means of the EMER DOWN pushbutton



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LANDING GEAR – GENERAL LAYOUT

SECTION 10 - MAIN GEAR AND DOORS

DESCRIPTION

The Main Landing Gear (MLG) comprises the left and right landing gears located symmetrically on either side of the fuselage under the respective sponsons. Each landing gear is constituted by:

- Retractable lever suspension mechanism with an oleo-pneumatic damping system
- Single tubeless wheel

The wheel hub incorporates a safety relief plug to permit release of overpressure and two fusible plugs which release tyre pressure in case of overheating.
The main landing gear is maintained in UP position by hydraulic power and in DOWN position by mechanical locks inside the actuators.
Each main landing gear includes:

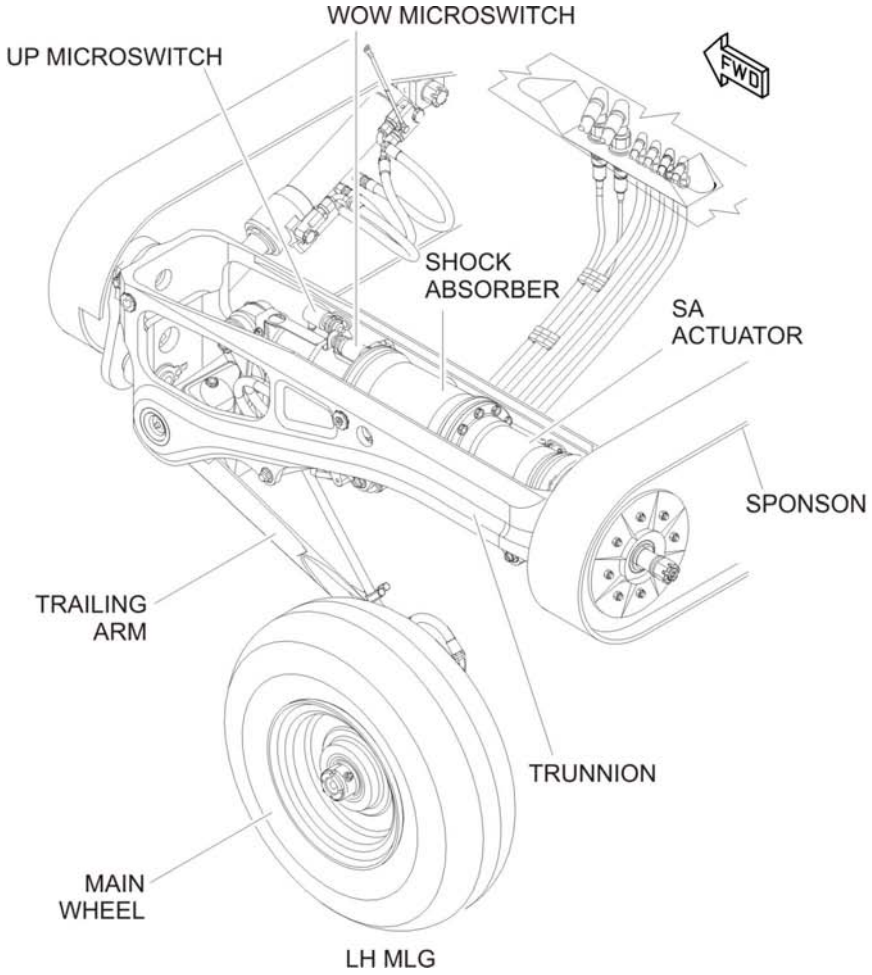
- Extension/retraction (RA) actuator
- Shortening actuator (SA)
- Shock absorber
- Weight-On-Wheel (WOW) microswitch

The extension/retraction (RA) actuator and the shortening actuator (SA) are part of the extension/retraction system described later.
The MLG can operate in three different ways: retraction nomal extension and emergency.

MAIN LANDING GEAR – MAIN COMPONENTS

SHOCK ABSORBER

The shock absorber is a gas/oil damper unit composed by a piston rod sliding inside a cylinder. One end of the shock absorber is connected to the shortening actuator, the other end is connected to the trailing arm which is a structural element that maintains the MLG aligned longitudinally with the helicopter axis.
The piston rod includes a separator piston that prevents mixing of the gas with the oil and a variable orifice valve unit that controls the landing gear damping and rebound action.
The variable orifice valve unit is composed by a spring loaded piston which, under the oil pressure generated during landing, opens calibrated holes through which the oil moves at a controlled flow rate.
The shortening actuator extends for MLG retraction and retracts for MLG extension.
The shock absorber and the shortening actuator (SA) are joined together to build up the shock absorber assembly.



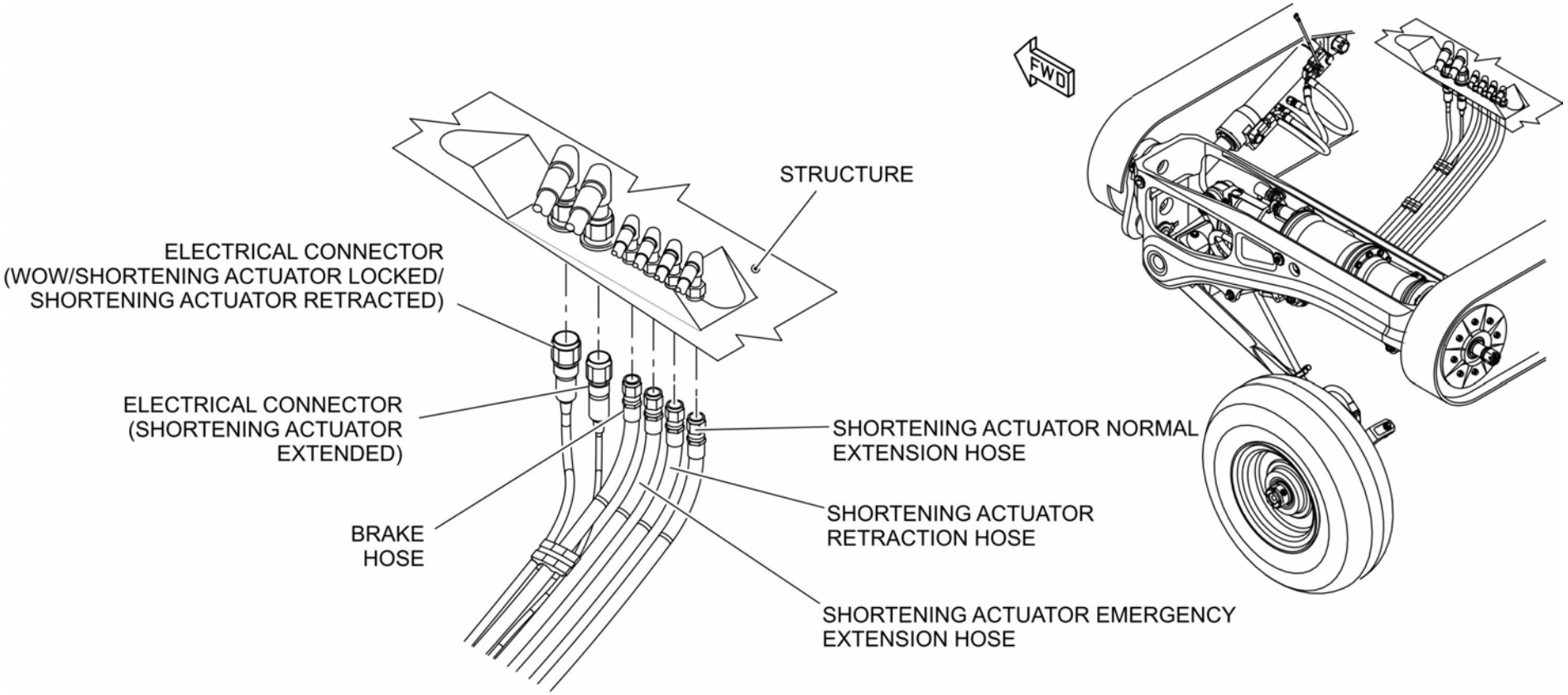


RH MLG

NOTE: LH and RH are arranged simmetrically

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MAIN LANDING GEAR



NOTE: LH and RH are arranged simmetrically

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MAIN LANDING GEAR - ASSEMBLY

LEADING PARTICULARS

- Hydraulic fluid..... MIL-PRF-5606
- MLG assembly (including wheel and brake) weight..... 61.8 Kg (136.2 lb)
- Shock absorber weight..... 12.8 Kg (28.2 lb)
- Trailing arm weight..... 11.6 Kg (25.6 lb)
- Trunnion weight..... 13.5 Kg (29.8 lb)
- Main landing gear shock absorber oil volume ..... 0.75 l (0,68 US quart)

SECTION 20 - NOSE GEAR AND DOORS

DESCRIPTION

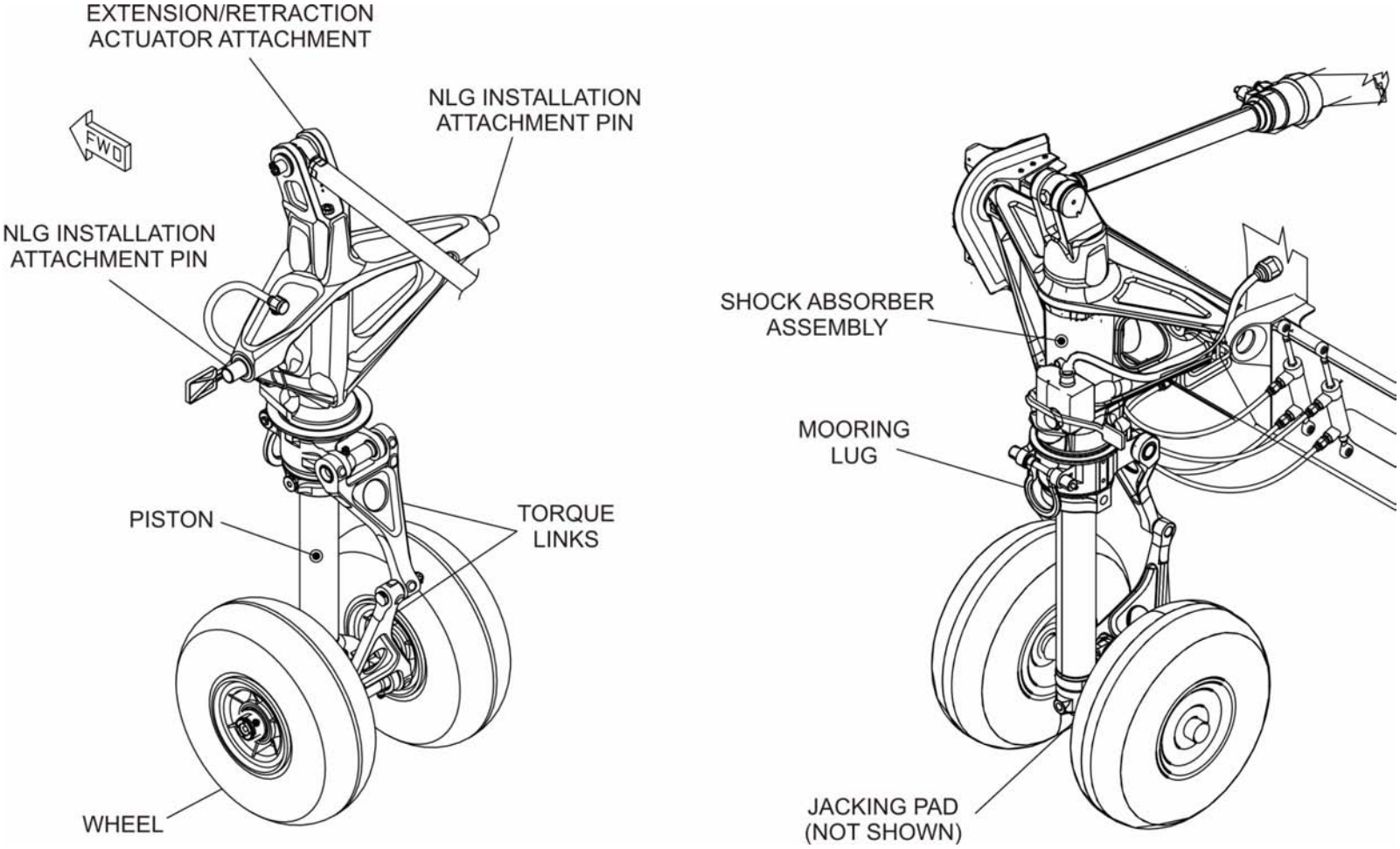
The Nose Landing Gear (NLG) is a telescopic, retractable type with oleo-pneumatic damping system, twin wheels.
The Nose Landing Gear is composed by the following main components located in the nose structure:

- shock absorber assembly
- torque links

On the Nose Landing Gear are also installed the extension/retraction (R/A) actuator, the centering assembly, the center lock assembly and the wheels.
The shock absorber assembly is a nitrogen/oil damper unit composed by a steel sliding piston rod inside an aluminium cylinder.
At the bottom of the sliding piston is installed the wheels axle. The head of the bolt fixing the axle to the sliding piston rod provides the point to individually lift the nose landing gear.
The cylinder is shaped such to provide the pivoting lateral attachments of the assembly.
The assembly is installed with two pins.
The sliding piston rod includes a separator piston that prevents mixing of the nitrogen with the hydraulic oil and a variable orifice valve unit that controls the landing gear damping and rebound action installed on the top.
The variable orifice valve unit is composed by a spring loaded piston which, under the fluid displacement forces during landing, opens calibrated holes through which the fluid moves at a controlled rate.
On the lower end of the sliding piston rod is fitted the lower torque link.
The shock absorber cylinder provides on the top the attachment of the extension/retraction actuator.
The lower end of the shock absorber cylinder provides the installation of a rotating collar, the centering assembly (inside the shock absorber), the center lock assembly and the upper torque link.

The torque links are two aluminium identical arms linked together which connect the rotating collar with the wheels and allow the rotation movement between the wheels and shock absorber cylinder during towing/taxing operations.
The nose landing gear is interfaced with:

- Forward fuselage
- Extension/Retraction system
- Wheels and brakes system
- Steering system
- Modular Avionic Unit



ICN-39-A-322000-G-A0126-03324-A-01-1

OPERATION

NLG SHOCK ABSORBER

- With no load on the unit (helicopter off ground) the nitrogen pre-charge pressure forces the separator piston to hold it against the oil.
- On landing the shock absorber piston moves up.
- The movement of the piston reduces the fluid volume in the cylinder by displacing it through the port of the variable orifice unit providing a first stage damping.
- When fluid pressure increases, the displacement forces compress the spring-loaded piston of the variable orifice unit which opens holes through which the fluid moves at a controlled rate to the separator piston.
- Any time the fluid is displaced from the shock absorber cylinder to the shock absorber piston through the variable orifice unit, the separator piston compresses the nitrogen gas storing the landing impact energy.
- Once the oil displacement is finished, the compressed gas starts to expand to force the fluid back to the shock absorber cylinder through the port of the variable orifice unit since the holes controlled by the spring loaded piston are now closed.

LEADING PARTICULARS

- Hydraulic fluid..... MIL-PRF-5606
- NLG assembly (including wheels) weight..... 34.1 Kg (75.2 lb)
- Nose landing gear shock absorber oil volume..... 1.78 l (1.61 US quarts)
- Nose landing gear shock absorber nitrogen charge at shipping condition..... 3.5 bar (51 psi)
- Nose landing gear nitrogen charge ..... 8 bar (116 psi) (off ground at 20° C)
- Normal rate of descent..... 2.3 m/s (452 ft/min)
- Hard landing rate of descent..... 3.05 m/s (600 ft/min)

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SECTION 30 - EXTENSION AND RETRACTION

DESCRIPTION

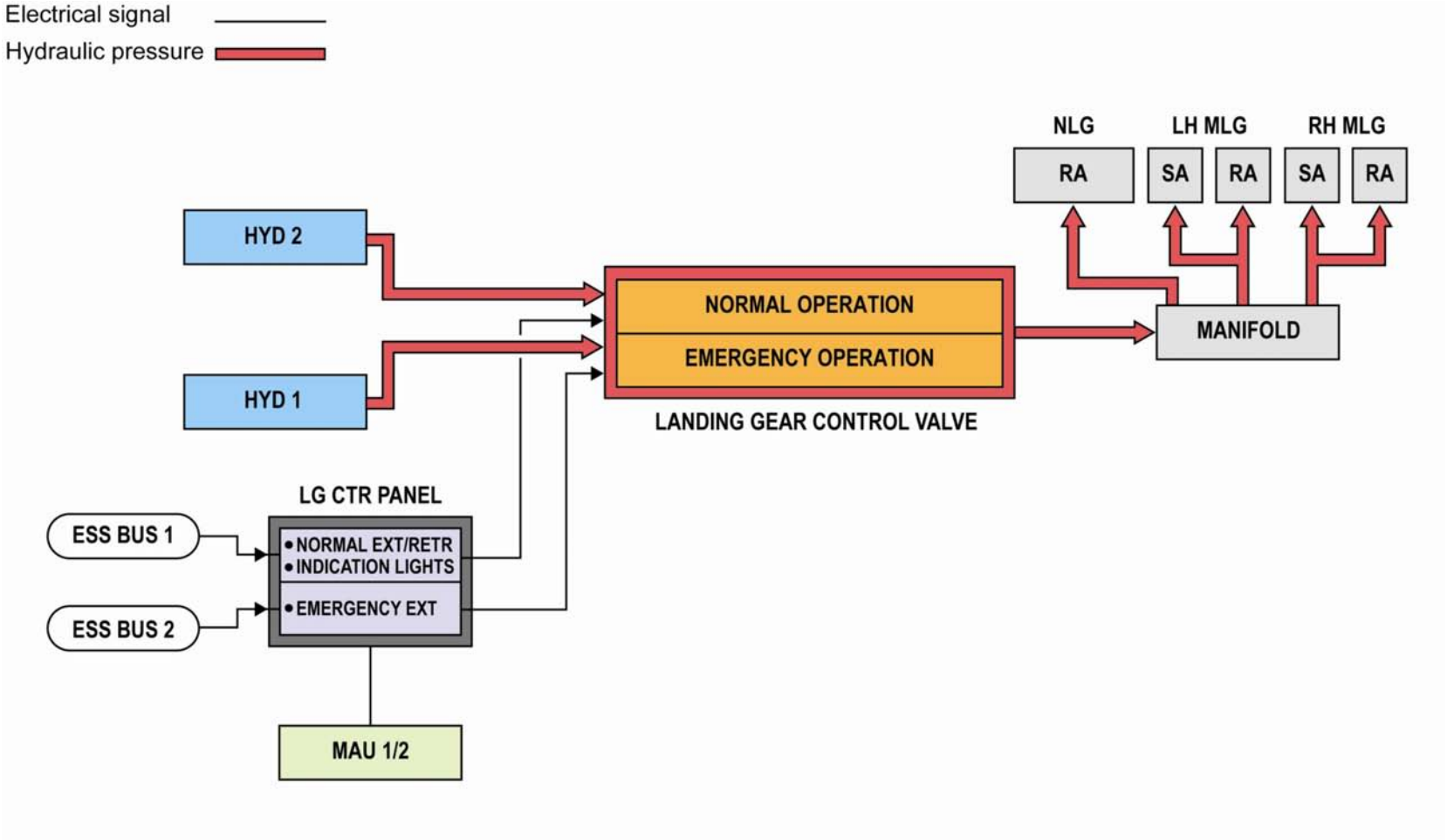
The extension/retraction system allows the extension and retraction of the main and nose landing gears under inputs controlled by the pilot.
The extension/retraction system includes:

- One Landing Gear Control Valve (LGCV)
- Two main and one nose landing gear extension /retraction actuators (RA)
- Two main landing gear shortening actuators (SA)
- Landing Gear Control Panel (LGCP)
- Landing Gear Control Lever (LGCL)
- Microswitches

The extension/retraction system is hydraulically powered by

- Hydraulic system No. 2 for normal extension/retraction
- Hydraulic system No. 1 for emergency extension.

The LGCV acts on a manifold, which interface and operates the hydraulic actuators of the nose and main landing gears.
The extension/retraction system is electrically operated through the Landing Gear Control Panel (LGCP) located in the cockpit and through a logical sequence controlled by electrical microswitches.
The landing gear is mechanically locked when fully extended and hydraulic pressure is received from actuators.
The landing gear is held in the retracted position by hydraulic pressure; in case of loss of hydraulic power the landing gear legs come out of their bays by effect of their own weight.



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EXTENSION AND RETRACTION SCHEMATIC (GENERAL)

COMPONENTS FUNCTION

LANDING GEAR CONTROL PANEL (LGCP) AND LANDING GEAR CONTROL LEVER (LGCL)

The Landing Gear Control Panel is located on the forward RH side of the central console. The LGCP includes the controls and indications for:

- LG normal extension/retraction (LGCL and LG status annunciators)
- LG emergency extension (EMER DOWN guarded push-button switch)
- Nose wheel centerlock (pushbutton switch)
- Park brake handle

The LGCL has two positions: UP and DOWN to control the landing gear control valve, LG normal retraction and extension. The control lever must be pulled before operating it. The LG status annunciators show the LG position with the following logic:

- 3 green lights for DOWN and LOCKED position with LGCL at DOWN
- All lights extinguished for LG in UP position with LGCL at UP
- 3 amber lights for any other condition (LG unsafe)
- Green/amber light to display the NLG center lock status

The LGCL locked in the DOWN position when the helicopter is on the ground (WOW microswitch).

EMERGENCY DOWN (EMER DOWN) PUSHBUTTON SWITCH

An EMER DOWN guarded switch control the emergency section of the Landing Gear Control Valve. The switch is able to move down the landing gears in case of failure in the normal hydraulic circuit of the control lever. In this case the emergency switch will be used to electrically actuate the emergency spool of the Landing Gear Control Valve.

LANDING GEAR CONTROL VALVE (LGCV)

The Landing Gear Control Valve (LGCV) controls the hydraulic fluid flow towards to the landing gear actuators. The LGCV comprises two solenoid valves identified as UP and DOWN solenoid valves.
When the pilot moves the Landing Gear Control Lever to the UP or DOWN positions, electrical power is supplied to the relevant solenoid valve that, in turn, supply hydraulic power to the UP or DOWN circuit respectively.
The landing-gear control valve is connected to the hydraulic operated components of the NLG and MLG via a manifold.

MLG EXTENSION / RETRACTION ACTUATOR

The extension/retraction actuator is an hydraulic actuator which, together with the shortening actuator, extends and retracts the MLG according to the pilot selection. The extension/retraction actuator is attached to a structural element of the MLG (the trunnion).

MLG SHORTENING ACTUATOR (SA)

The main landing gear Shortening Actuator (SA) increases its length for the retraction of the landing gear and decreases its length for the extension.

NLG EXTENSION / RETRACTION ACTUATOR (RA)

The NLG extension/retraction actuator is an hydraulic actuator which extends and retracts the NLG according to pilot selections on the Landing Gear Control Lever (LGCL).

POSITION MICROSWITCHES

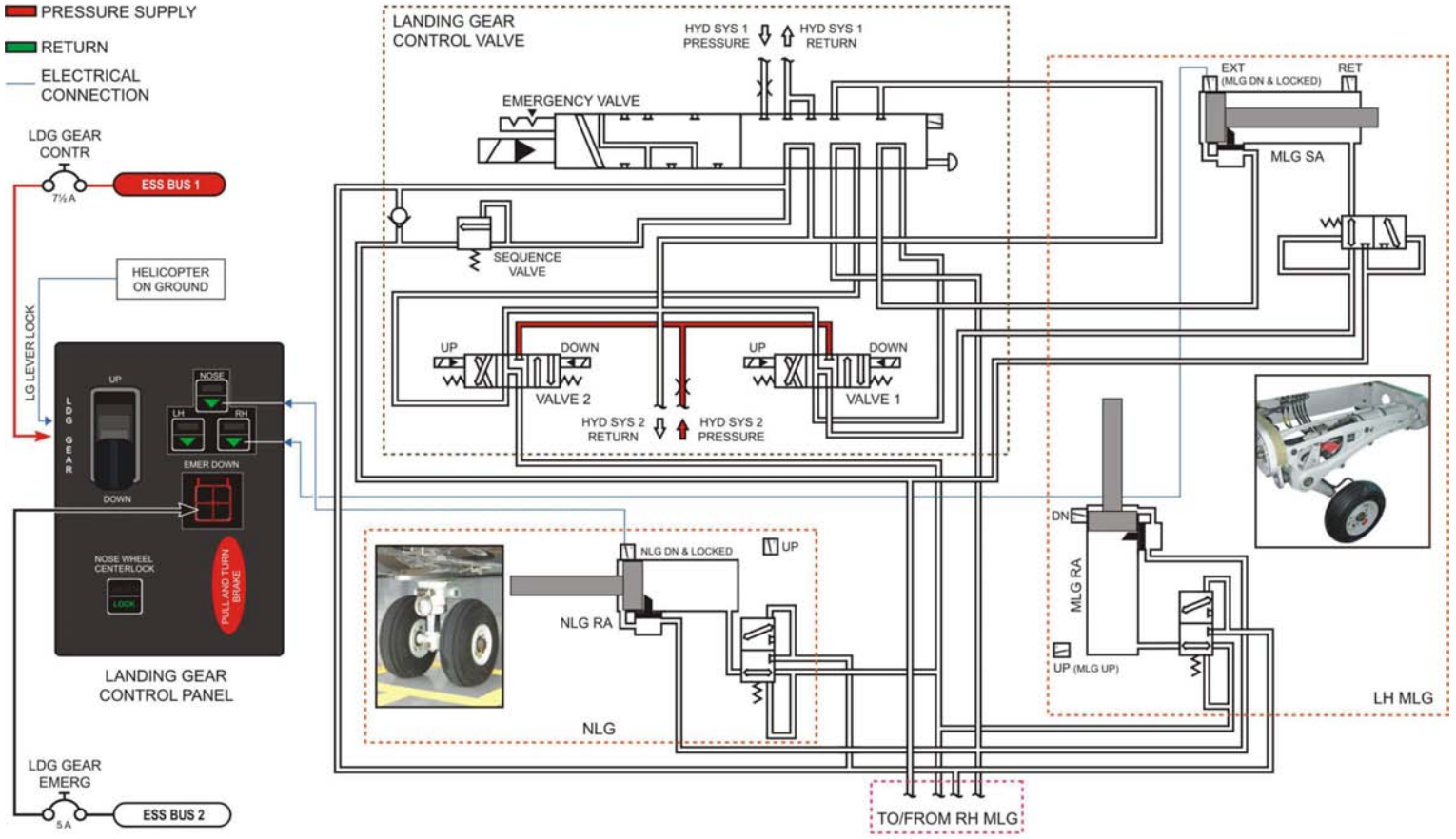
Position microswitches monitor the extended, retracted and locked conditions of each actuator of the MLG and NLG to control the normal extension/retraction sequences and provide LG status indications on the LGCP.

- MLG and NLG retracted microswitches. These microswitches are operated when the landing gears are fully stowed in the aircraft bays (following UP selection)
- MLG and NLG RA extended and locked microswitches. These microswitches are operated when the internal mechanical lock is engaged (following DOWN selection)
- MLG SA actuator extended microswitches. These microswitches are operated by the shock absorber when the SA actuator is fully extended (following UP selection)
- MLG SA actuator retracted microswitches. These microswitches are operated when the SA actuator is fully retracted (following DOWN selection)
- MLG SA actuator locked microswitches. These microswitches are located inside the actuator and are operated by an internal device when the SA actuator is fully retracted (following DOWN selection)
- Weight On Wheels (WOW) microswitches. These microswitches are located on the shock absorber of each leg of the MLG and provide information of “aircraft on the ground” or “aircraft in flight” status to various helicopter systems.

LANDING GEAR – PRINCIPLE OF OPERATION

ON GROUND

When the helicopter is on the ground the Landing Gear Control Lever (LGCL) is locked DOWN by a pin controlled by the Weight-On-Wheel (WOW) microswitches.



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ON GROUND – LANDING GEAR DOWN AND LOCKED

NORMAL OPERATIONS

The pilot operates the normal retraction and extension of the landing gear acting on the LGCL which controls the Landing Gear Control Valve (LGCV).
The following figures show the extension and retraction operation sequenced in phases.

NOTE A

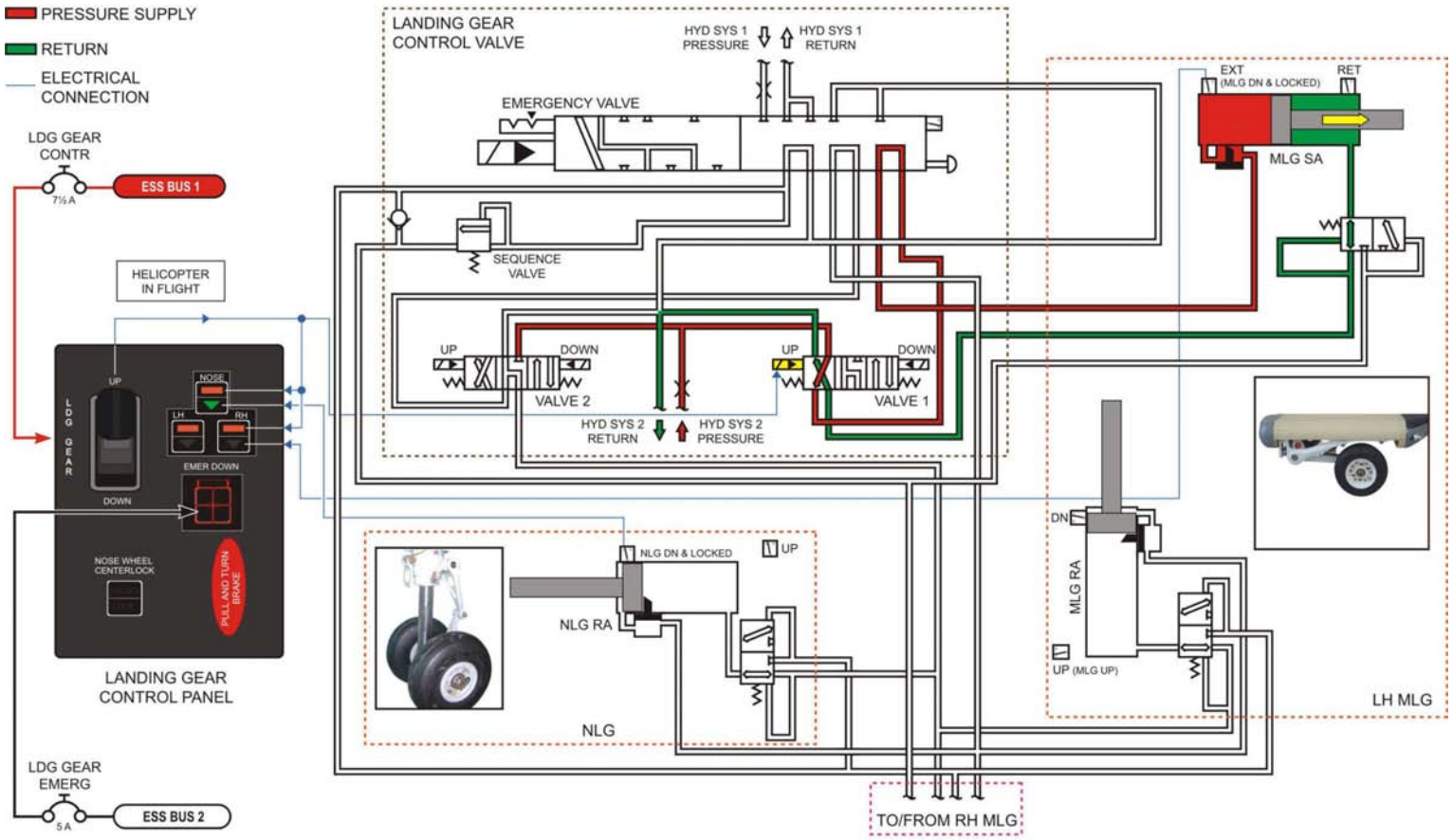
Only the LH MLG is shown in the diagram; RH MLG operates in the same way.
Actuator position microswitches control all the extension/retraction phases.

NORMAL RETRACTION

Normal retraction occurs in two phases:

PHASE 1 – MLG SHORTENING

When the helicopter lifts-off, the WOW release the LGCL locking pin and the pilot can move the LGCL to UP position. The NOSE, LH and RH amber lights on the LGCP are illuminated because the LGCL position is UP while the LG is not.
The retraction procedure starts only if the NLG is locked in the center position (LOCK illuminated green in the NOSE WHEEL CENTERLOCK pushbutton).
The VALVE 1 UP solenoid is energized and pressure from the hydraulic system no.2 is delivered to the MLG Shortening Actuator (MLG SA) to unlock from DN position and to shorten the MLG. As the SA piston unlocks, the position microswitch extinguishes the relevant green triangular annunciator on the LGCP.

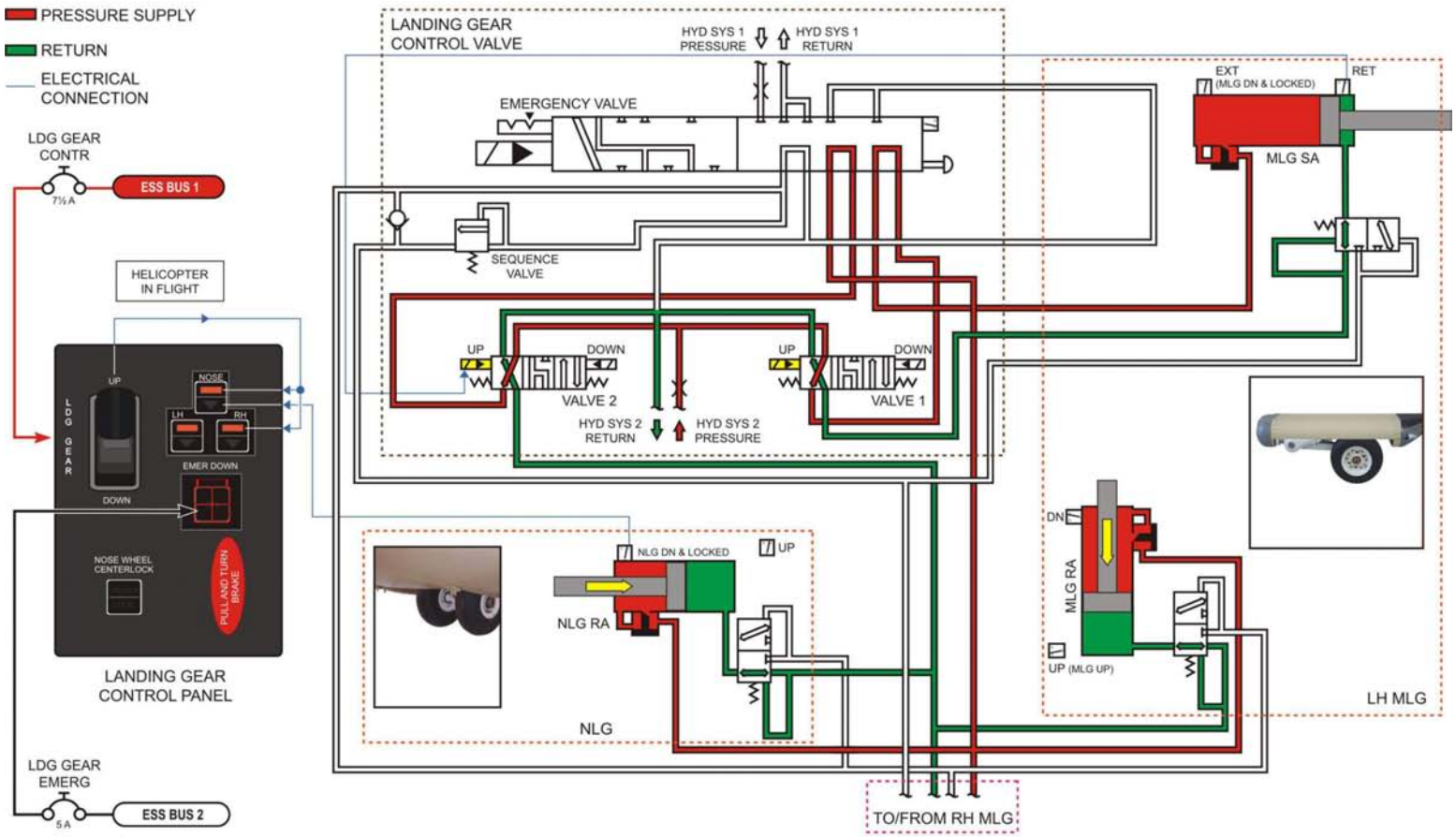


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NORMAL RETRACTION PHASE 1 – MLG SHORTENING

PHASE 2 – NLG and MLG RETRACTION

When both SA are fully retracted, the VALVE 2 UP solenoid is also energized and pressure is delivered to NLG and MLG Retraction Actuators (RA).
As the NLG RA piston unlocks, the position microswitch extinguishes the NOSE green triangular annunciator on LGCP.

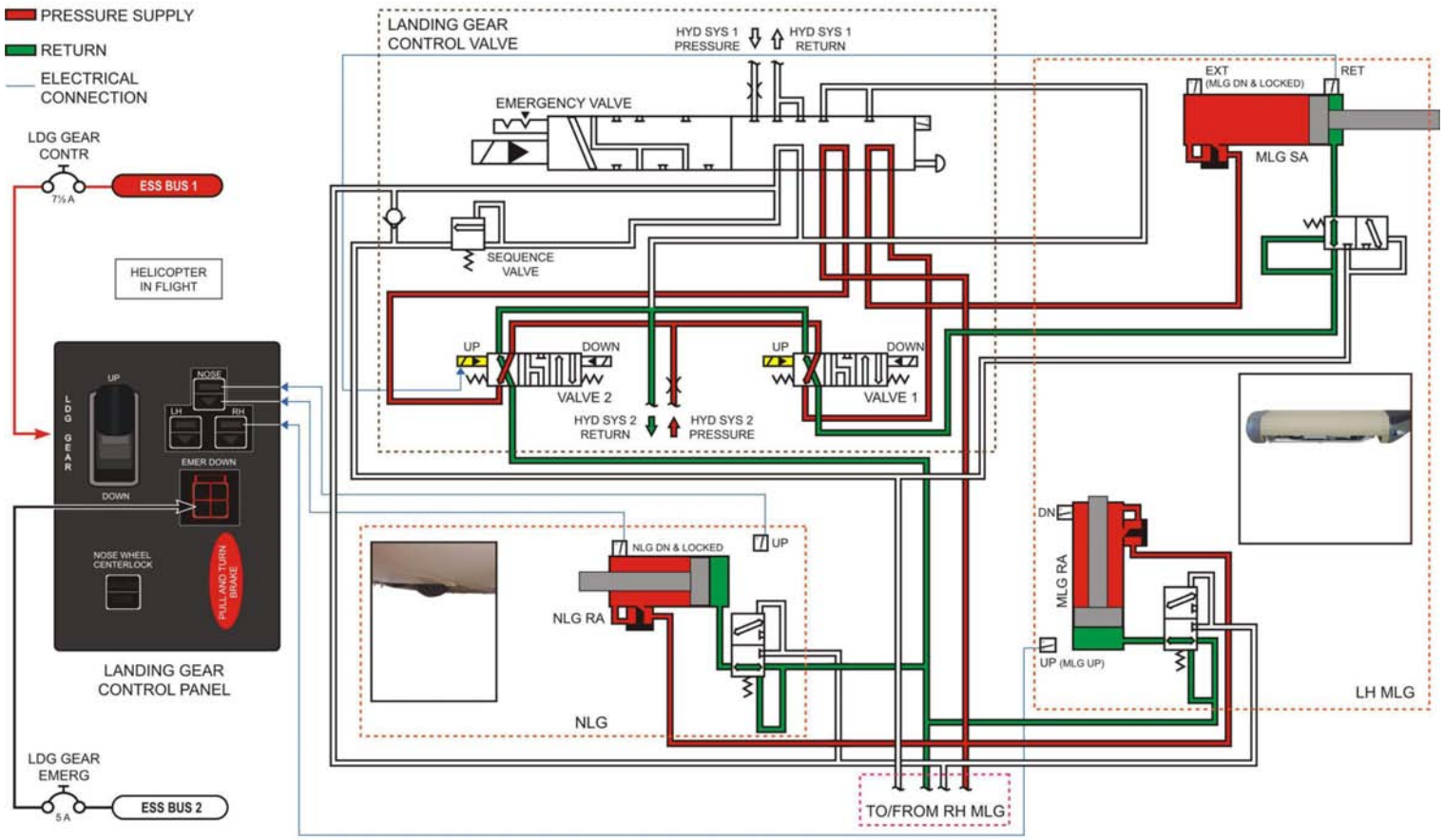


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NORMAL RETRACTION PHASE 2 – NLG AND MLG RETRACTION

LANDING GEAR RETRACTED

When the NLG RA and MLG RA are fully retracted and each landing gear contacts the relevant UP microswitch on the airframe, the corresponding NOSE, LH or RH amber light on the LGCP extinguishes.
Both VALVE 1 and VALVE 2 UP solenoids remain energized to keep the pressure in all actuators as long as the LG is retracted.



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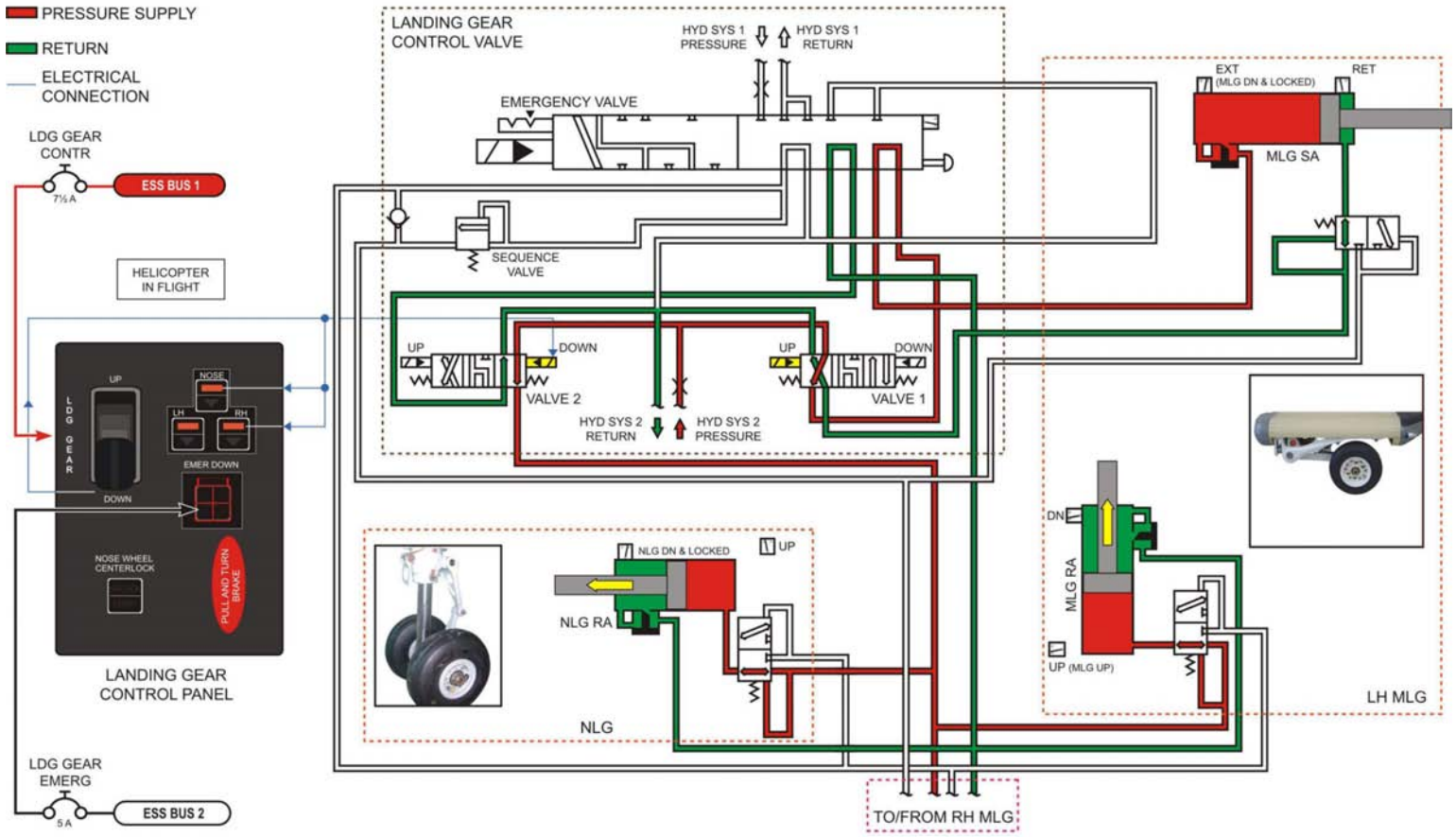
LANDING GEAR RETRACTED

NORMAL EXTENSION

Normal extension occurs in two phases:

PHASE 1 – NLG and MLG EXTENDING

When the pilot moves the LGCL to DOWN position, the NOSE, LH and RH amber lights on the LGCP illuminate because the LGCL is DOWN while the LG is not DOWN and LOCKED.
With LGCL at DOWN, the VALVE 2 DOWN solenoid energizes and pressure from the hydraulic system no.2 is delivered to the NLG and MLG retraction actuators to retract.

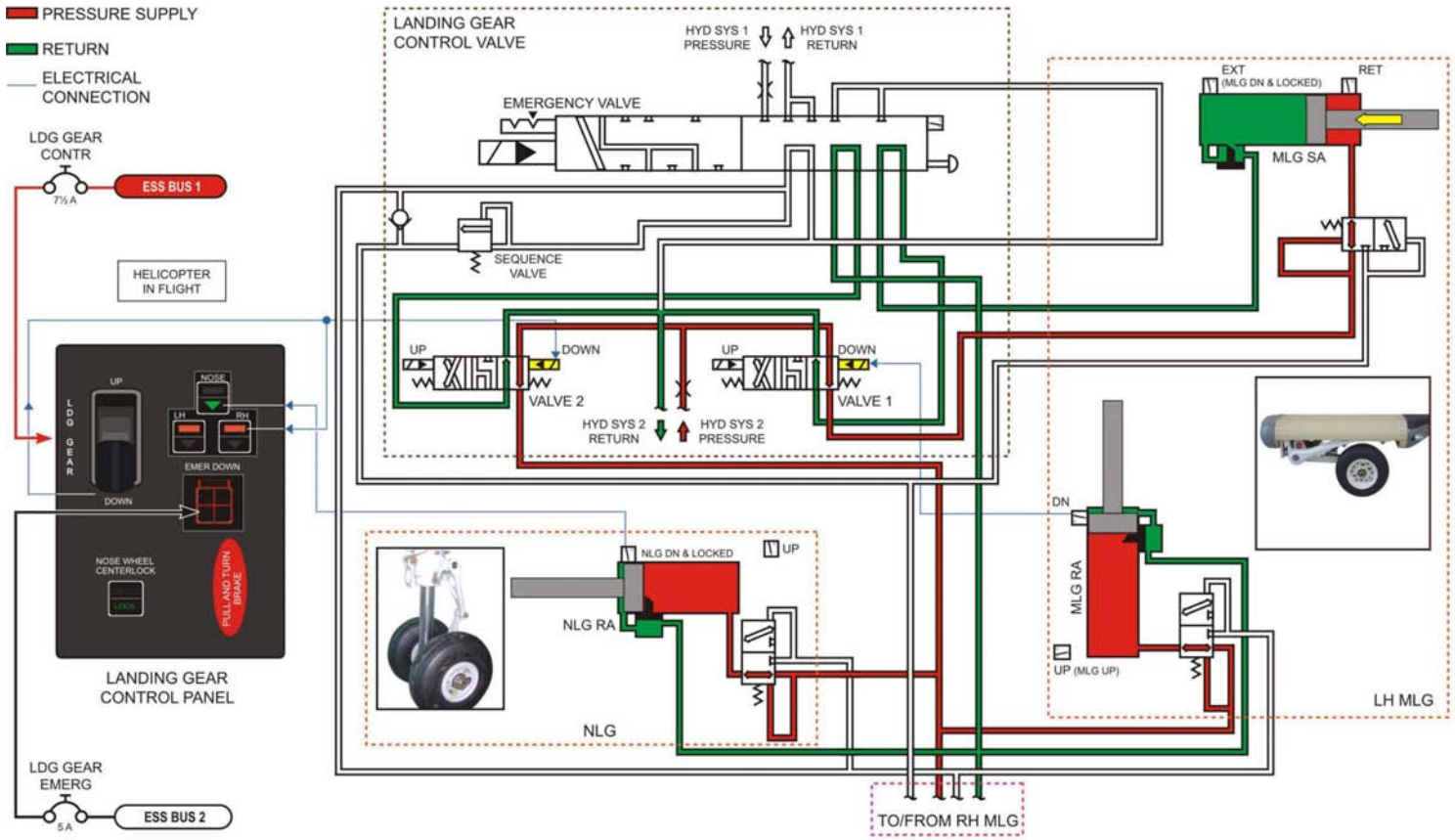


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NORMAL EXTENSION PHASE 1 – NLG AND MLG EXTENDING

PHASE 2 - NLG DN and LOCKED, MLG ELONGATING

When the NLG RA is fully extended and locked, the position microswitch illuminates the NOSE green triangular annunciator (NLG DN & LOCKED) and extinguishes the corresponding amber light.
When both MLG RA are fully extended (DN) the position microswitches energize, the VALVE 1 DOWN solenoid and pressure is delivered to the MLG SA to elongate (extend) the MLG.

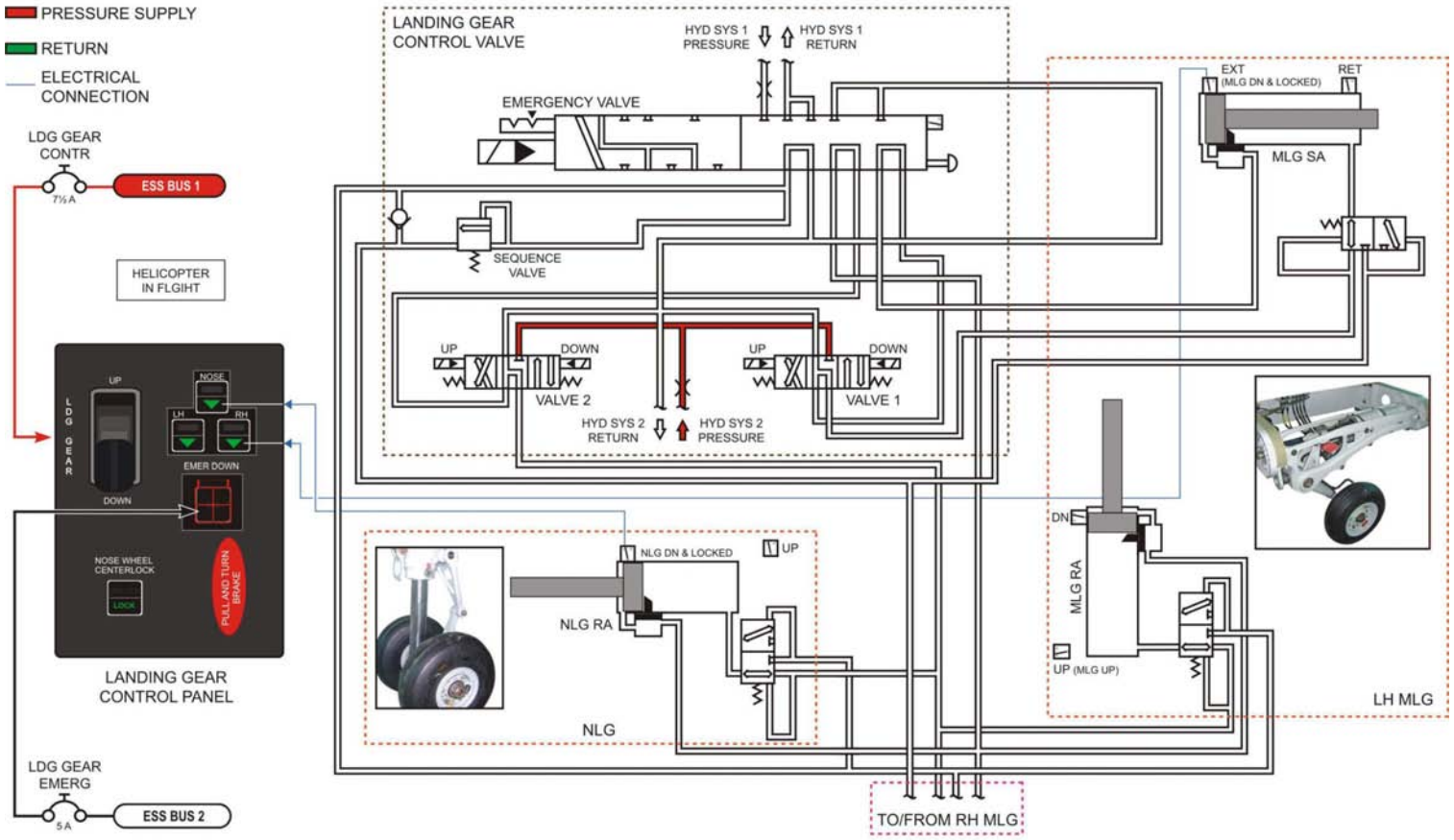


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NORMAL EXTENSION PHASE 2 – NLG DN AND LOCKED, MLG ELONGATION

LANDING GEAR DOWN AND LOCKED

When both SA are fully extended and locked, the VALVE 1 DOWN and VALVE 2 DOWN solenoids are de-energized so that the system is no more pressurized.
The three green lights NOSE, LH and RH are all illuminated and the three amber lights are all extinguished.



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LANDING GEAR DOWN AND LOCKED

EMERGENCY EXTENSION

Emergency extension permits lowering of the LG in case the normal operation is ineffective.

Emergency extension of the landing gear is controlled using the EMER DOWN pushbutton which acts on the emergency valve of the LGCV and utilities:

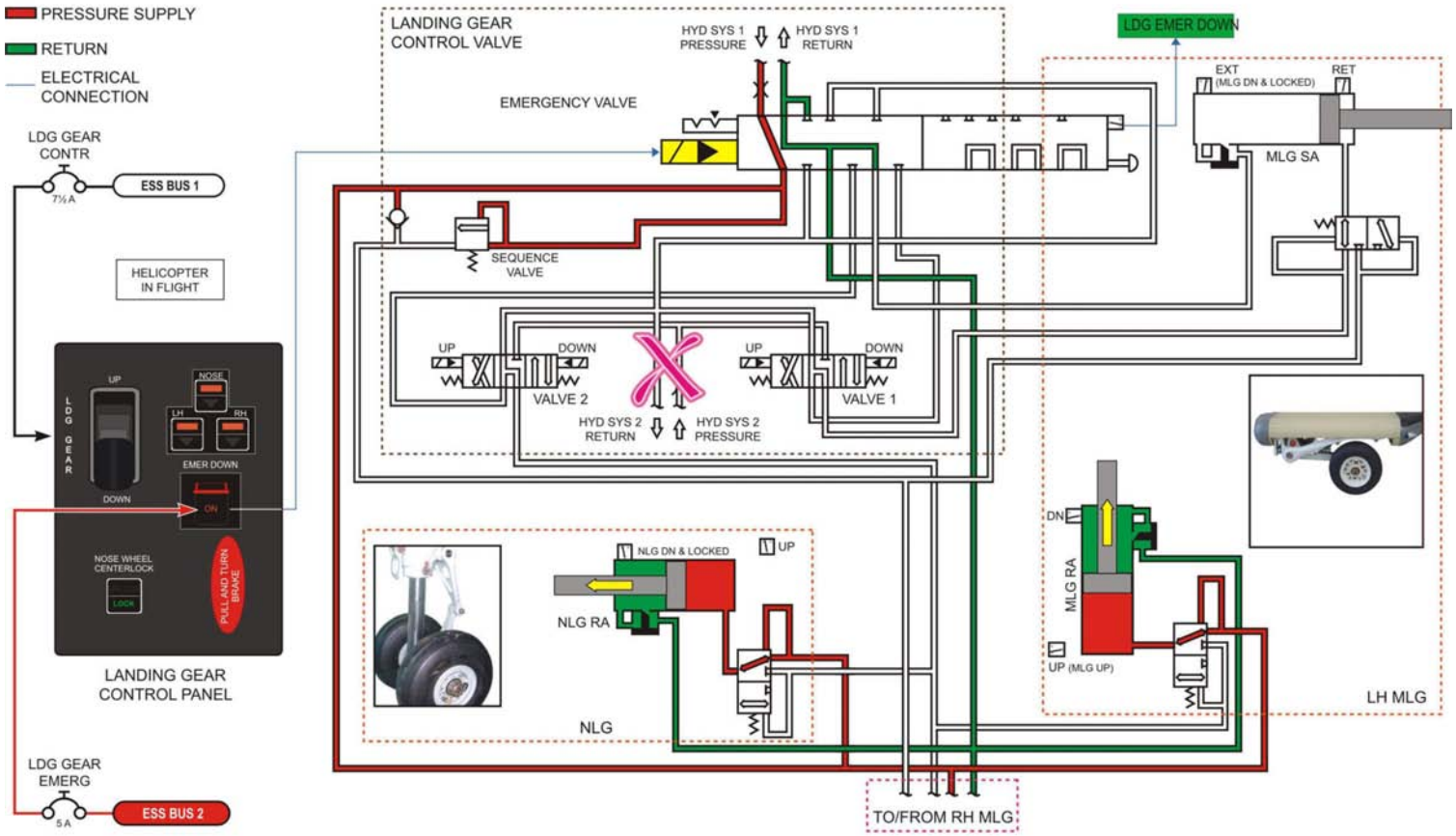
- hydraulic power from system no.1
- electrical power from ESS BUS 2

The position of the Landing Gear Control Lever (UP or DOWN) is not relevant for emergency extension.

Emergency extension occurs in two phases.

PHASE 1

If, after selecting the LGCL to DOWN, any LG position indicators remain blank or amber and normal landing gear extension is confirmed to be ineffective, the emergency extension is to be carried out by lifting the guard and pressing the EMER DOWN pushbutton switch which illuminates (amber ON legend).
The emergency valve solenoid is energized and pressure from hydraulic system no.1 is delivered to the MLG SA to elongate (extend) the MLG and the NLG RA.
The NOSE, LH and RH amber lights illuminate.
The emergency valve position microswitch provides the advisory LDG EMER DOWN on the CAS window of the MFD.

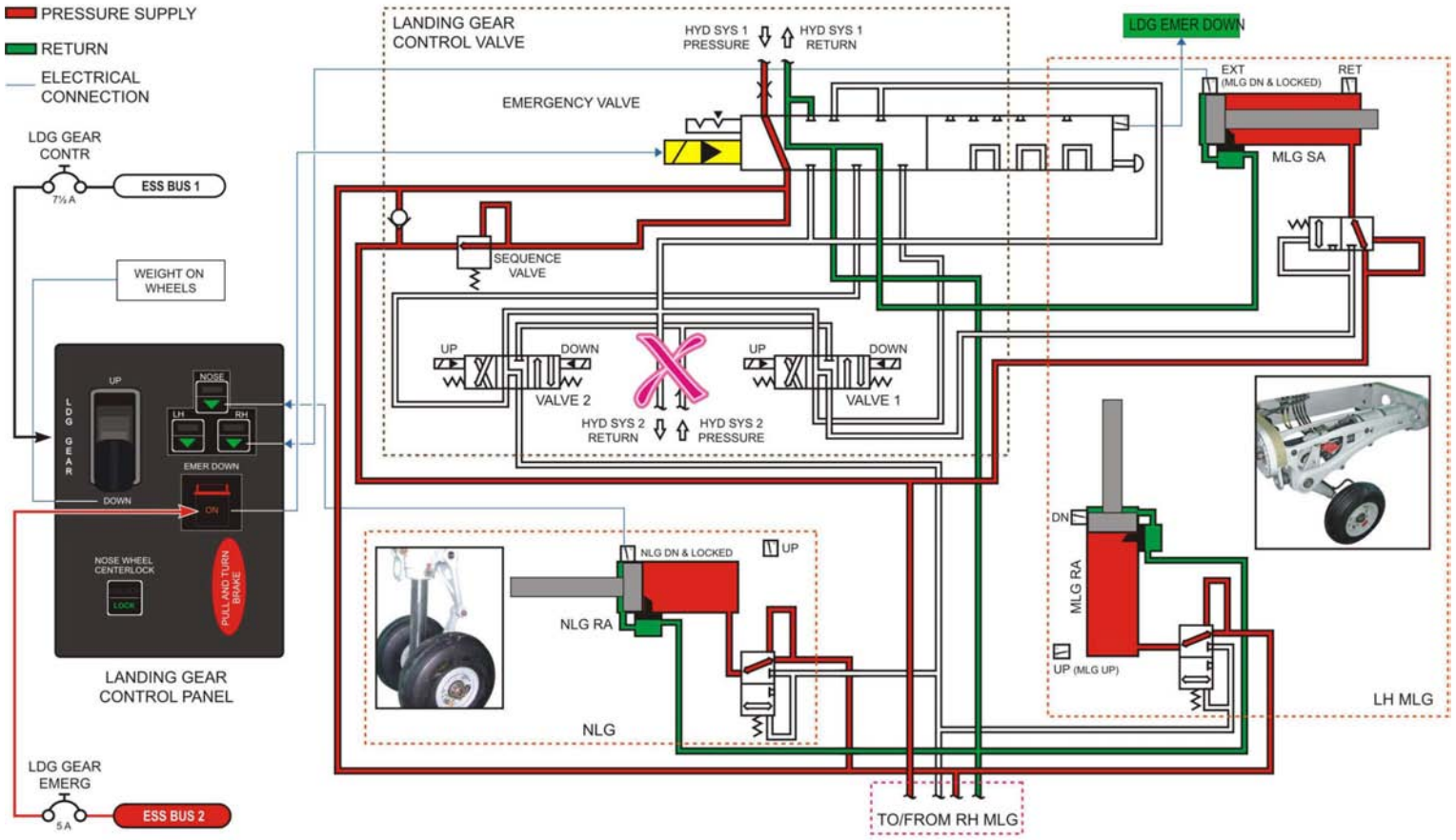


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EMERGENCY EXTENSION – PHASE 1 (TRANSITIONING)

PHASE 2 (TRANSITIONING)

When MLG RA and NLG RA are fully down, pressure restores in the lines and the sequence valve opens.
Pressure is delivered to MLG SA to elongate (extend) the MLG.

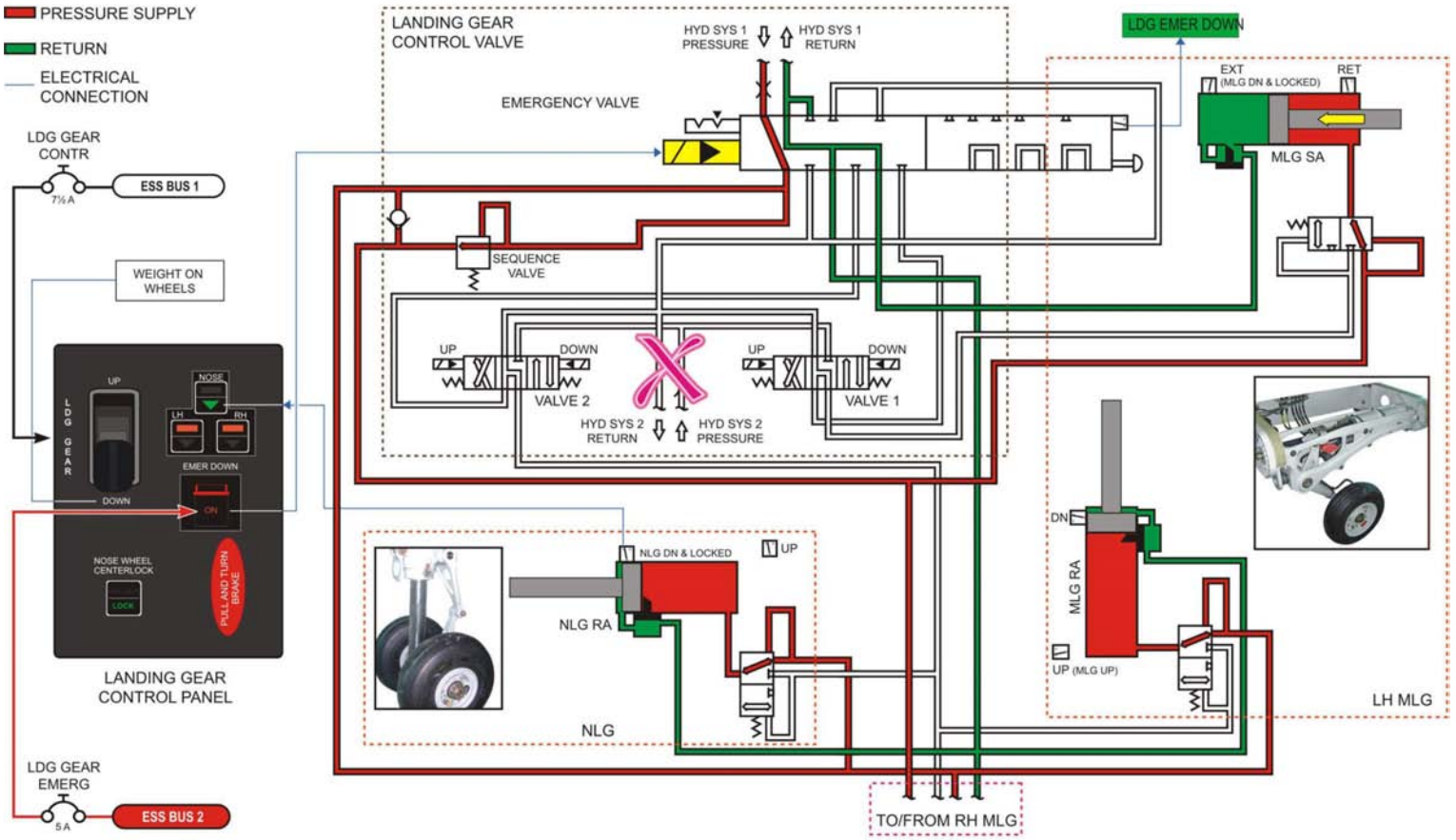


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EMERGENCY EXTENSION – PHASE 2 (TRANSITIONING)

PHASE 2 (END)

When all legs are down and locked, the three triangular green lights illuminate.

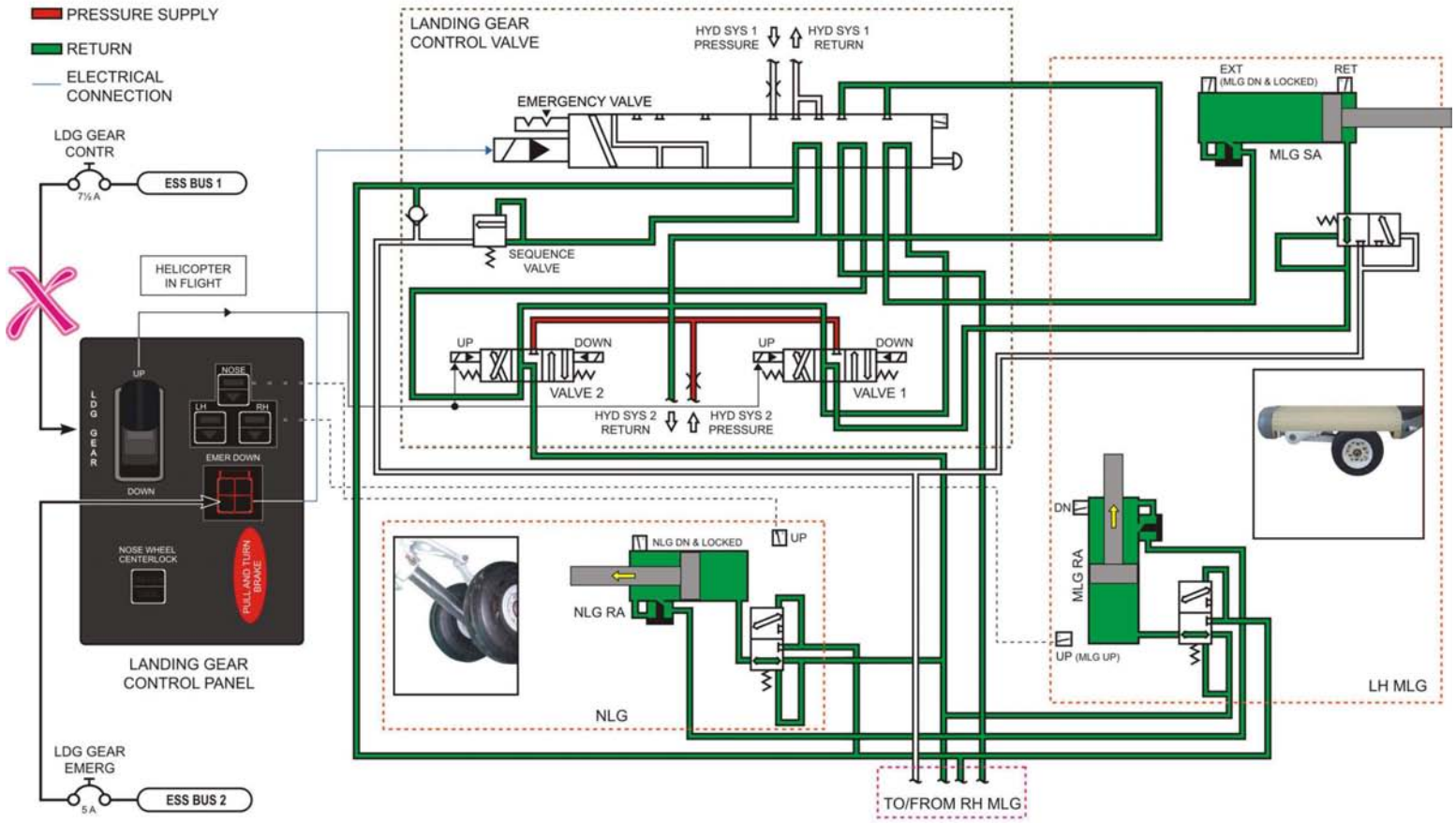


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EMERGENCY EXTENSION – PHASE 2 (END)

LOSS OF ELECTRICAL POWER WITH LG UP

In case of loss of electrical power from ESS BUS 1 or LDG GEAR CONTR circuit breaker tripped, the solenoids inside the LGCV de-energize and pressure is relieved from LG actuators and landing gear fall because of gravity. The LGCL is not effective and the lights in the LGCP cannot illuminate. Emergency extension is to be carried out.



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LOSS OF ELECTRICAL POWER WITH LG UP

CONTROLS AND INDICATORS

The retraction and normal/emergency extension controls and indicators consist of a Landing Gear Control Panel installed on the interseat console and caution and advisory messages provided in the CAS message window of the MFD in the instrument panel.

LANDING GEAR CONTROL PANEL

1 The Landing Gear Control Lever provides the following functions (lever must be lifted to operate):

- UP the landing gear is retracted (this position is inhibited on ground).
- DOWN the landing gear is extended.

2 The Landing Gear Control Panel lights provide the following indication:

- Green triangle: landing gear down and locked
- Amber bar: landing gear unlocked and/or in transition between up and down and viceversa
- Blank: landing gear retracted.

3 EMER DOWN pushbutton: ..... provides the emergency extension when selected. When pressed, the amber legend ON is lit.

4 LANDING GEAR: ..... when landing gear is still retracted at 150 feet of radio altimeter.

NOTE B

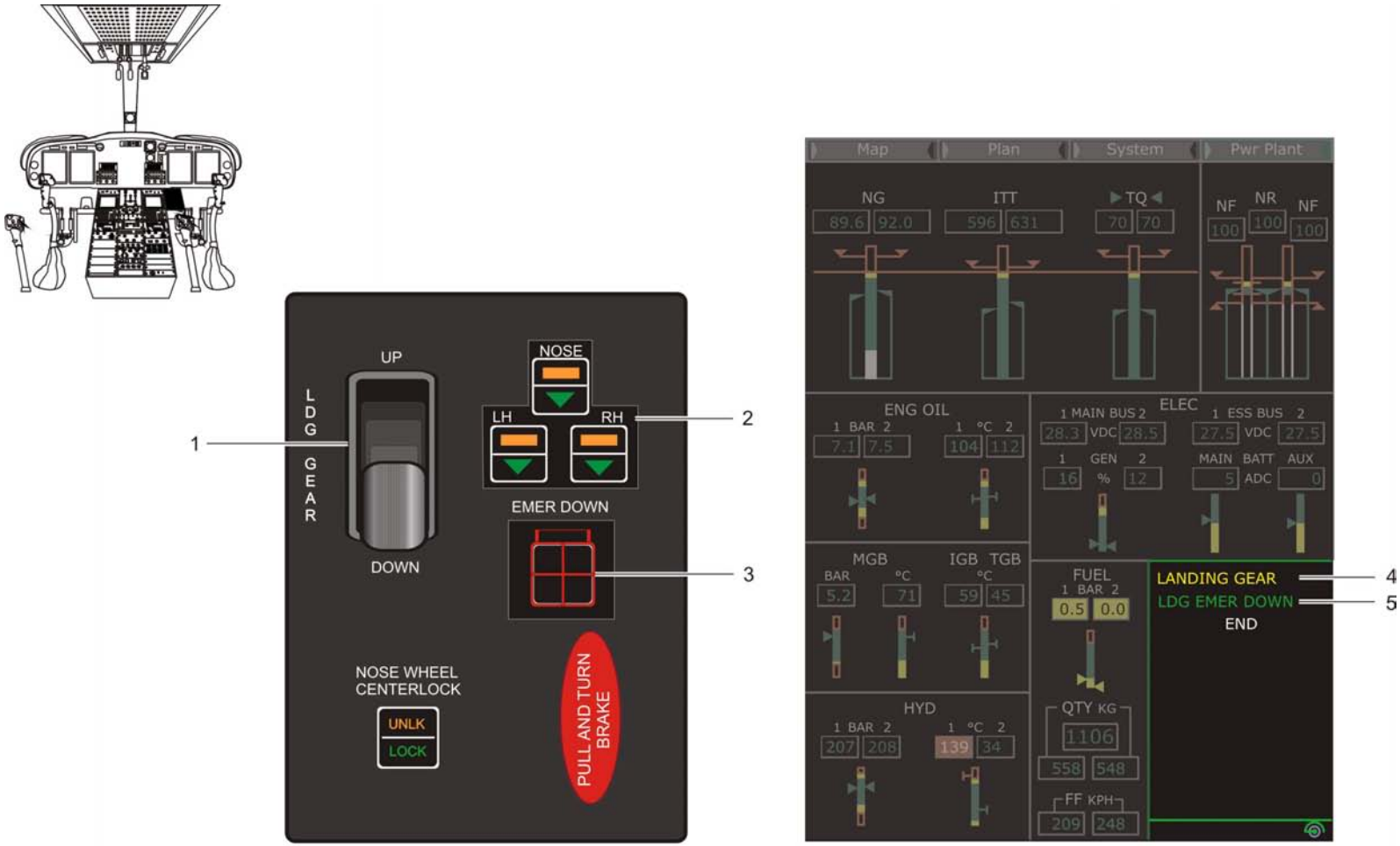
When this caution is detected, besides the MCL activation, the AWG provides the following:

- a pair of tones (554 thru 440 Hz) + LANDING GEAR (aural message).

This sequence is transmitted once only.
This message has priority number 8.

The advisory provided in the CAS message window is the following:

5 LDG EMER DOWN: ..... when the EMER DOWN pushbutton is selected.

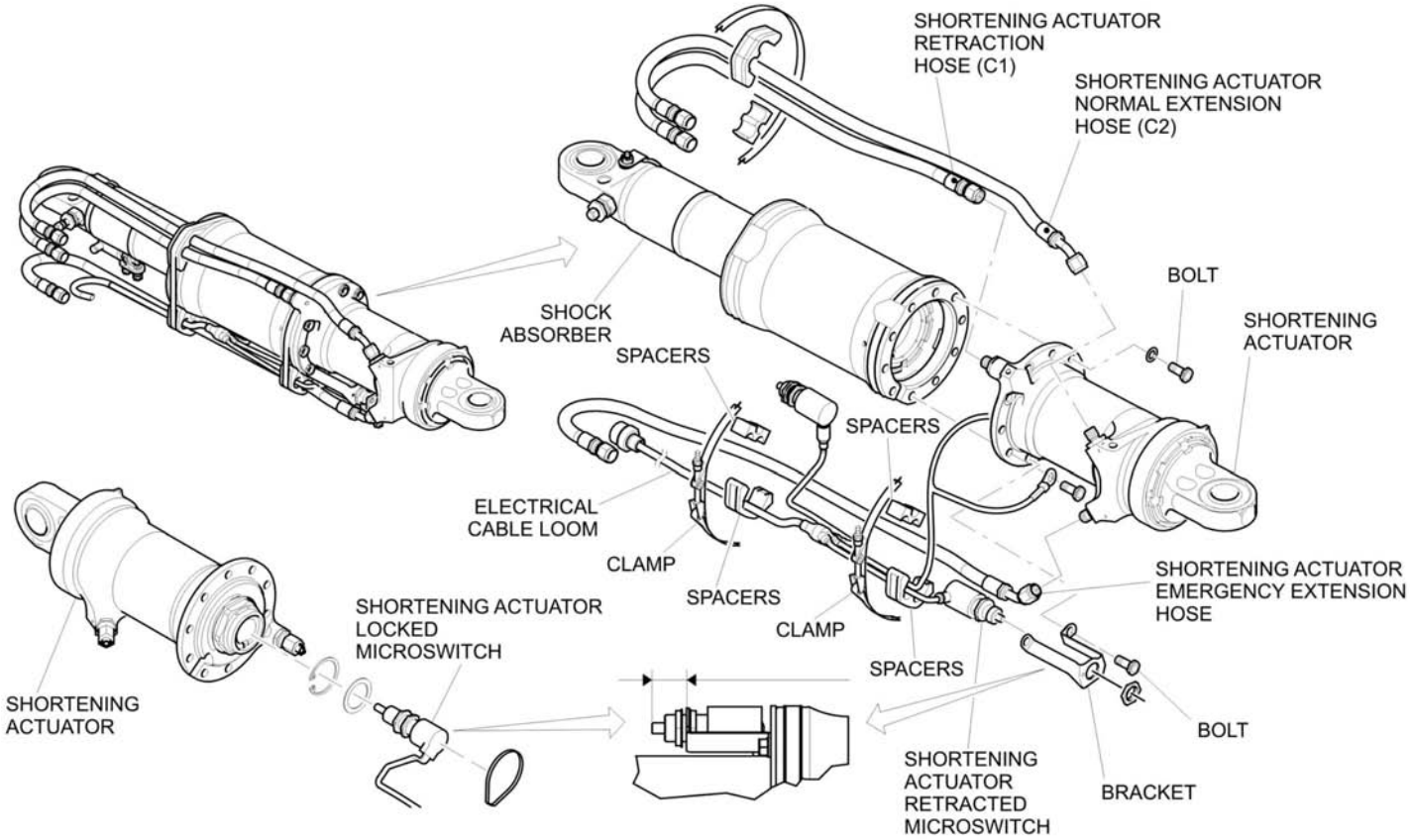


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CONTROLS AND INDICATORS

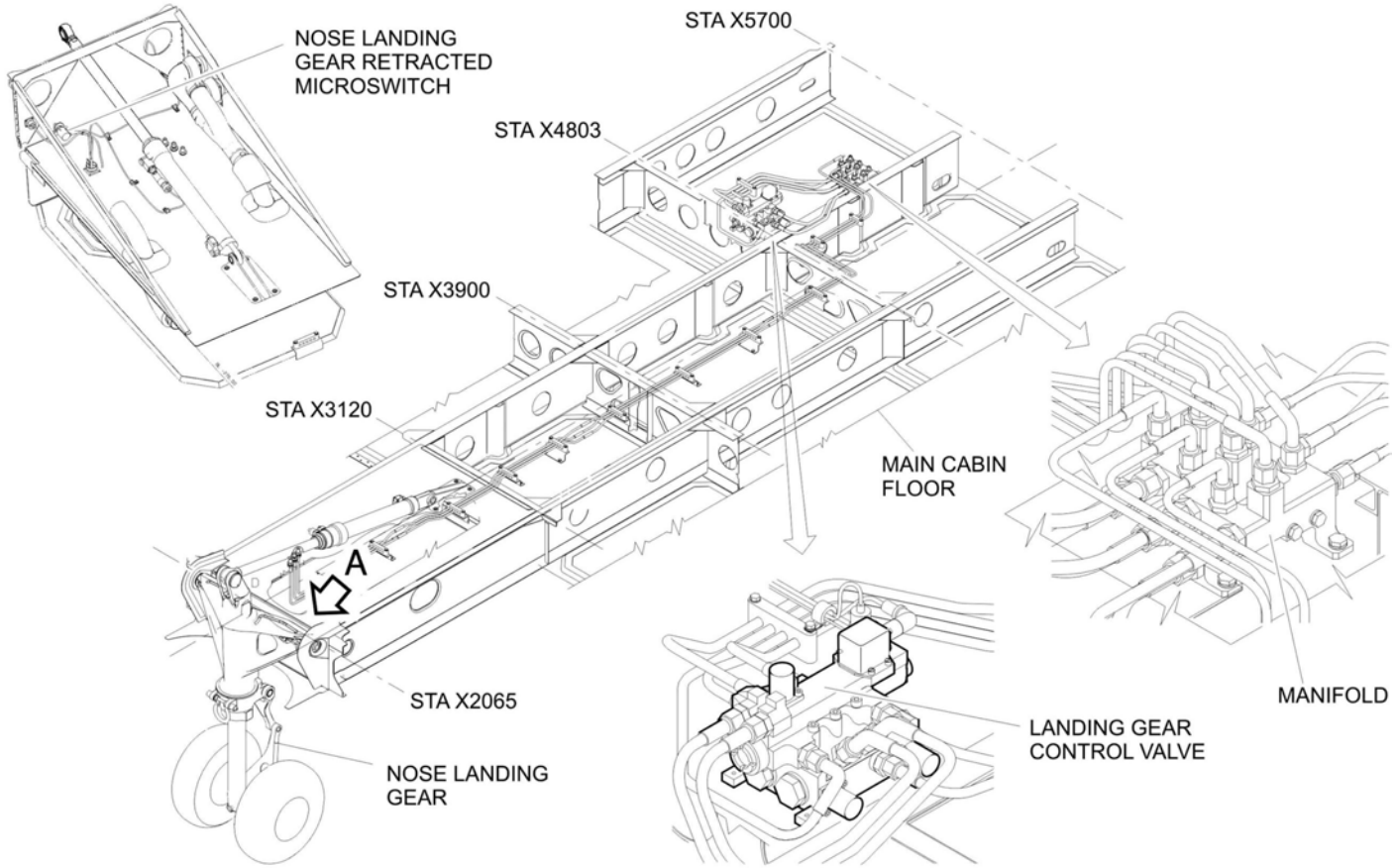
MAINTENANCE INFORMATION

- Identify the location of the electrical cables tie-down straps and record routing of the electrical cables in the landing gear bay before actuators removal.
- Shock absorber and shortening actuator must be aligned for installation.
- Lift A/C on jack for NLG and MLG actuators removal.
- Identify the position of the hydraulic lines and tie-down straps connected to valve block before shortening actuator removal.
- Bleed and operational check of the landing gears system must be carried out after replacement of
  - Landing gear control valve
  - Manifold
  - NLG and MLG extension/retraction actuator
  - MLG shortening actuator



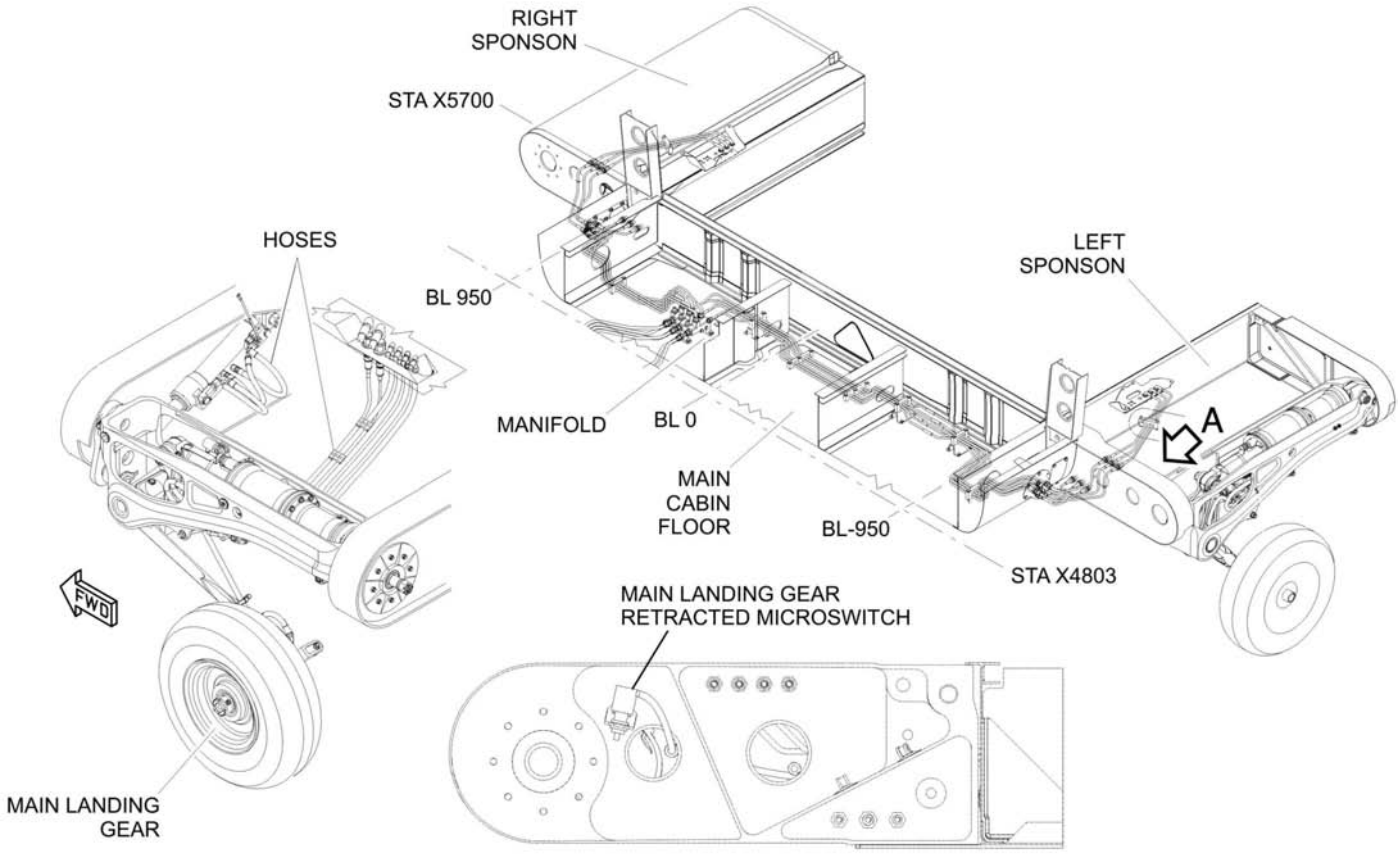
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MLG PARTIAL RETRACTION ACTUATOR – MICROSWITCHES (SHEET 1 OF 3)



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EXTENSION/RETRACTION SYSTEM - MICROSWITCHES (SHEET 2 OF 3)



ICN-39-A-323000-G-A0126-01372-A-01-1

EXTENSION/RETRACTION SYSTEM - MICROSWITCHES (SHEET 3 OF 3)

LEADING PARTICULARS

- Landing gears extension time (normal)..... 8 thru 10 sec
- Landing gears extension time (emergency)..... 12 thru 18 sec
- Landing gears retraction time..... 10 thru 15 sec
- Force required to lift the LGCL..... 18 N
- Force required to move the LGCL up or down..... 27 N
- Landing Gear Control Valve weight..... 4.3 Kg (9.5 lb)
- MLG extension/retraction actuator weight..... 3.4 Kg (7.5 lb)
- NLG extension/retraction actuator weight..... 4.9 Kg (10.8 lb)
- LGCP electrical supply:
  - Normal..... 28VDC ESS BUS1
  - Emergency..... 28VDC ESS BUS2

SAFETY PRECAUTIONS

The material that follow are dangerous.
Make sure that you know all the safety precautions and first aid instructions for these materials:

- Solvent
- Grease

SPECIAL TOOLS/TEST EQUIPMENT

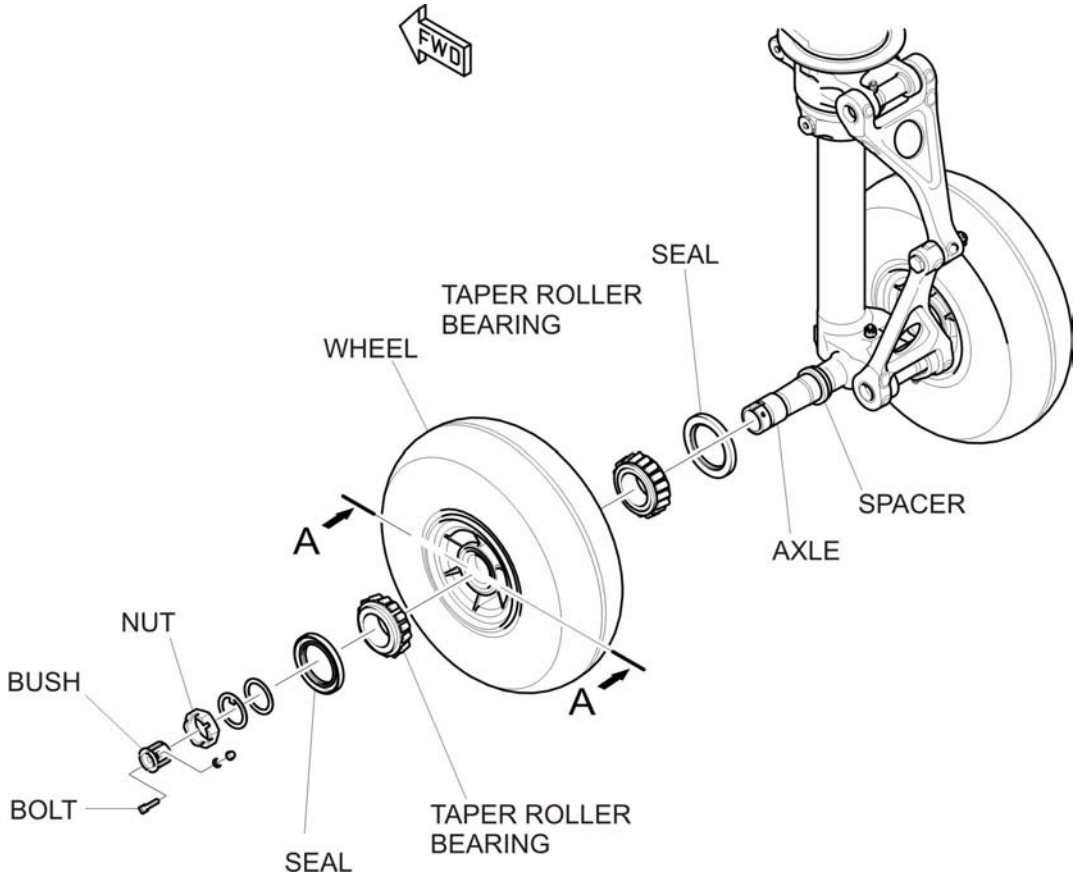
- Wrench

SECTION 40 - WHEELS AND BRAKES

DESCRIPTION

NOSE LANDING GEAR WHEELS

The nose landing gear wheel assembly consists of two units of a tube type tyre installed on a bolted two half aluminium hub.
The wheels are installed on the axle at the bottom of the sliding piston rod using two cone bearings protected by rubber seals.
Each wheel assembly is locked on to the axle by a nut secured with two bolts through a bush which is also used for the attachment of the pins of the towing bar.

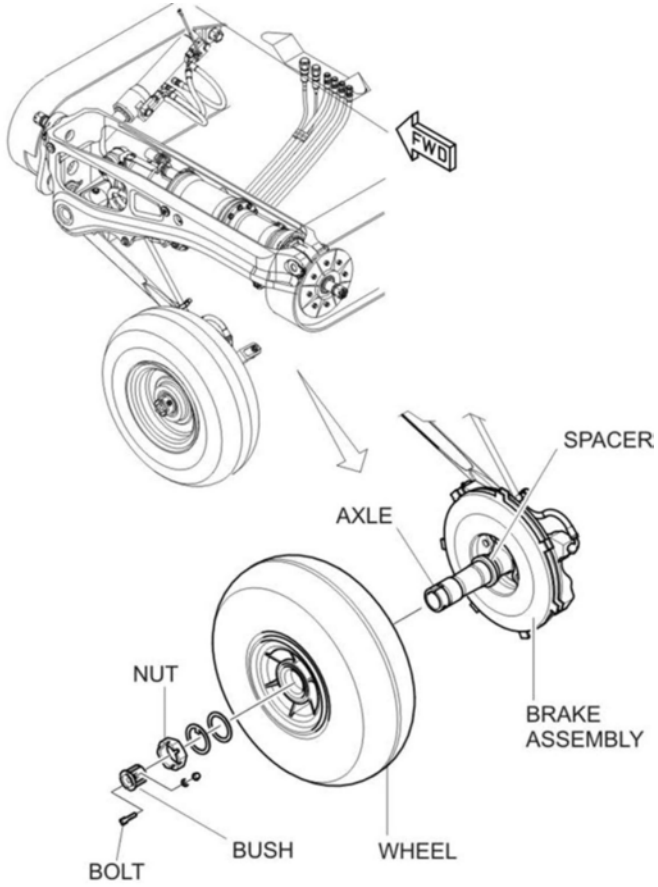


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NOSE LANDING GEAR WHEELS

MAIN LANDING GEAR WHEELS

The main landing gear wheel assembly consists of a tubeless tyre installed on a bolted two half aluminium hub. The outer hub incorporates a safety relief plug to permit a release of over pressure in the wheel.
The inner hub half is provided with grooves for the interface with the brake assembly and with two fusible plugs which release tyre pressure in case of the wheel overheating.
The wheel is installed on the axle at the bottom of the trailing arm using two cone bearings protected by rubber seals.
The wheel assembly is locked on to the axle by a nut secured with two bolts through a bush.



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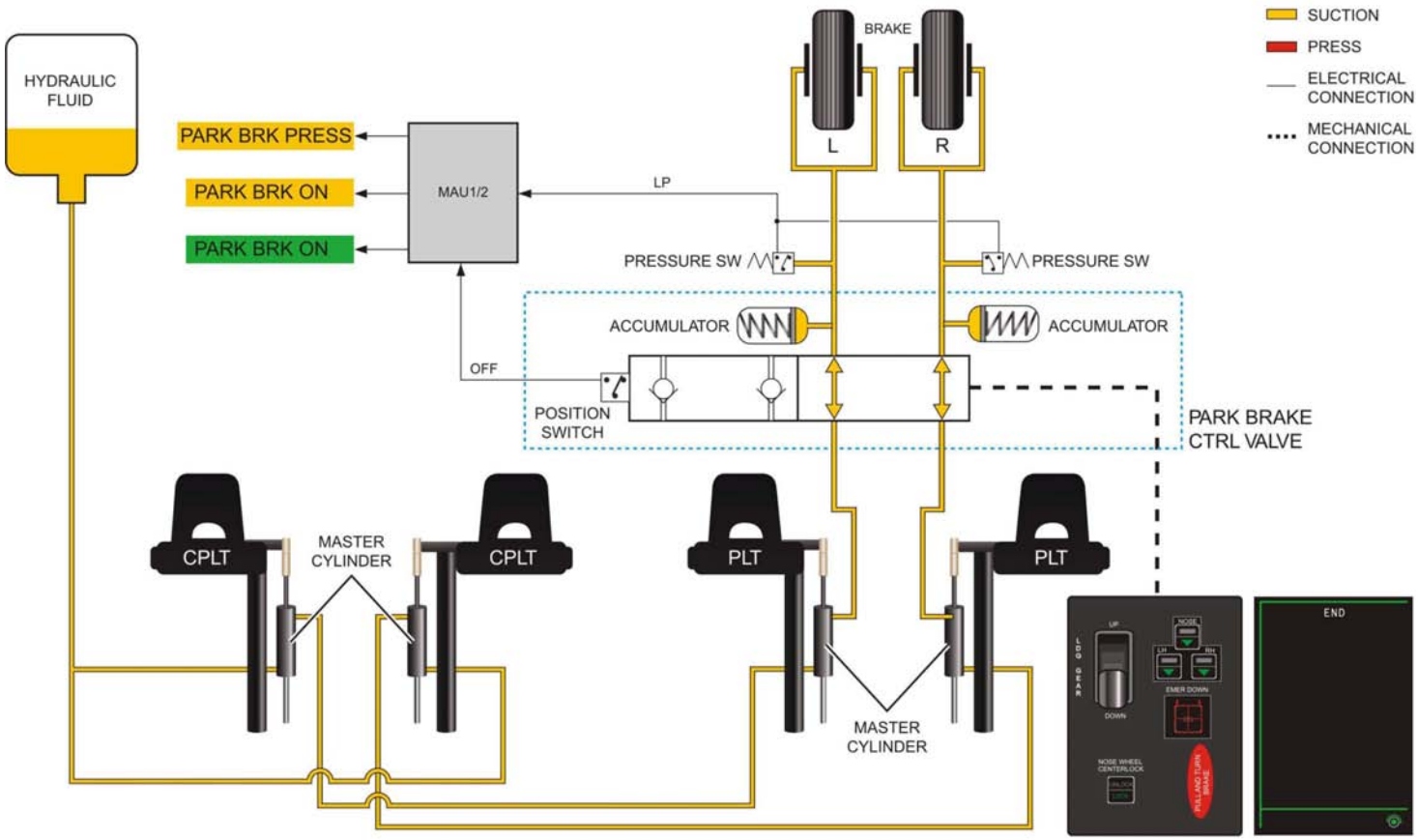
MAIN LANDING GEAR WHEELS

MAIN LANDING GEAR BRAKES

The main landing gear brake assembly consists of an aluminium housing containing five pistons which operate through an independent hydraulic circuit, pressure and reaction plates, two rotor and one stator disks and a wear indicator. The brake assembly is not provided with an automatic wear adjustment.
Two hydraulic fittings are located on the housing, one for hydraulic pressure supply, the other for hydraulic bleeding.
The dynamic brake function is achieved through pedal levers installed on the yaw pedals each operating hydraulic master cylinder located underneath which generates the required pressure to the brake assembly using the hydraulic fluid contained in a reservoir located in the nose compartment. The pilot hydraulic master cylinders are supplied through the copilot ones.
The static braking is achieved through a parking brake handle located in the Landing Gear Control Panel which is used to trap pressure provided by the brake pedals by operating, through a push-pull cable, a parking brake valve such to keep the brake assembly engaged.
Pressure is trapped by operating a shuttle valve provided with two check valves.
A spring loaded pin locks the push-pull cable in the engaged position every time the handle is pulled.
To remove the locking action is required to push the left brake pedal such that the hydraulic pressure overcome the spring load of the locking pin which retracts and allows cable movement.
A hydraulic accumulator in the parking brake valve is provided for each hydraulic pipeline to compensate for fluid leaks and/or pressure changes due to temperature effect.
Parking on slopes up to 10° is permitted for a maximum of 1 hour.
The parking brake valve is installed on the right side in the nose landing gear bay.
A pressure switch in installed on each hydraulic pipeline supplying the brake assembly.
A position microswitch is installed in the parking brake valve.

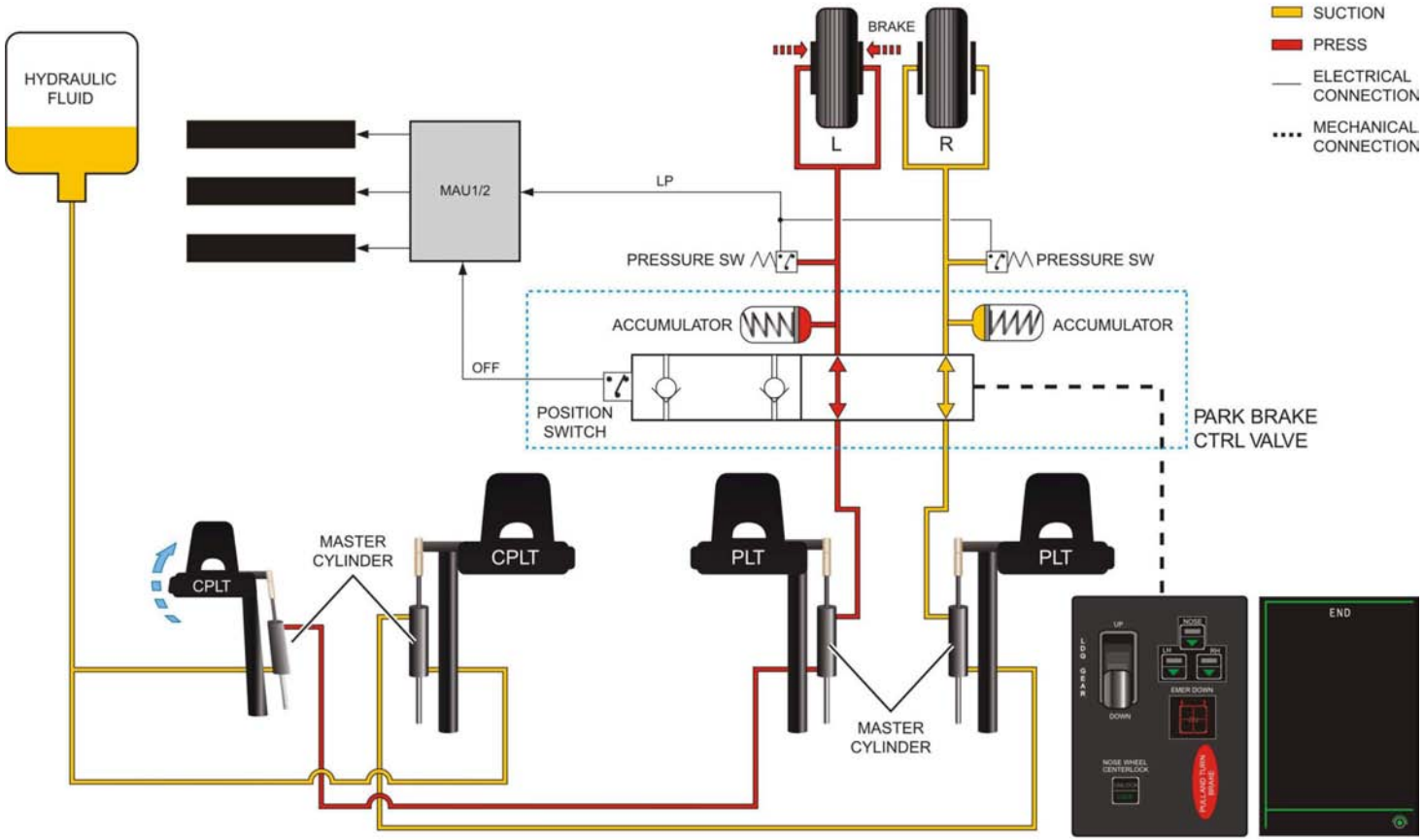
MAIN LANDING GEAR BRAKES OPERATION

- Pilot or copilot pushes the brake pedal sucking oil from the reservoir to each right and left master cylinder which generates the required pressure as a function of the force applied on the pedal.
- Pressure is generated by force applied on the pedal acting on a specific designed area in the hydraulic master cylinder.
- Pressure is delivered through hydraulic pipelines connected with the parking brake valve to the brake assembly callipers.
- There’s no priority between pilot and copilot that means that whoever is applying greater force on the pedals takes control of the braking function.
- Differential braking can be achieved during taxiing on ground by operating only left or right pedal to obtain the helicopter dynamic steering (the lock of the nose wheel in center position must be disengaged).
- Static braking is achieved when the helicopter is parked by pulling and rotating the parking brake handle and pushing on the brake pedals.
- The two check valves contained in the parking brake valve trap the pressure provided by the brake pedals between the parking brake valve and the brake assembly callipers.
- If pressure, detected by the pressure switch in the pipeline, is satisfactory the advisory PARK BRK ON is provided to the MFD.
- If pressure is not satisfactory (below a required value), the caution PARK BRK PRESS is provided to the MFD. In this case pilot/copilot needs to push again/harder on the brake pedals until the caution disappears.
- The caution and advisory messages are operative only if the parking brake handle is pulled (detected by the parking brake valve position microswitch).



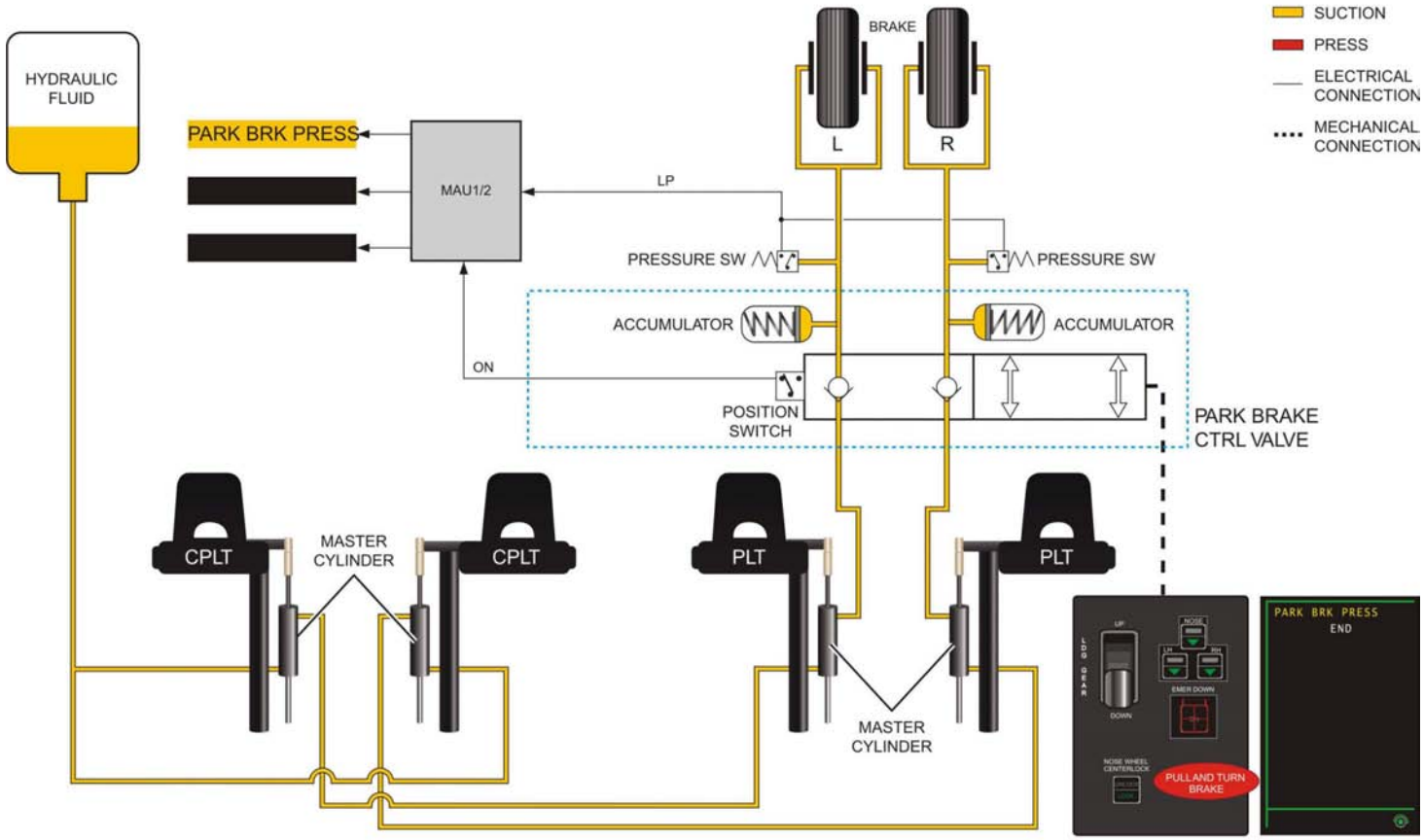
ICN-39-A-324000-G-A0126-01380-A-01-1

WHEELS BRAKE SYSTEM SCHEMATIC



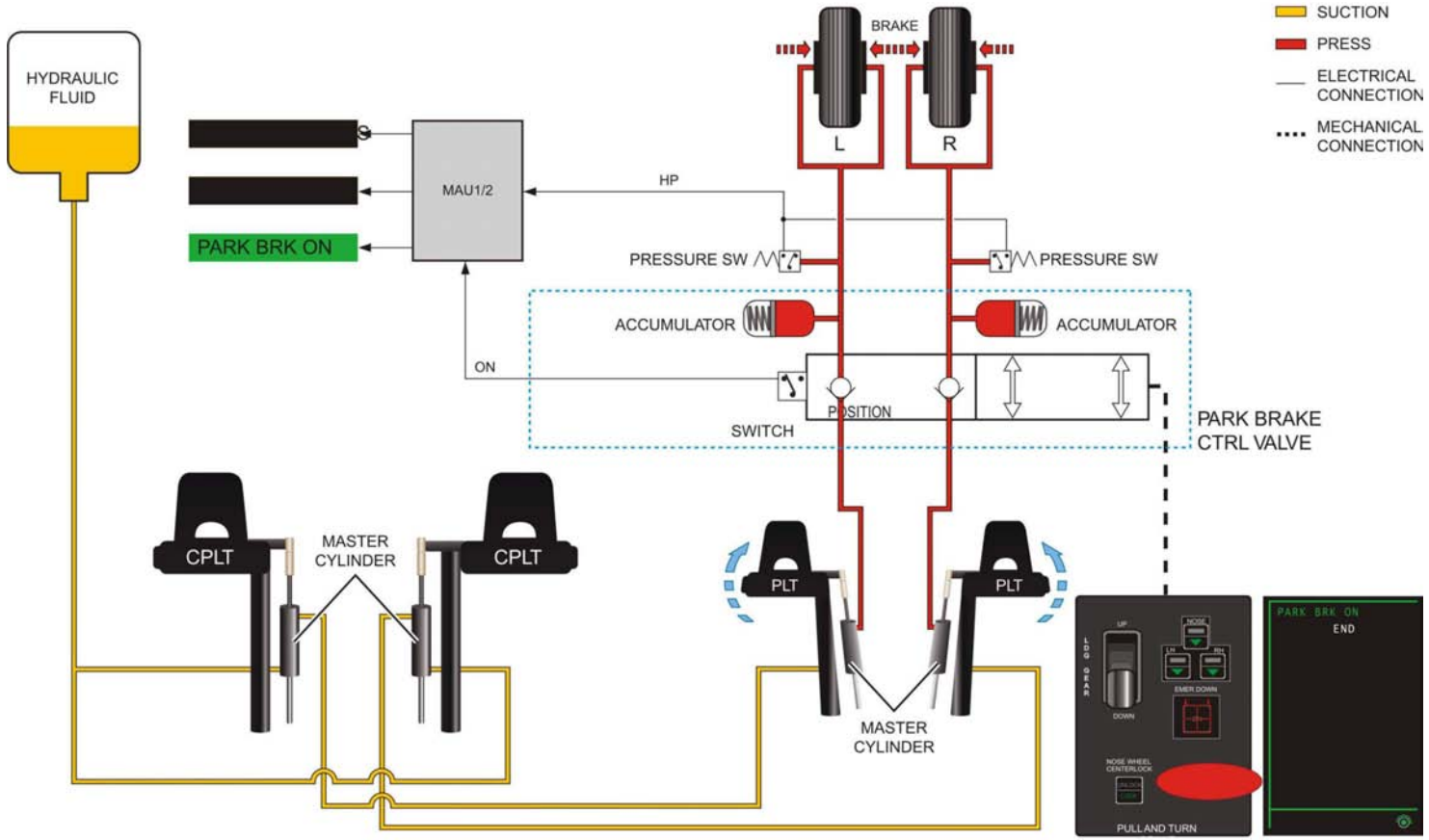
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EXAMPLE OF NORMAL BREAKING (CPLT)



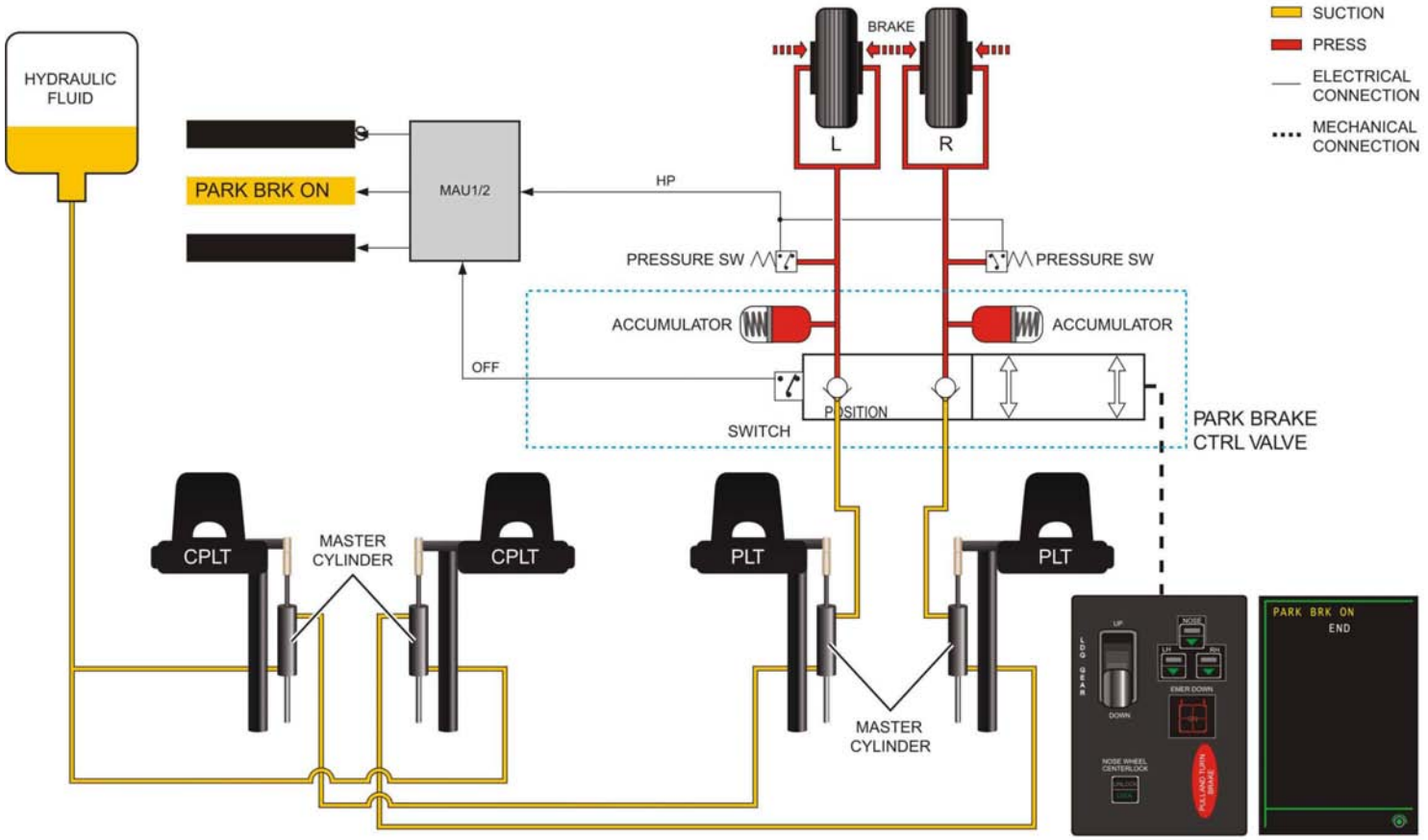
ICN-39-A-324000-G-A0126-01382-A-01-1

PARKING BRAKE (1 OF 2)



ICN-39-A-324000-G-A0126-01383-A-01-1

PARKING BRAKE (2 OF 2)



ICN-39-A-324000-G-A0126-01384-A-01-1

WHEELS BRAKE SYSTEM SCHEMATIC (PARKING BRAKE APPLIED)

CONTROLS AND INDICATORS

The MLG wheel and brake controls and indicators consist of a pilot/copilot brake pedal on the yaw control, a park brake handle on the LGCP and caution/ advisory messages provided in the CAS message window of the MFD in the instrument panel.

1 Pilot/Copilot Brake Pedals: ….. the brake pedals when operated supply pressure to the brake assembly.

2 Landing Gear Park Brake Handle: ..... the handle can be pulled and rotated 90° clockwise. When operating the pilot/copilot brake pedals pressure is trapped in the pipelines to the brake assembly to keep the wheels locked when the trapped pressure is above a required value.
It is possible to reset the handle only if the left brake pedal is pushed, then pressure is drained and the lock removed.

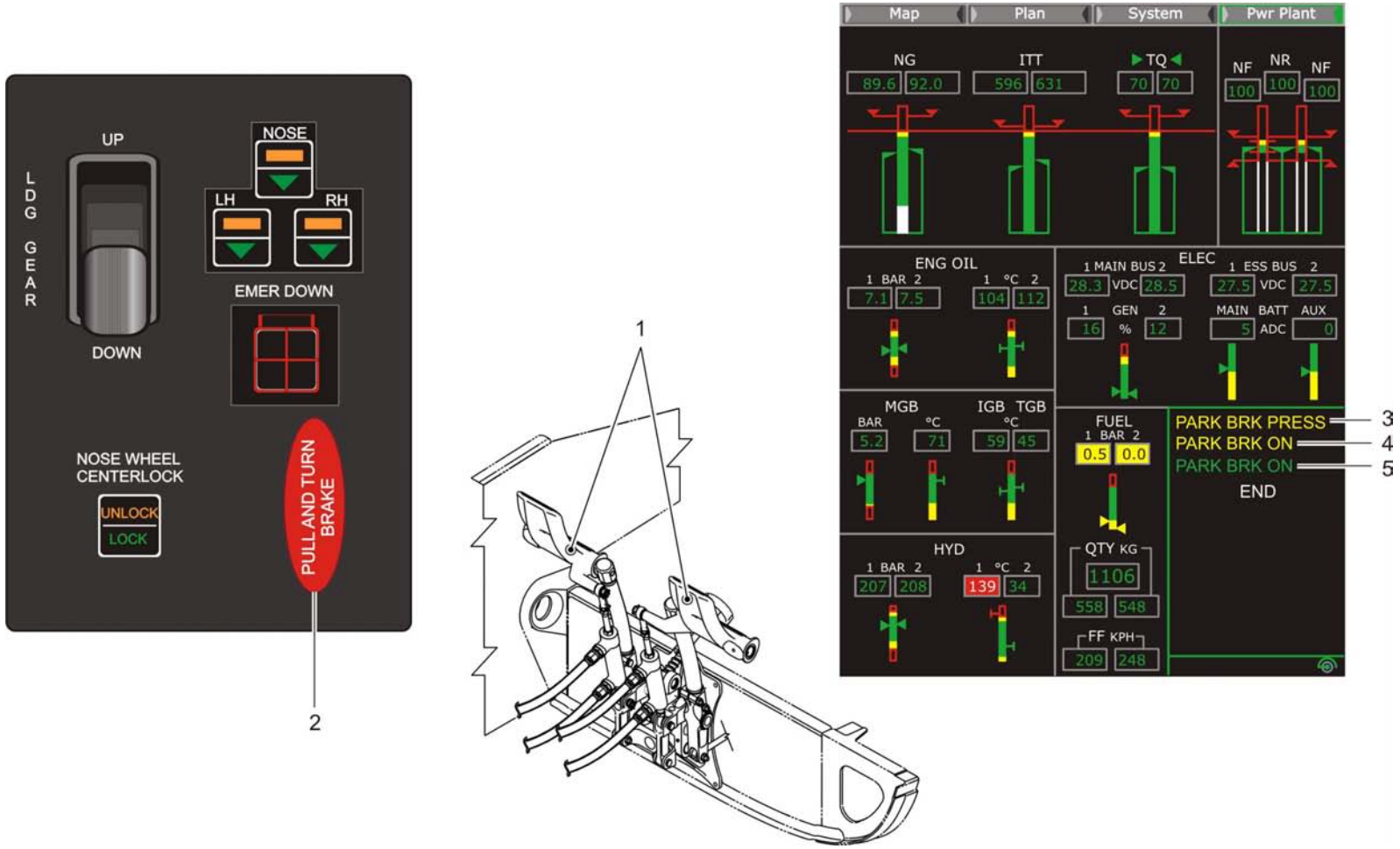
The cautions provided in the CAS message window are the following:

3 PARK BRK PRESS: ..... when the trapped pressure on each line is below the required value when the park brake handle is pulled.

4 PARK BRK ON: ..... when the trapped pressure on each line is above the required value when the park brake handle is released.

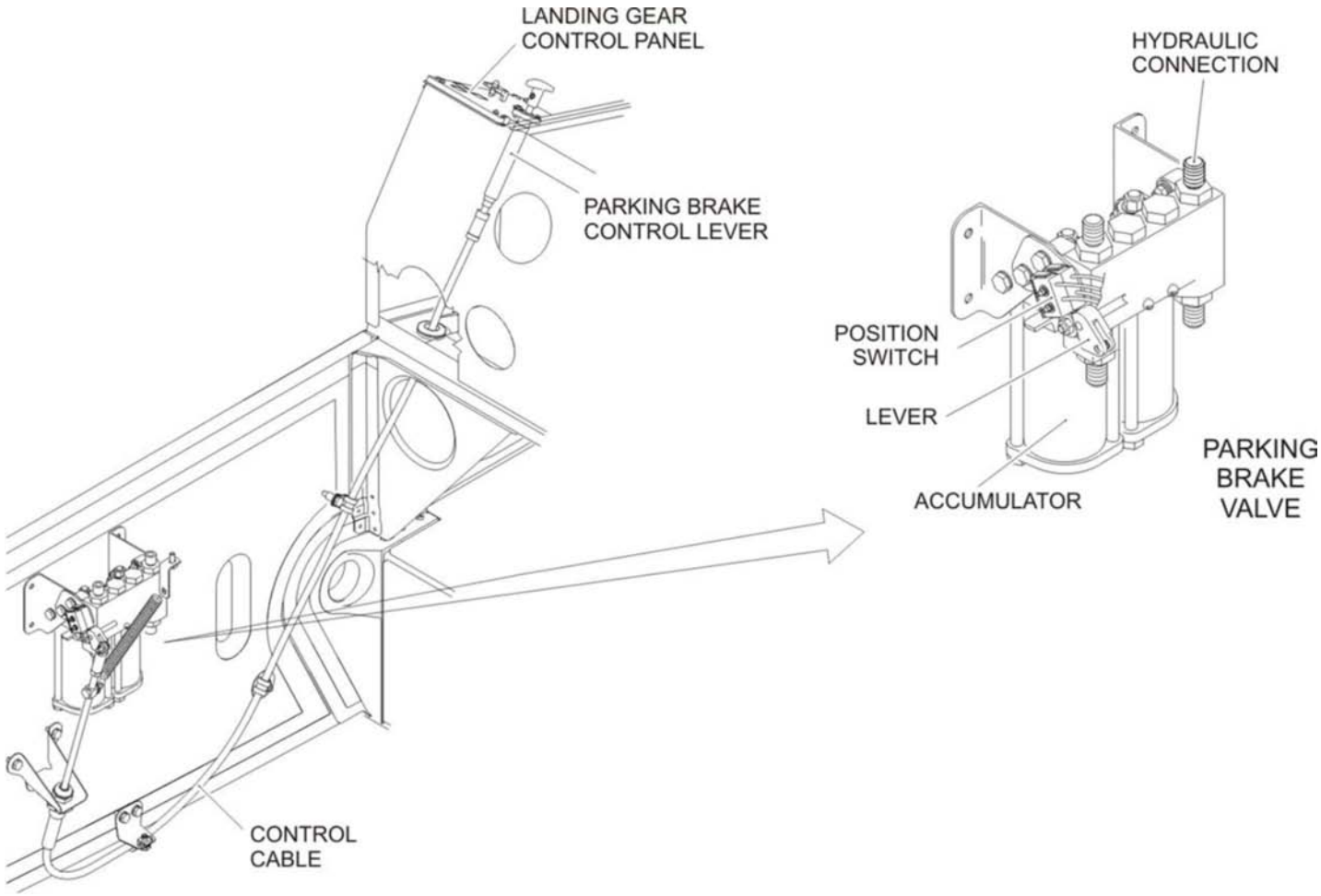
The advisory provided in the CAS message window is the following:

- PARK BRK ON :..... when the handle is pulled (and rotated) and the trapped pressure is above the required value.



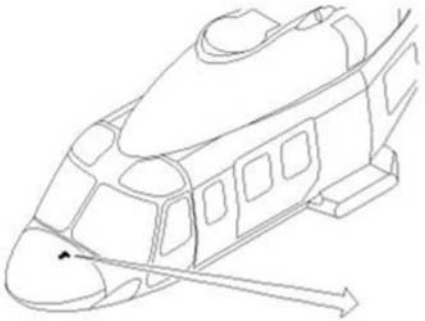
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CONTROLS AND INDICATORS

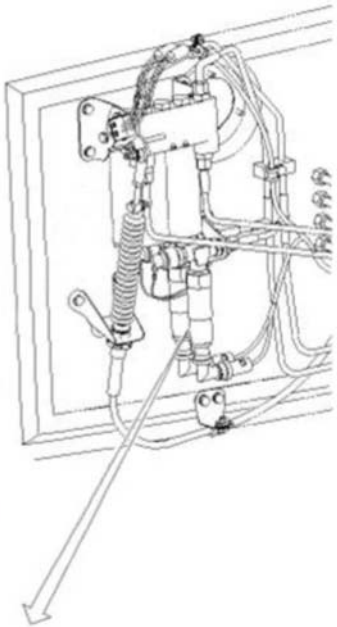


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PARK BRAKE CONTROL LEVER - PARK BRAKE VALVE



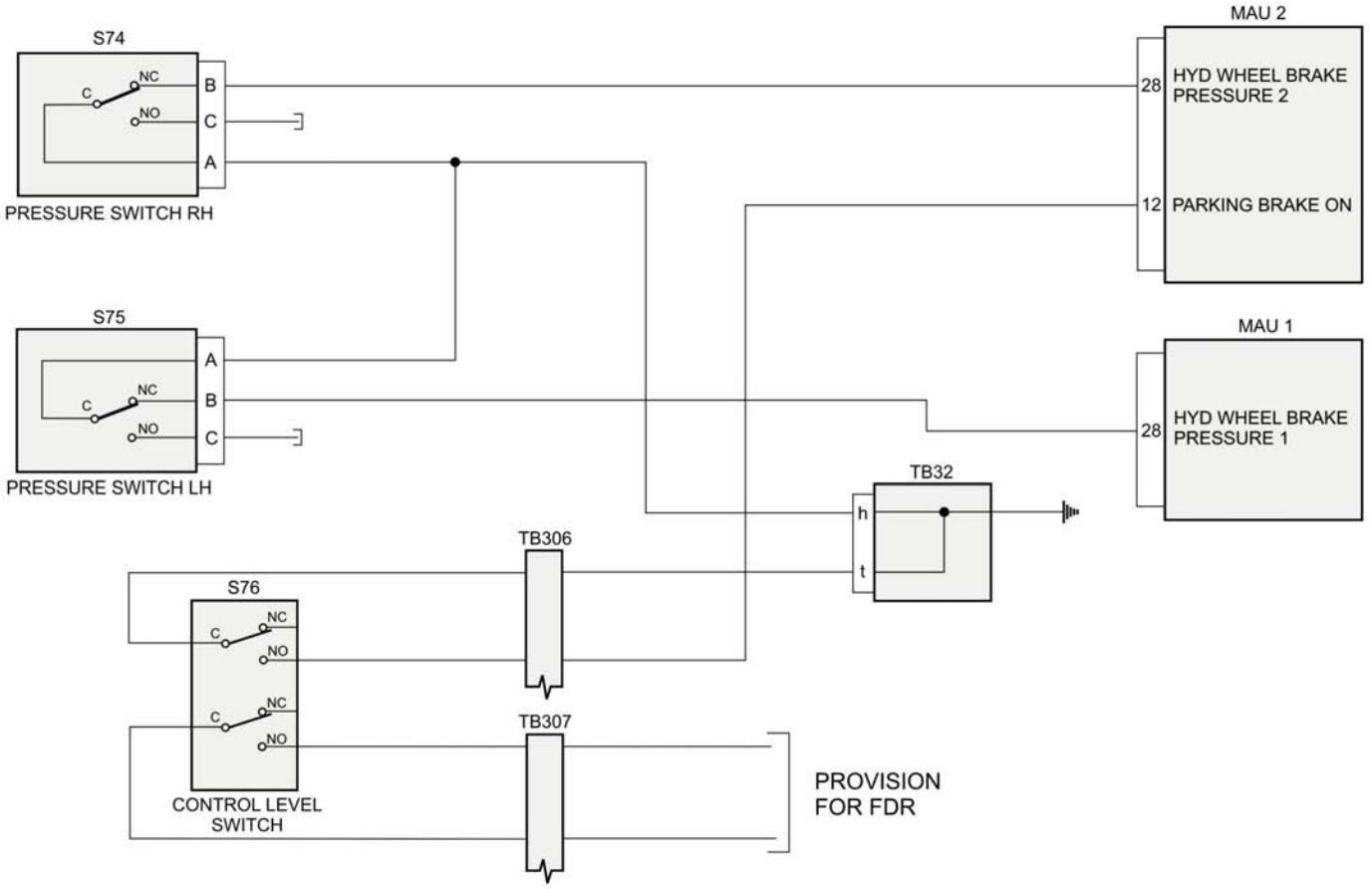
HELICOPTERS [1J] SHOWN
TYPICAL FOR HELICOPTERS [1L]





ICN-39-A-324000-G-A0126-03329-A-01-1

PRESSURE SWITCHES



ICN-39-A-324000-G-A0126-01391-A-01-1

WHEEL BRAKE SYSTEM SCHEMATIC

PARTICULARS

- Hydraulic fluid..... MIL-PRF-83282
- NLG wheel (including tyre and tube) weight..... 4.9 Kg (10.8 lb)
- MLG wheel (including tyre) weight..... 10.4 Kg (22.9 lb)
- Nose wheels tyre inflation pressure..... 9.1 bar (132 lb)
- Main wheels tyre inflation pressure..... 15.85 bar (230 psi)
- Brake assembly weight..... 7.5 Kg (16.5 lb)
- Hydraulic brake system reservoir capacity… 0.140 l (0.12 Us quart)
- Parking brake pressure switch:
  - Decreasing..... 29 thru 35 bar (420 thru 510 psi)
  - Increasing..... 35 thru 45 bar (510 thru 625 psi)

SAFETY PRECAUTIONS

The materials that follow are dangerous.
Make sure that you know all the safety precautions and first aid instruction for these materials:

- Grease
- Hydraulic fluid
- Solvent

SPECIAL TOOLS/EQUIPMENT

- Wheel brake bleeding kit.

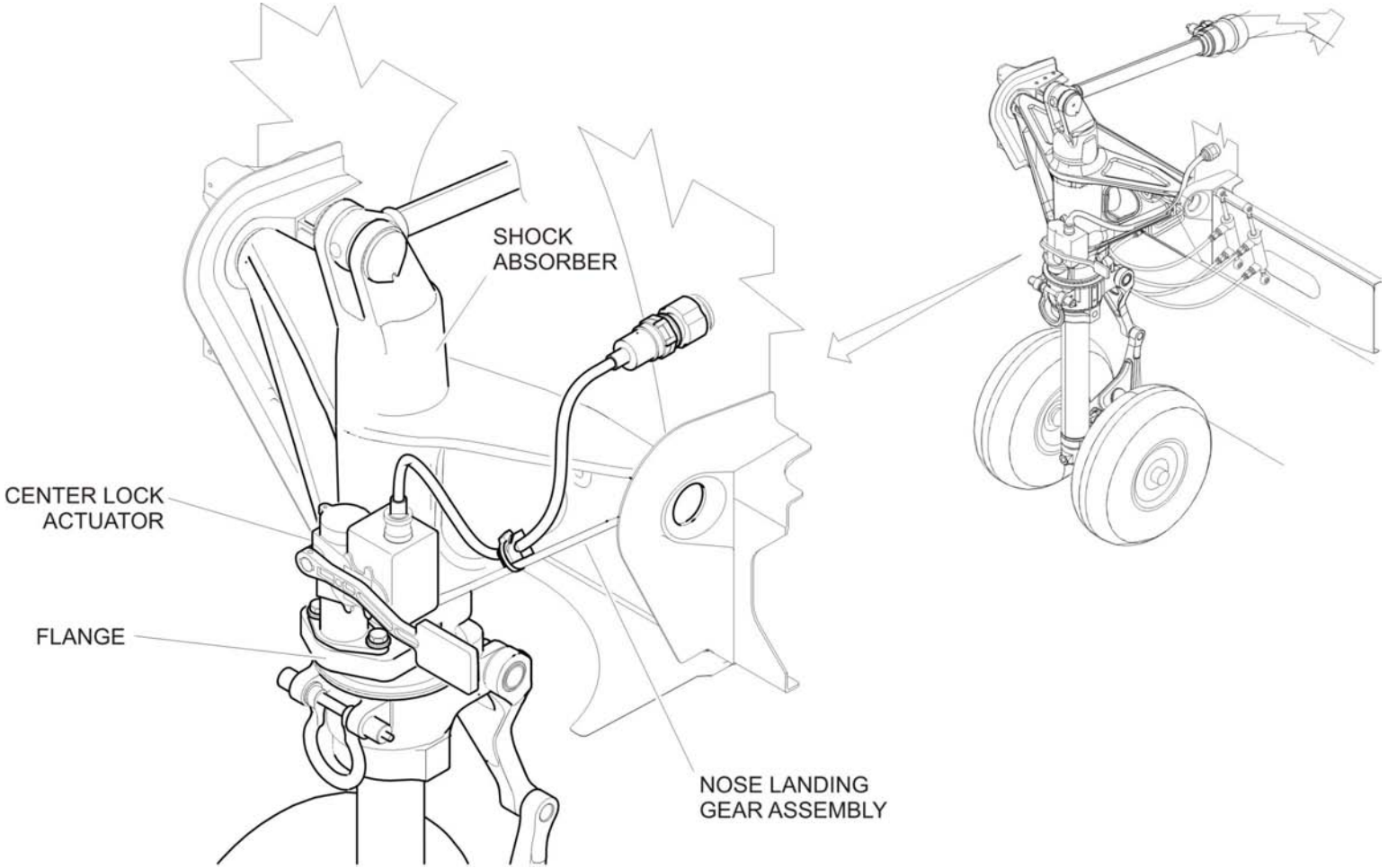
SECTION 50 - STEERING

DESCRIPTION

The steering system is designed so that the nose landing gear can not be retracted when the wheels are not aligned.
The steering system provides an automatic nose landing gear wheels centering and pilot/ground crew wheels position lock engagement/ disengagement.
The nose landing gear steering system consists of the following main components:

- Centering assembly
- Center lock assembly including nose landing gear locked in center position microswitch.

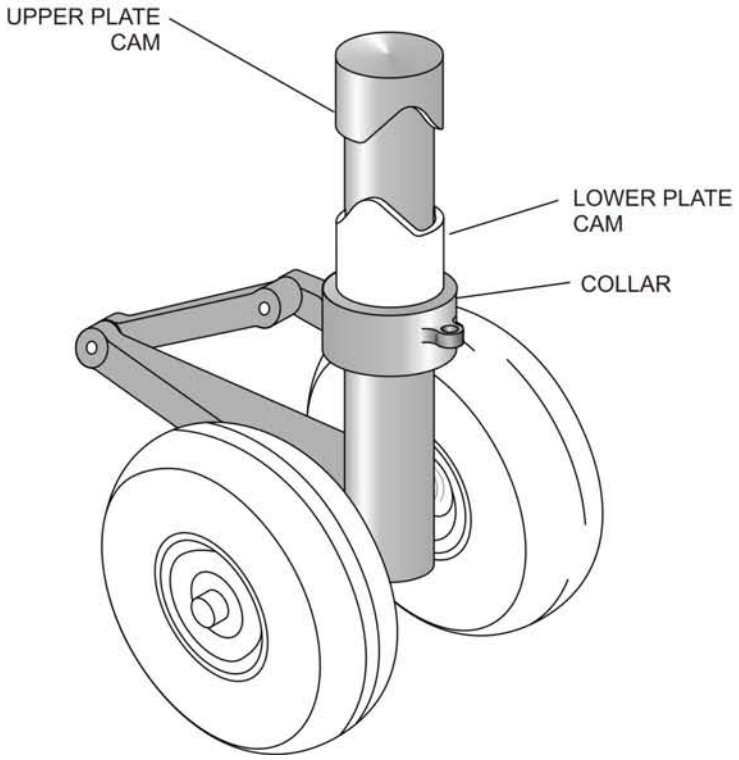
The centering assembly consists of two cam shaped plates inside the shock absorber, one installed on the top outside of the sliding piston rod and the other installed on the bottom inside of the shock absorber cylinder.
When the nose landing gear is centered the two cam plates are matched together providing the fore and aft alignment of the nose landing gear.
Should the pilot take-off with the nose landing gear out of the fore and aft position, when the sliding piston rod moves downwards to its fully extended position by effect of its own weight, the upper plate slips on the lower plate profile forcing the sliding piston rod to rotate (together with the wheels) such that the two shaped plates match together in the fore and aft direction.
The centering system can work properly if the disalignment angle between wheels and helicopter fore and aft axis is a maximum of ± 115°.
The center lock assembly consists of an electrical DC motor incorporating a gear assembly that drives a locking pin. The assembly is installed on the bottom front side of the shock absorber cylinder.
The locking pin can be engaged/disengaged electrically through a switch on the Landing Gear Control Panel or mechanically by the rotation of a lever connected to the assembly.
The locking pin fits into a hole provided on a flange which is part of a collar installed on a bush on the bottom of the shock absorber cylinder.
The collar rotates on the bush through the upper torque link connected on the back side.
Two grease nipples provide the lubrication between the bush and the collar.
On the front side of the collar is installed a mooring ring.
The NLG locked in center position microswitch is located inside the center lock assembly and is operated when the center lock assembly locking pin is engaged in the hole of the flange of the collar installed at the bottom end of the shock absorber cylinder.
The steering system also includes an anti-shimmy device which is composed by an oil film filled between the collar and the bush.
The oil viscosity changes in accordance with the speed of the relative movement between the bush and the collar.

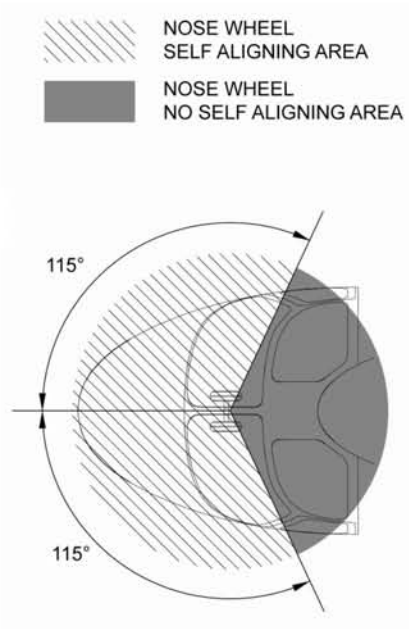


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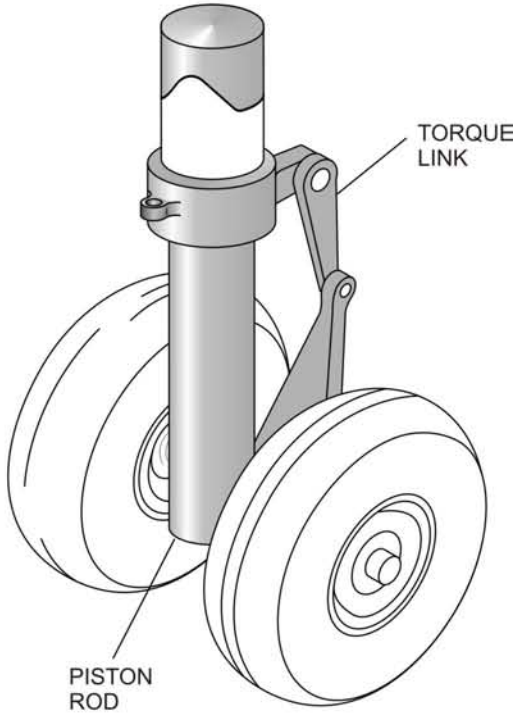
CENTER LOCK ASSEMBLY

NOSE LANDING GEAR NOT ALIGNED
(ON GROUND)





NOSE LANDING GEAR ALIGNED
(IN FLIGHT)



ICN-39-A-325000-G-A0126-03332-A-01-1

CENTERING ASSEMBLY

CONTROLS AND INDICATORS

The NLG steering controls and indicators consist of a pushbutton on the Landing Gear Control Panel and a caution message provided in the CAS message window of the MFD in the instruments panel.

Landing Gear Control Panel

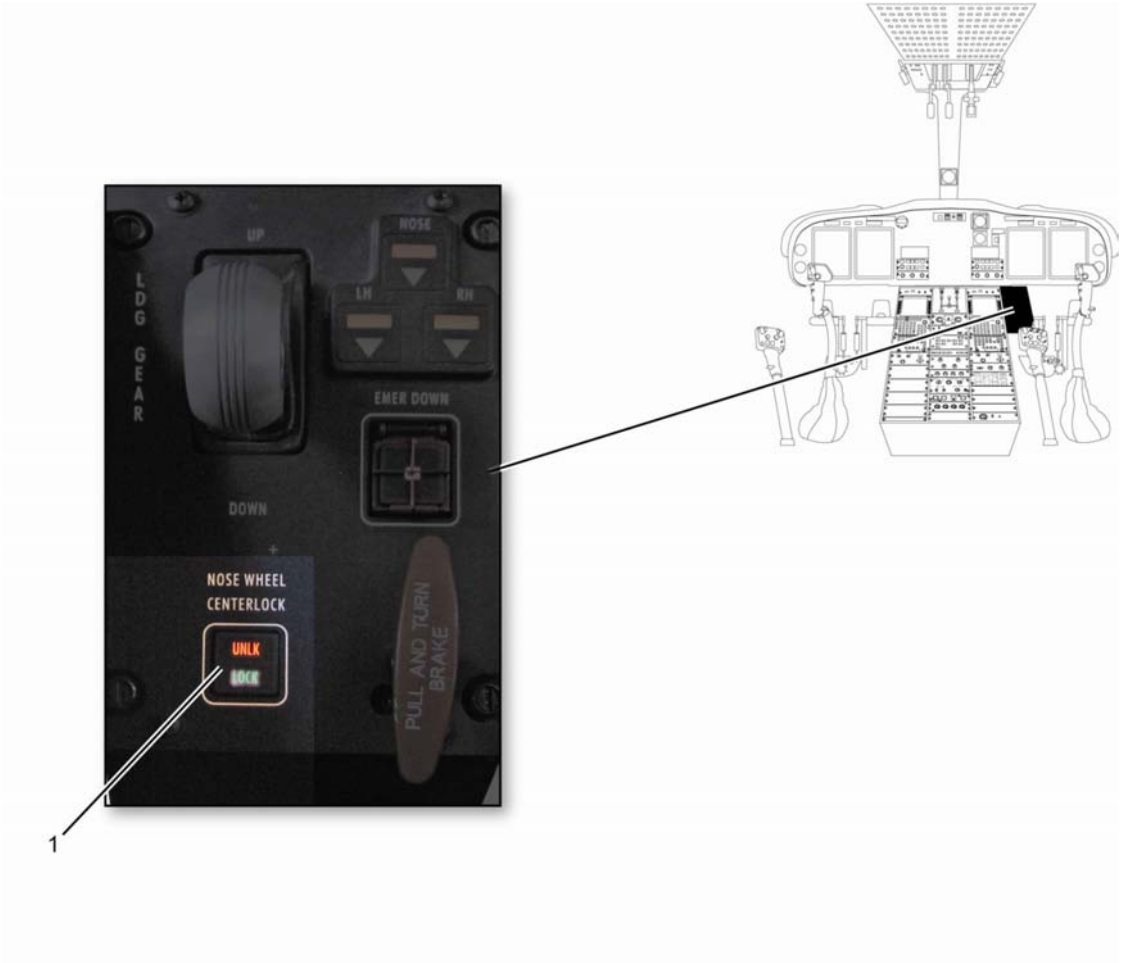
1 NOSE WHEEL CENTERLOCK pushbutton

- LOCK lighted green ..... Removes the lock of the nose wheel in the center position.
- UNLK lighted amber ..... Provides the lock of the nose wheel in the center position.

NOTE A

When the green LOCK legend is illuminated the lock is engaged. If the green legend is lighted-off and the amber UNLK legend is blinking than the lock has been armed but not engaged.
The locking logic is controlled by the WOW (Weight On Wheel) microswitch according to this rule:

- During take-off, if the NLG is unlocked, the amber light UNLK will blink during the locking phase. When the NLG is centered and locked the green light LOCK will illuminate. At this stage it is possible to retract the landing gear.
- During landing it is possible to unlock the NLG only when the helicopter will be on ground (WOW signal ON).



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NOSE WHEEL STEERING AND CENTER LOCK SYSTEM – CONTROLS AND INDICATORS

COCKPIT OPERATIONS

- Nose landing gear wheels locked in center position. Pressing the NOSE WHEEL push-button on the LGCP, the electrical motor is powered and the locking pin is removed from the hole. The green caption LOCK lights off and the amber caption UNLK lights on (the caption blinks during locking pin travel). The caution NOSE WHL UNLK is provided to the MFD. The NLG is free to swivel for taxiing or towing. Static steering can be achieved by using yaw control. Dynamic steering can be achieved by using differential braking.
- Before take-off, pressing the NOSE WHEEL push-button the actuator is powered to insert the locking pin inside the hole. The amber caption UNLK lights off and the green caption LOCK lights on. If the hole is not aligned with the locking pin, the amber caption UNLK keeps blinking until they are aligned by steering the helicopter using yaw pedals (alternatively the helicopter can be lifted off such that the centering assembly will align the nose landing gear). The caution NOSE WHL UNLK on the MFD disappears.
- If the nose wheels are not centered and the helicopter lifts-off, the WOW microswitches automatically provide the electrical power to the center lock actuator to engage the locking pin. When the landing gears are retracted the green caption LOCK lights off.
- When the helicopter is in flight and the NOSE WHEEL remains unlocked, the caution NOSE WHL UNLK is illuminated and the retraction of the landing gear system is inhibited.

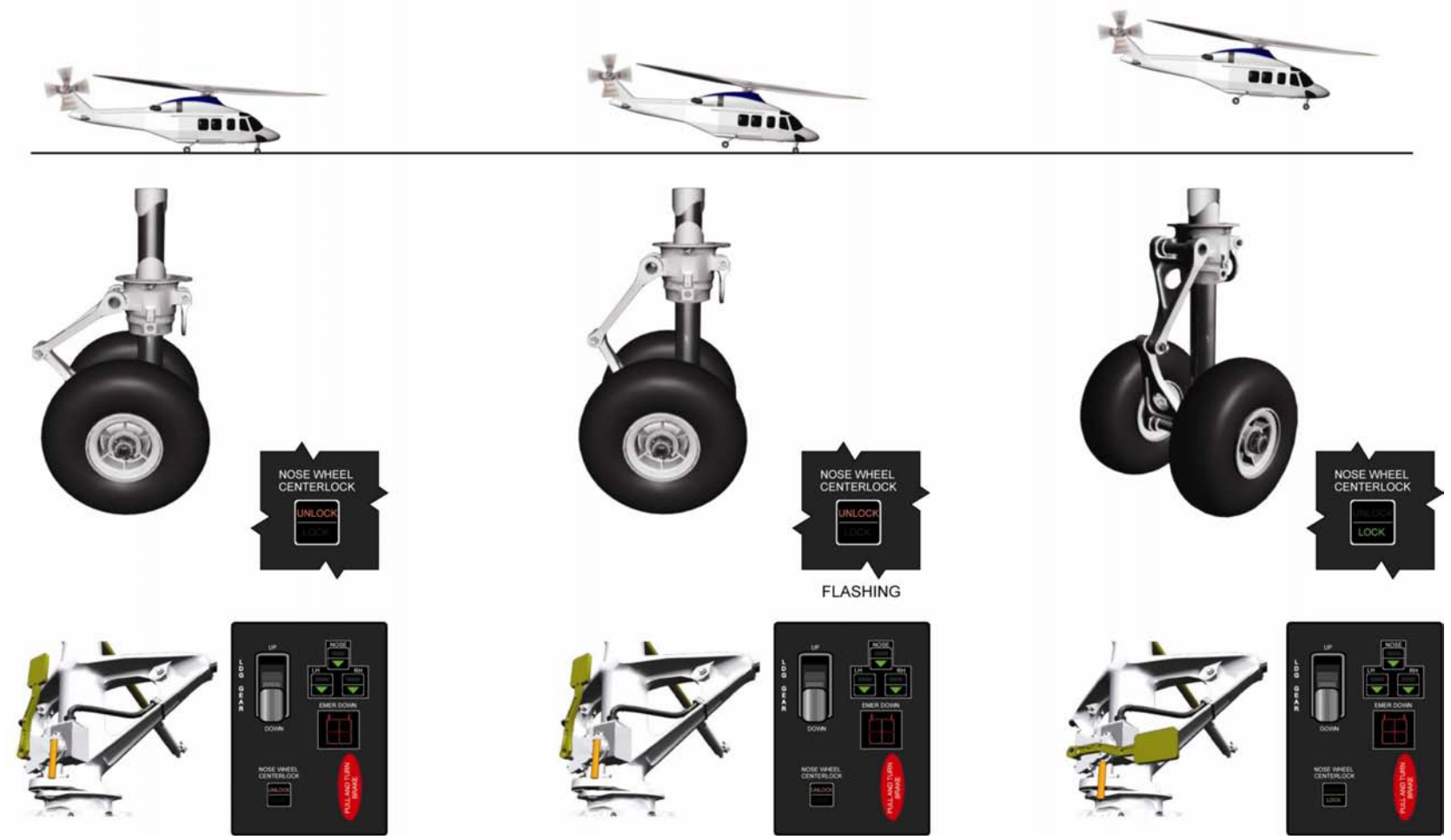
NOTE B

- Pilot can take-off with NLG unlocked.
- A mechanical system (CAM) provides for NLG centering.
- When airborne (Weight Off Wheels) the centerlock actuator automatically moves the locking pin into the flange hole.

MANUAL OPERATIONS

It is possible to manually engage/disengage the actuator locking pin through a yellow lever located in front of the assembly

- When the lever is horizontal the pin is engaged
- When the lever is vertical the pin is disengaged



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NOSE WHEEL STEERING – COCKPIT OPERATIONS



UNLOCKED POSITION



LOCKED POSITION

ICN-39-A-325000-G-A0126-03335-A-01-1

NOSE WHEEL STEERING – MANUAL OPERATIONS



ICN-39-A-325000-G-A0126-03336-A-01-1

MANUAL OPERATIONS - CENTERING PIN POSITIONS

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CHAPTER
33
LIGHTS

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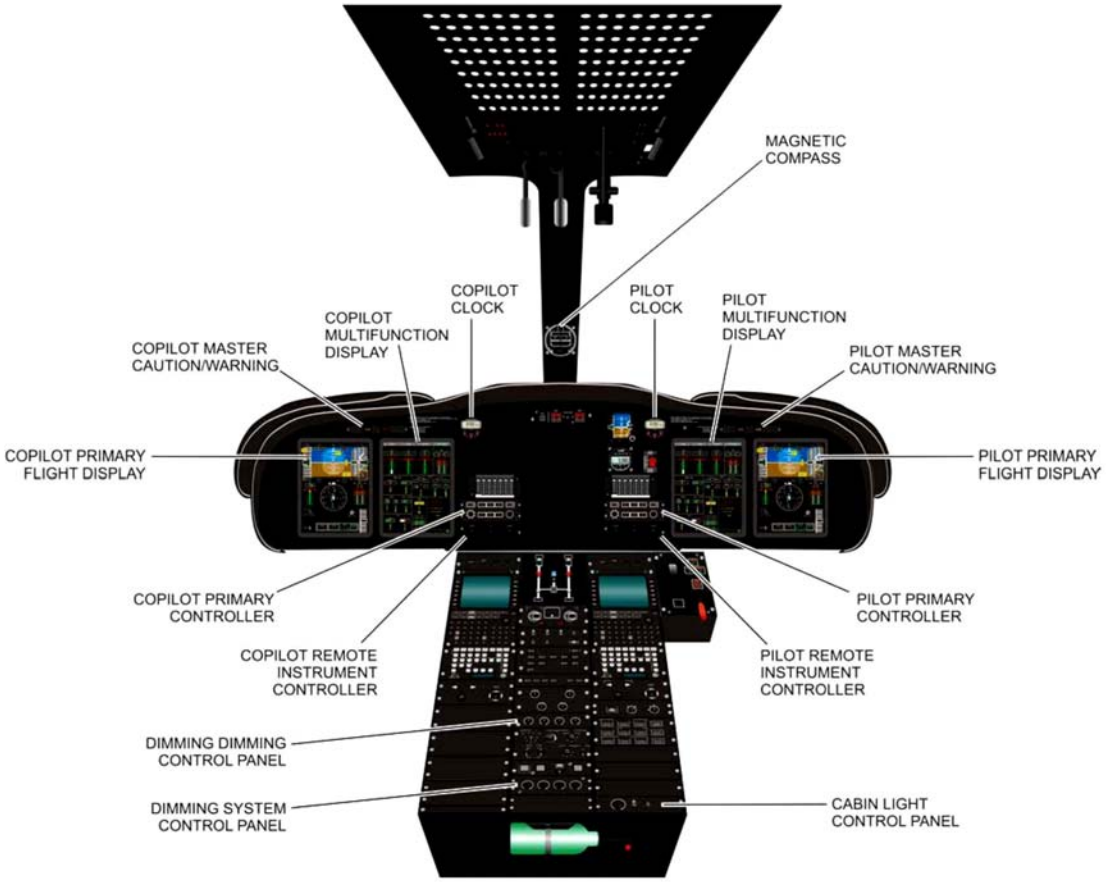
SECTION 10 - FLIGHT COMPARTMENT

DESCRIPTION

The flight compartment lights comprise:

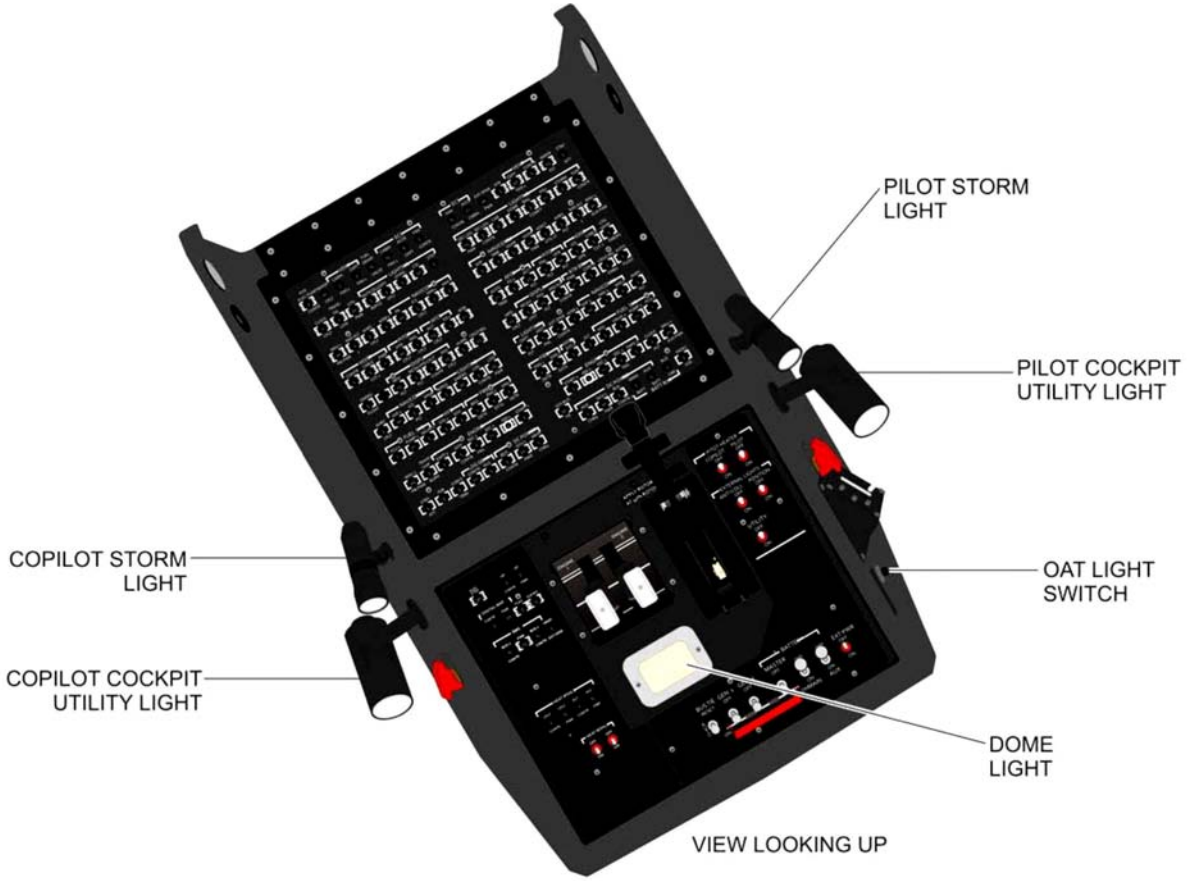
- The cockpit lights are provided with four power supplies 28V/5V variable output of 80W each for the instrument panel, the console (2 each), the overhead and the dome together with the Dimming System Control Panel.
- The illumination of the MFD, the pilot PFD and the copilot PFD displays.
- Two cockpit/utility pilot lights.
- Two pilot storm lights that prevent the disturb of the storm flashes.
- A magnetic compass and the Outer Air Temperature (OAT) lights.

The instrument panel power supply unit provides lighting to the instruments (ADI standby, FIRE control panel, etc.) and the control panels installed on the instrument panel.
Two power supply units provide lighting to the control panel (landing gear, AFCS, engine control panels, grips, etc.) installed on the interseat console.
The overhead power supply unit provides lighting to the plexiglass panel (CB panel, elect control panel, etc) installed on the overhead console.
The dome lights installed on the same overhead panel, has a large beam light as secondary lighting to illuminate the entire cockpit and the console. It is controlled by a switch and potentiometer.
The potentiometer signal is addressed to a voltage regulator, composed of a NPN transistors.
The illumination of the three displays (MFD and pilot/copilot PFD) is provided individually through the relative potentiometers installed on the Display Dimming Control Panel located on the console.
The two cockpit/utility pilot lights are installed either side of the overhead panel and each has an integral ON/OFF switch and dimmer.
The lights can be used either mounted in their bases as secondary lighting for the cockpit adjusted to the required angle or removed from their bases as hand held wander lights or as secondary lighting for the overhead panel.
Each light is equipped with two rotating cover-rings: one forward and the other aft.
Rotating the forward ring changes the colour of the emitted (white or red) light.
Rotating the aft ring changes the light intensity from maximum to the minimum.
These lights cover the whole cockpit.
On the aft face of the light is installed an ON/OFF push button for use as a signal lamp.
The magnetic compass is installed on the glazing bar at the center of the windshield has an internal light that can be dimmed.
The Outer Air Temperature light installed close to the outer air temperature probe has a dedicated light on the right side of the overhead panel.
The storm lights are installed on the overhead panel. They are very powerful halogen lights, with narrow beam light, used to compensate flashes caused by a storm.



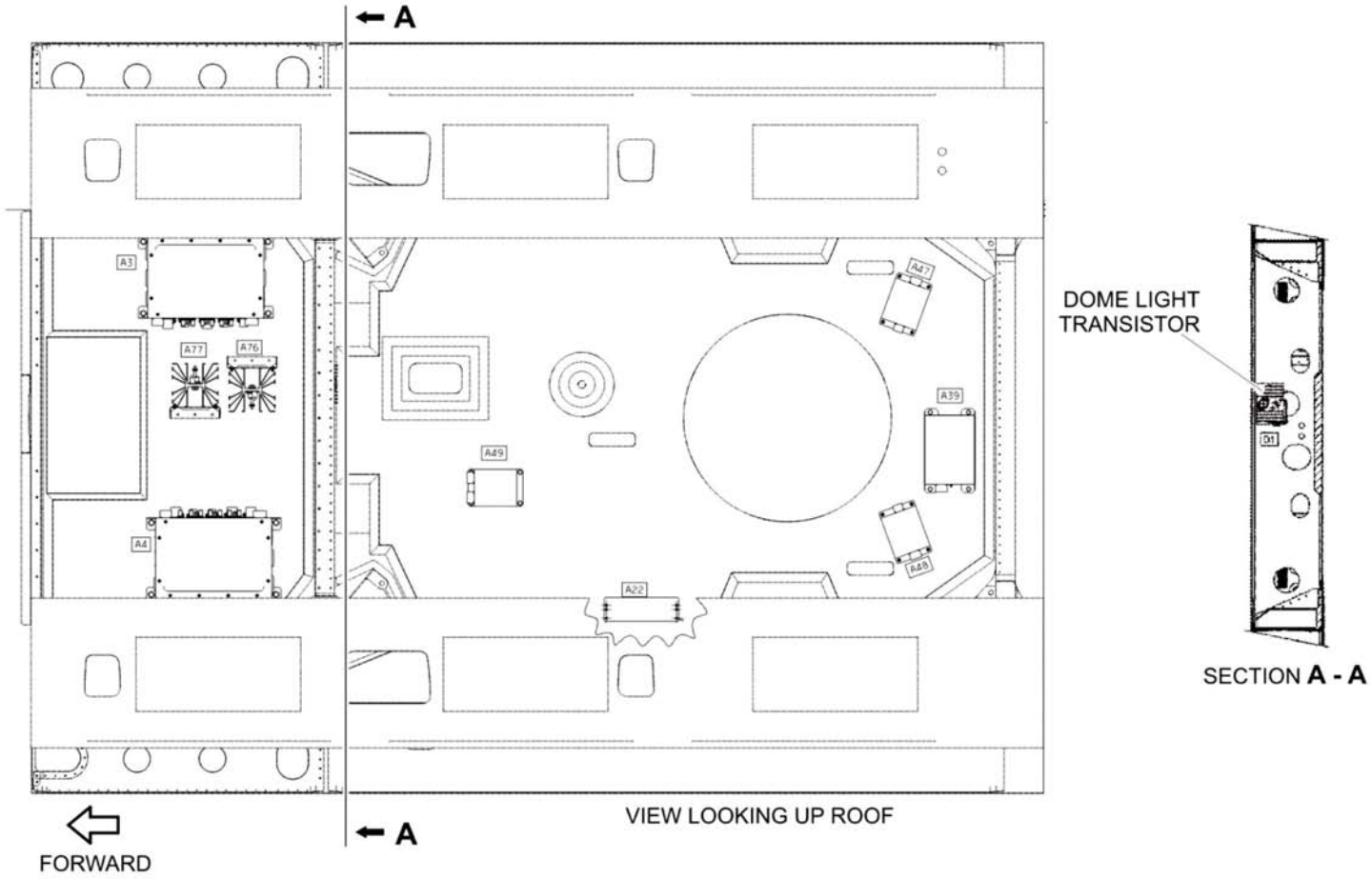
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FLIGHT COMPARTMENT LIGHT COMPONENT LOCATION (SHEET 1 OF 3)



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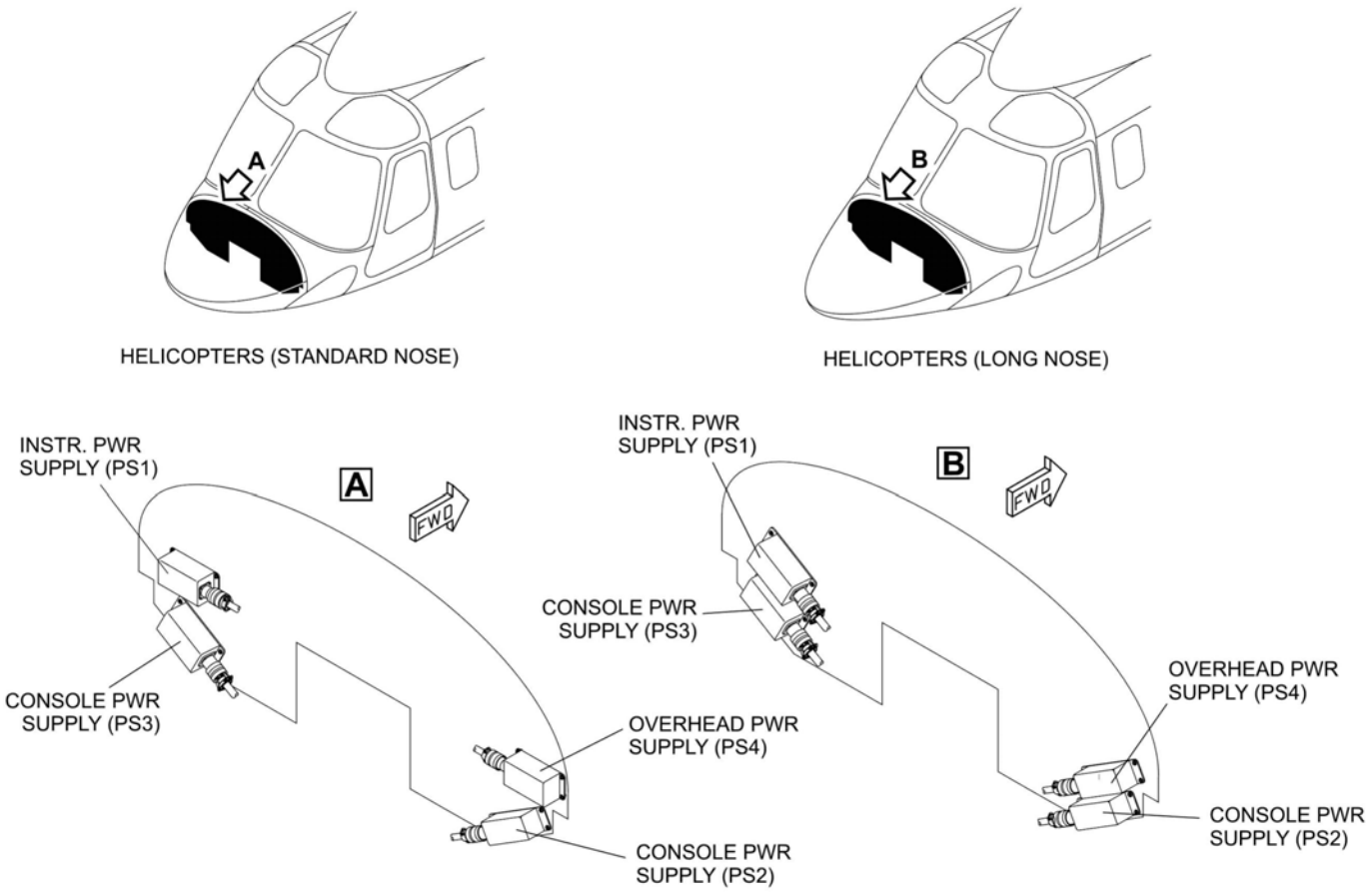
FLIGHT COMPARTMENT LIGHT COMPONENT LOCATION (SHEET 2 OF 3)



ICN-39-A-331000-G-A0126-02989-A-01-1

FLIGHT COMPARTMENT LIGHT COMPONENT LOCATION (SHEET 3 OF 3)

They create an ambient light in the cockpit in order to minimise storm flashes to the pilot.
Their position is normally adjustable in order to improve their function.
The electrical power required is 20 W but due to the inside reflector, the light intensity emitted is greater than the light emitted by the dome light.



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CABIN LIGHTS POWER SUPPLY LOCATION

OPERATION

The Dimming System Control Panel installed in the right console (pilot side), includes four switches/potentiometers labelled “INSTR”, “CONSOLE”, “OVERHEAD” and “DOME”.
The pilots switch on and control the dimming of the Instrument, Console Overhead and Dome by moving the potentiometers clockwise from maximum to minimum.
The Display Dimming Control Panel installed in the right console (pilot side), includes three potentiometers labelled “CPLT PFD”, MFD and “PLT PFD”.
The pilots control the dimming by moving the potentiometers clockwise from minimum to maximum.
The beam of the pilots cockpit/utility lights can be dimmed, controlled between spot and flood and can be white or red.
The magnetic compass dimming light is controlled by the “INSTR” potentiometer on the dimming system control panel.
The storm lights are controlled via a dedicated “STORM” switch installed on the Cabin Light Control Panel.
The outer air temperature probe has a dedicated light that can be lit, but not dimmed, when the OAT switch is kept pressed.

CONTROLS AND INDICATORS

The Flight Compartment lights have the controls housed on the Dimming System Control Panel and the Display Dimming Control Panel.
The two cockpit/utility pilot lights have the controls on the light itself.
The OAT light has the control on the overhead panel.
The magnetic compass light is under the control of the INSTR potentiometer.

DIMMING SYSTEM CONTROL PANEL

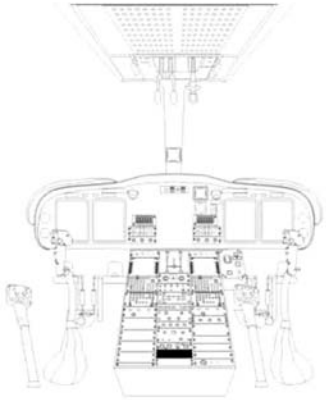
The Dimming System Control Panel provides the following functions:

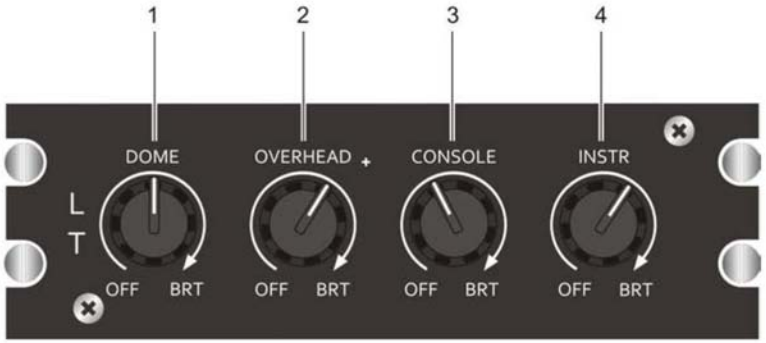
1 DOME knob: ..... rotated clockwise turns on the switch and adjusts the dome light intensity from the minimum to the maximum.

2 OVERHEAD knob: ..... rotated clockwise turns on the switch and adjust the light intensity of the panels/plexiglass from the minimum to the maximum.

3 CONSOLE knob: ..... rotated clockwise turns on the switch and adjusts the light intensity of the panels/plexiglass from the minimum to the maximum.

4 INSTR knob: ..... rotated clockwise turns on the switch and adjust the light intensity of the panels from the minimum to the maximum.





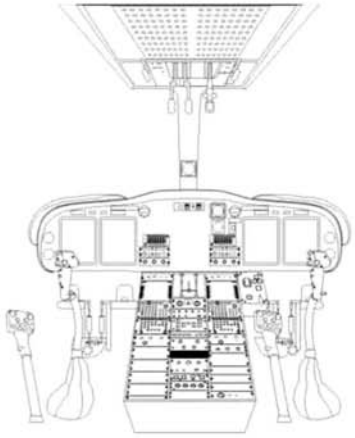
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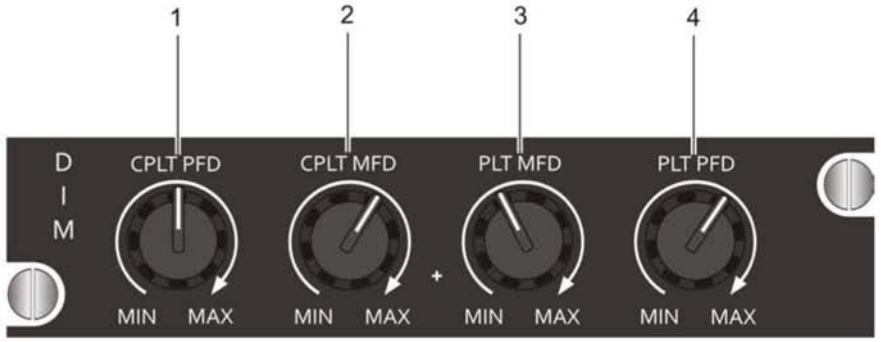
DIMMING SYSTEM CONTROL PANEL

DISPLAY DIMMING CONTROL PANEL

The Display Dimming Control Panel provides the following functions:

1	CPLT PFD knob:	rotated clockwise adjust the brightness of the copilot PFD from the minimum to the maximum.
2	CPLT MFD knob:	rotated clockwise adjusts the brightness of the CPLT MFD from the minimum to the maximum.
3	PLT MFD knob:	rotated clockwise adjusts the brightness of the PLT MFD from the minimum to the maximum.
4	PLT PFD knob:	rotated clockwise adjust the brightness of the pilot PFD from the minimum to the maximum.





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DISPLAY DIMMING CONTROL PANEL

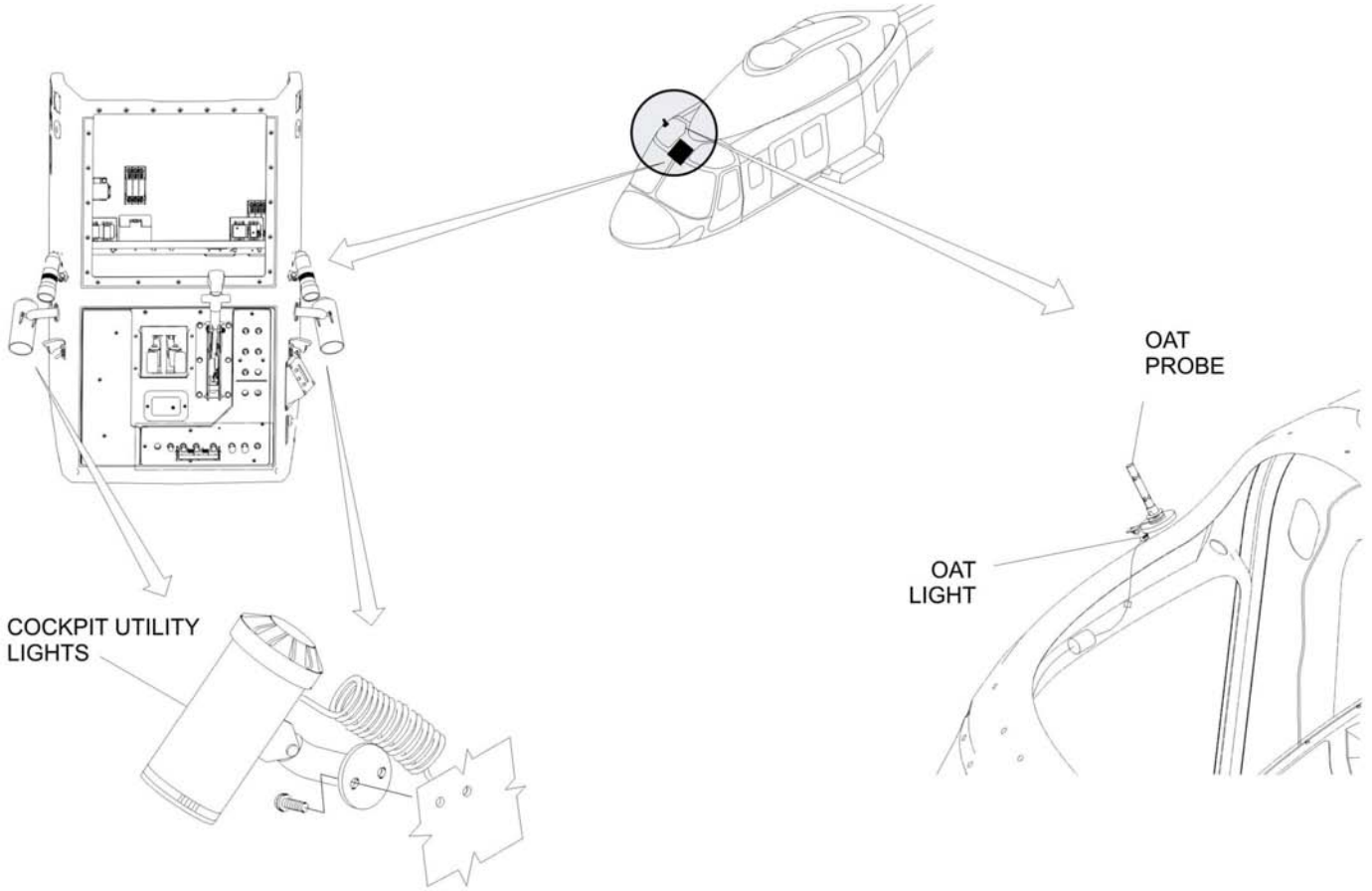
PILOT COCKPIT UTILITY LIGHTS

The Pilot Cockpit Utility Lights have the controls on the light itself and the functions are as follows:

- White push-button (on the back): ..... is a momentary illumination push-button and causes the light to put on only as long as the push-button is held pressed.
- OFF/DIM/BRIGHT knob:..... turned clockwise causes the light to put on and adjusts the light intensity.
- Colour lamp selector:..... allows to select the red or white lamp beam.
- FLOOD/SPOT ring:..... turned clockwise select the SPOT light beam type. Turned counterclockwise selects the FLOOD light beam type.

OAT LIGHT

The OAT light switch when held pressed allows the OAT light to be/put ON.



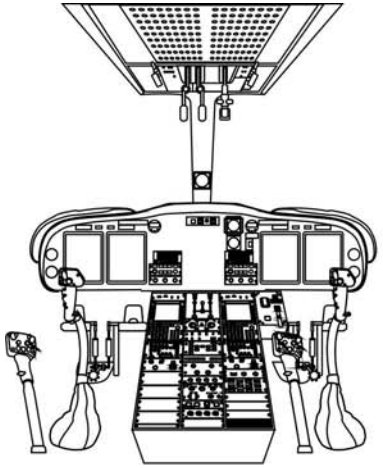
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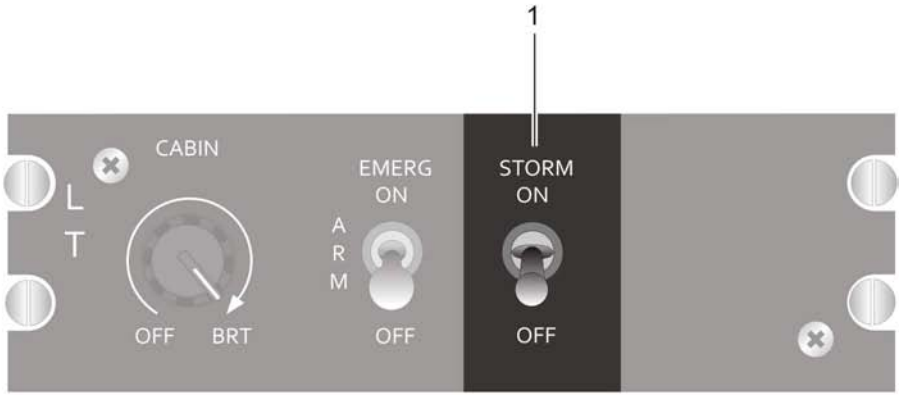
COCKPIT UTILITY LIGHTS AND OAT LIGHT CONTROL

CABIN LIGHT CONTROL PANEL

The light in the cabin comes from six fluorescent lamps that are installed on the cabin roof. The intensity of the lamp can be decreased to 35% of the maximum brightness.
The function of the lights is as follows:

1	STORM switch	allows the storm lights operational condition.





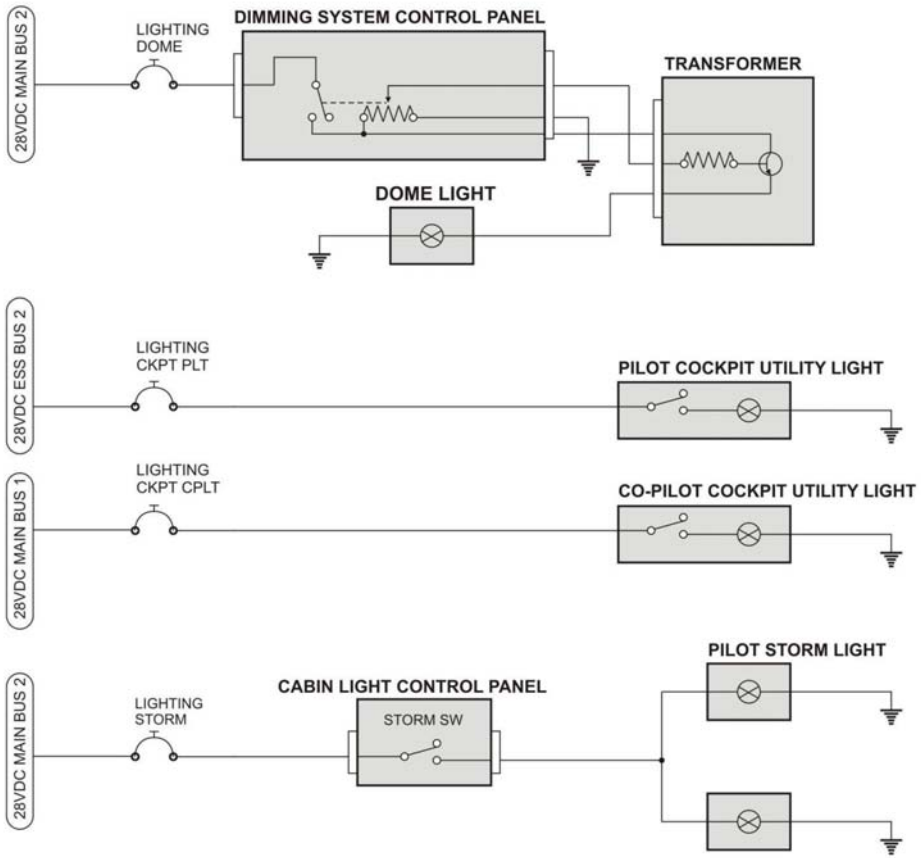
ICN-39-A-331000-G-A0126-02992-A-01-1

CABIN LIGHT CONTROL PANEL

LEADING PARTICULARS

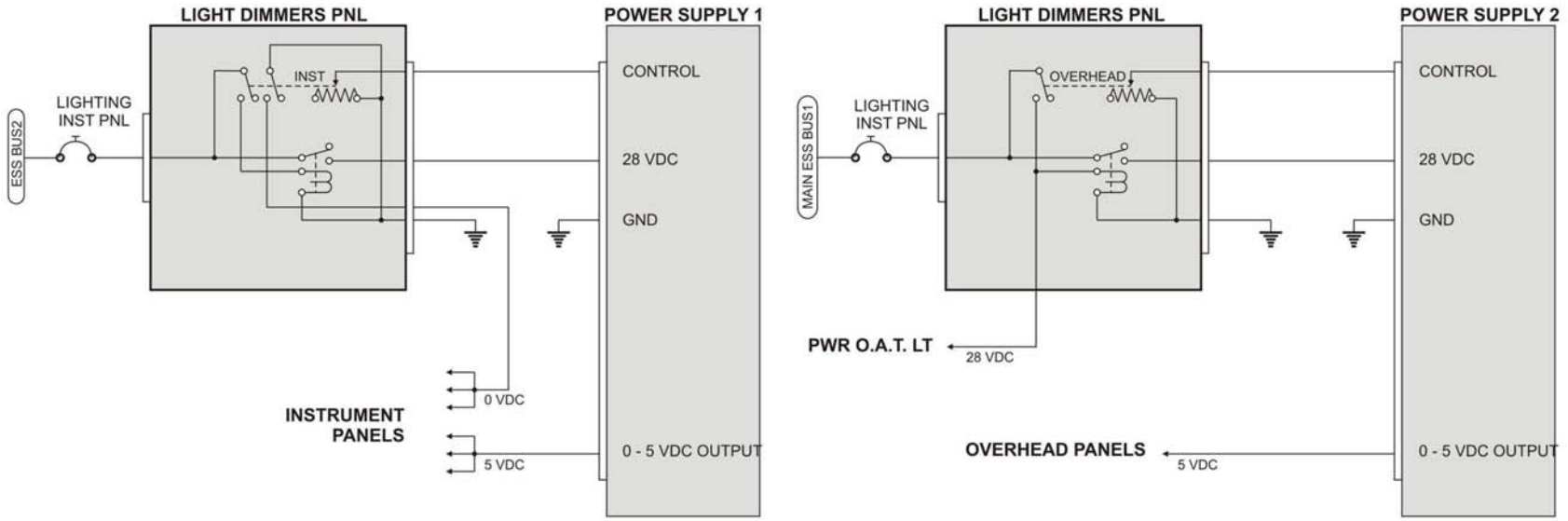
Power Supply Unit

- Instruments power supply:
  - Nominal supply voltage..... 28VDC (ESS BUS2)
  - Output supply voltage..... 0 V to 5 V
  - Output power..... 80 W
- Console power supply:
  - Nominal supply voltage..... 28VDC (MAIN BUS1)
  - Output supply voltage..... 0 V to 5 V
  - Output supply voltage..... 80 W
- Overhead power supply:
  - Nominal supply voltage..... 28VDC (MAIN BUS1)
  - Output supply voltage..... 0 V to 5 V
  - Output supply voltage..... 80 W
- Dimming system control panel:
  - Nominal supply voltage..... 28VDC (see each power supply)
- Display dimming control panel:
  - Nominal supply voltage..... 28VDC (from each display unit)
- Pilot storm light:
  - Nominal supply voltage..... 28VDC (MAIN BUS2)
  - Current consumption..... 0.70 A
  - Output power..... 20 W
- Copilot storm light:
  - Nominal supply voltage..... 28VDC (MAIN BUS2)
  - Current consumption..... 0.70 A
  - Output power..... 20 W
- Pilot cockpit utility light:
  - Nominal supply voltage..... 28VDC (ESS BUS2)
  - Current consumption..... 0.3 A
  - Output power..... 5 W
- Copilot cockpit utility light:
  - Nominal supply voltage..... 28VDC (ESS BUS2)
  - Current consumption..... 0.3 A
  - Output power..... 5 W
- Dome light:
  - Nominal supply voltage..... 28VDC (MAIN BUS2)
  - Current consumption..... 0.7 A



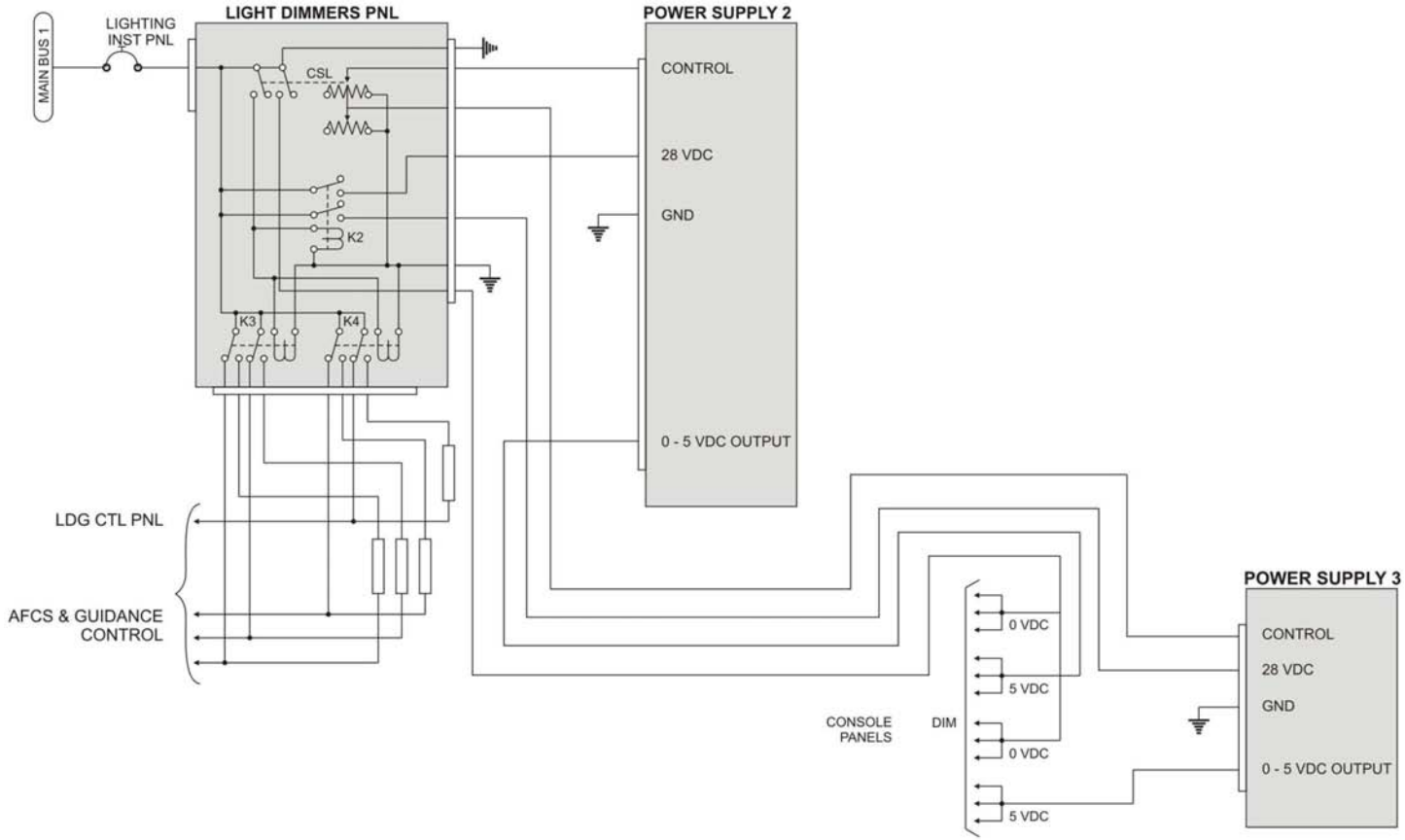
ICN-39-A-331000-G-A0126-01418-A-01-1

DOME/UTILITY/STORM LIGHT ELECTRICAL SCHEMATIC



ICN-39-A-331000-G-A0126-01419-A-01-1

INSTRUMENTS AND OVERHEAD LIGHTING ELECTRICAL SCHEMATIC



ICN-39-A-331000-G-A0126-01420-A-01-1

CONSOLE LIGHTING ELECTRICAL SCHEMATIC

SECTION 20 - PASSENGER COMPARTMENT

DESCRIPTION

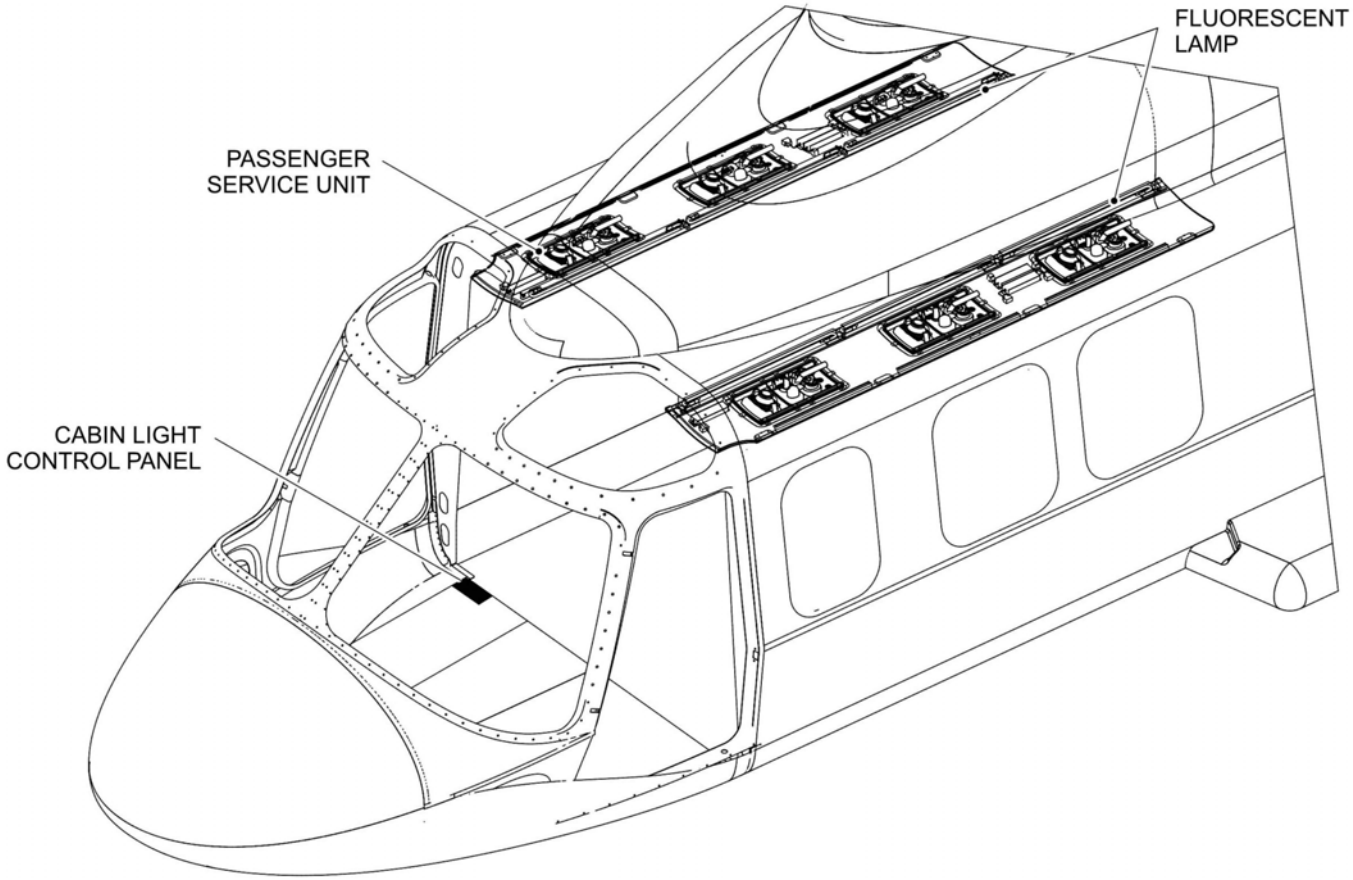
The passenger compartment lights comprise:

- six Passenger Service Units (PSU) that include two not dimmable reading lights that are set to ON through two membrane switches, a red membrane button with a steward call led and two air outlets.
- Six loudspeaker panels that include also the no smoking/fasten seat belt advisory lights.
- The cabin lighting by six fluorescent lamps.

The passenger service units together with the loudspeaker panels are installed on the cabin roof, three of them on each side.
The fluorescent lamps are installed on the cabin roof, three of them on each side.
Each lamp is powered by a dedicated inverter which contains a solid state oscillator which regulates the current through the lamp.
Each inverter is equipped with a safety switch: the output voltage is switched off in case of output open circuit or failure of the lamp.
All the inverters are connected together in order to form one Master and five Slave. The lighting adjustment circuit is connected directly to the Master inverter while its control output channel, provides the adjustment lighting to the other five inverter Slave.
In case of an open failure of the adjustment circuit, the lighting automatically is set to bright.

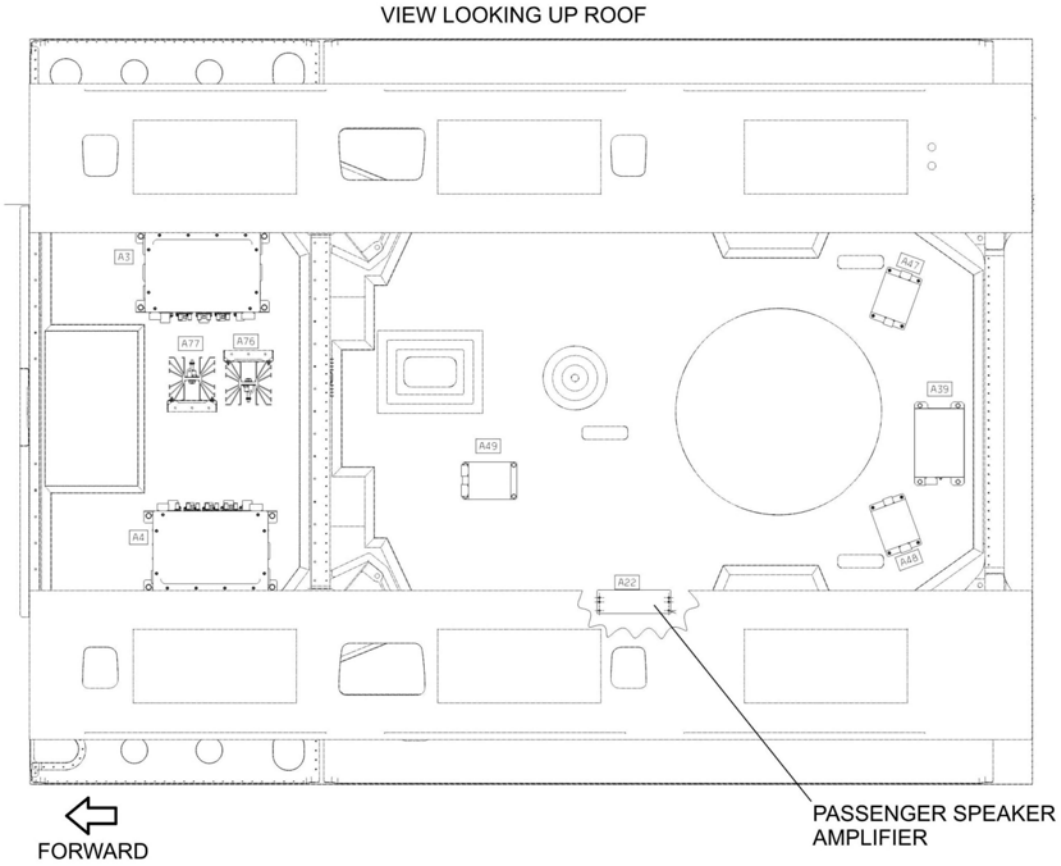
OPERATION

By means of the PSUs, the passengers can switch ON/ OFF each reading light as well as the attendant call light for board service via the membrane switches.
In order to show the passenger that the call has been activated a red colored LED lights up.
This is located between the reading lights and the air outlets.
Through the two air outlets the air supply can be adjusted appropriately.
When the two advisory lights are set to ON/OFF by pilots through the pilot/ copilot Audio panel, a chime is heard at the loudspeakers.
The fluorescent lamps are switched on through the Cabin Light Control Panel.



ICN-39-A-332000-G-A0126-01421-A-01-1

PASSENGER COMPARTMENTS COMPONENT LOCATION (SHEET 1 OF 2)



ICN-39-A-332000-G-A0126-01422-A-01-1

PASSENGER COMPARTMENTS COMPONENT LOCATION (SHEET 2 OF 2)

PSU - CONTROLS AND INDICATORS

The passenger service units controls are housed on the cabin roof.
The advisory lights are controlled by the Audio panels while the fluorescent lamp the controls are housed on the console.
The PSU provides the following functions:

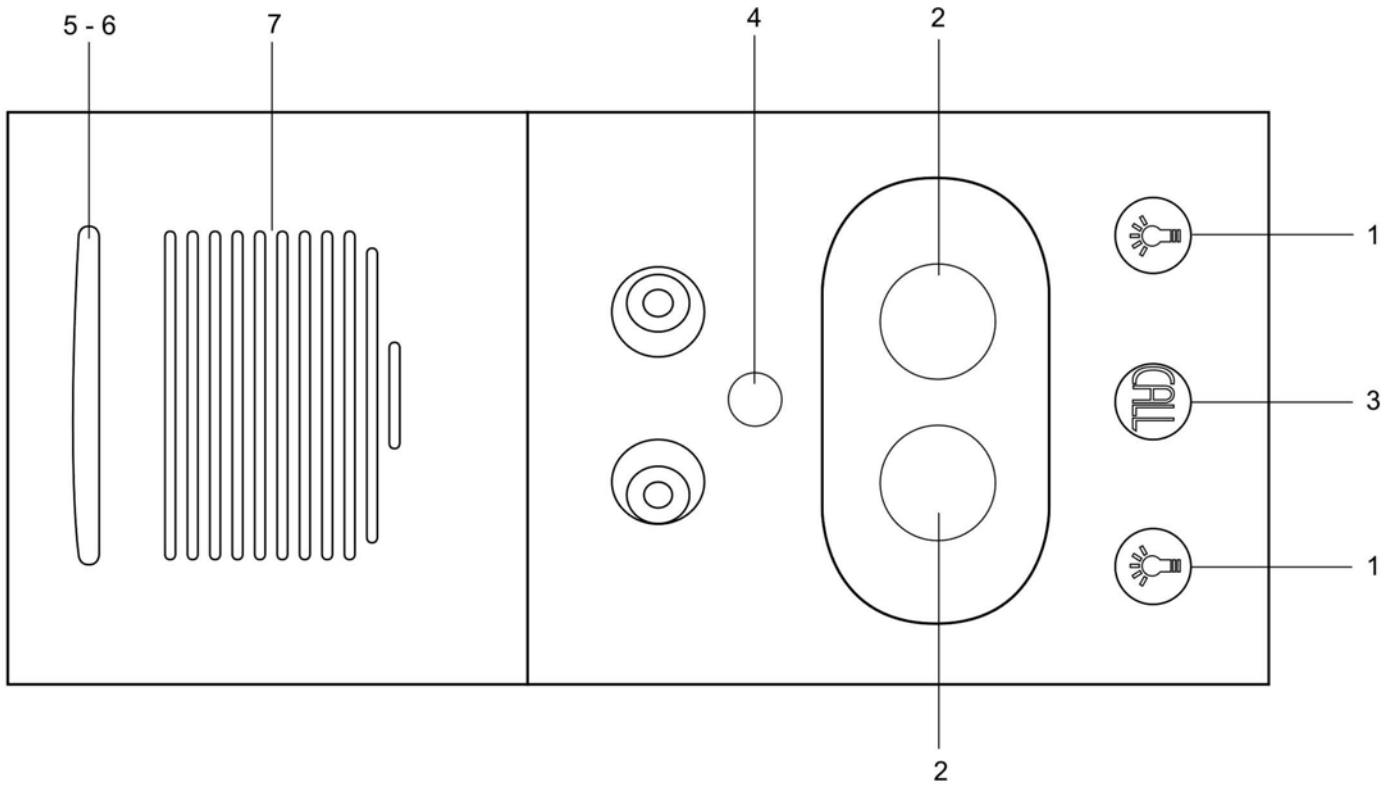
1	READING LT MEMBRANE SWITCH:	when pressed the relative reading light is lit
2	READING LIGHTS	are ON when the relative switch is pressed
3	STEWARD CALL MEMBRANE SWITCH (red):	when pressed the steward call white LED is lit and becomes red.

On the loudspeaker panel the functions are:

4	STEWARD CALL WHITE/RED LED:	becomes red when the Steward Call Membrane switch is pressed.

On the loudspeaker panel the functions are:

5	NO SMOKING advisory light: ……	is lit when the NO SMK push-button pressed on the pilot/ copilot audio panel.
6	FASTEN SEAT BELT advisory light:	is lit when the SEAT BLT push-button pressed on the pilot/copilot audio panel.
7	Loudspeaker:	allows the chime to be heard when one of the two above advisory light comes ON or OFF.



ICN-39-A-332000-G-A0126-01423-A-01-1

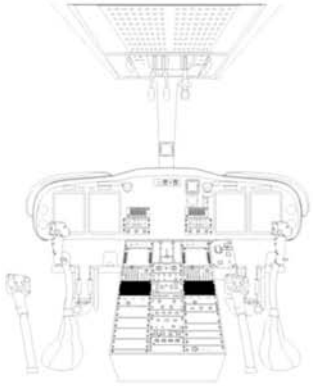
PSU-CONTROLS AND INDICATORS

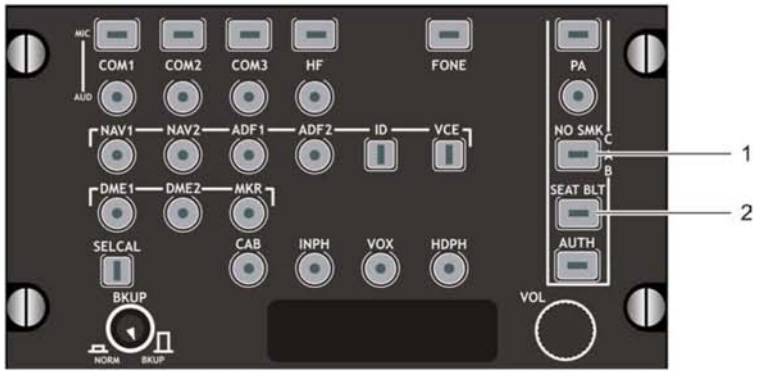
AUDIO CONTROL PANEL – CONTROL AND INDICATORS

On the Audio Control Panel there are the NO SMK and the SEAT BLT push-buttons. These two push-buttons provide the following functions:

1 NO SMK: ..... when pressed the NO SMOKING lights on the PSUs are lit. The green bar inside this push-button is then lit.

2 SEAT BLT: ..... when pressed the SEAT BEALTS lights on the PSUs are lit. The green bar inside this push-button is then lit.





ICN-39-A-332000-G-A0126-02993-A-01-1

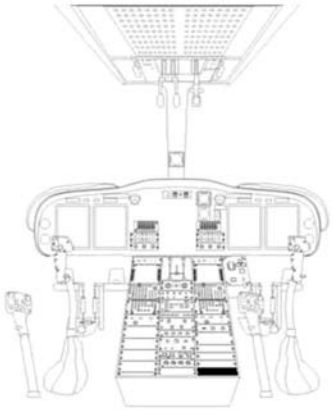
AUDIO CONTROL PANEL – CONTROLS AND INDICATORS

LIGHT CONTROL PANEL – CONTROLS AND INDICATORS

The light control panel provides the following function:

1 CABIN knob:

- OFF/BRT: ..... rotated clockwise turns on the switch and adjusts the light intensity of the cabin from the minimum to the maximum.



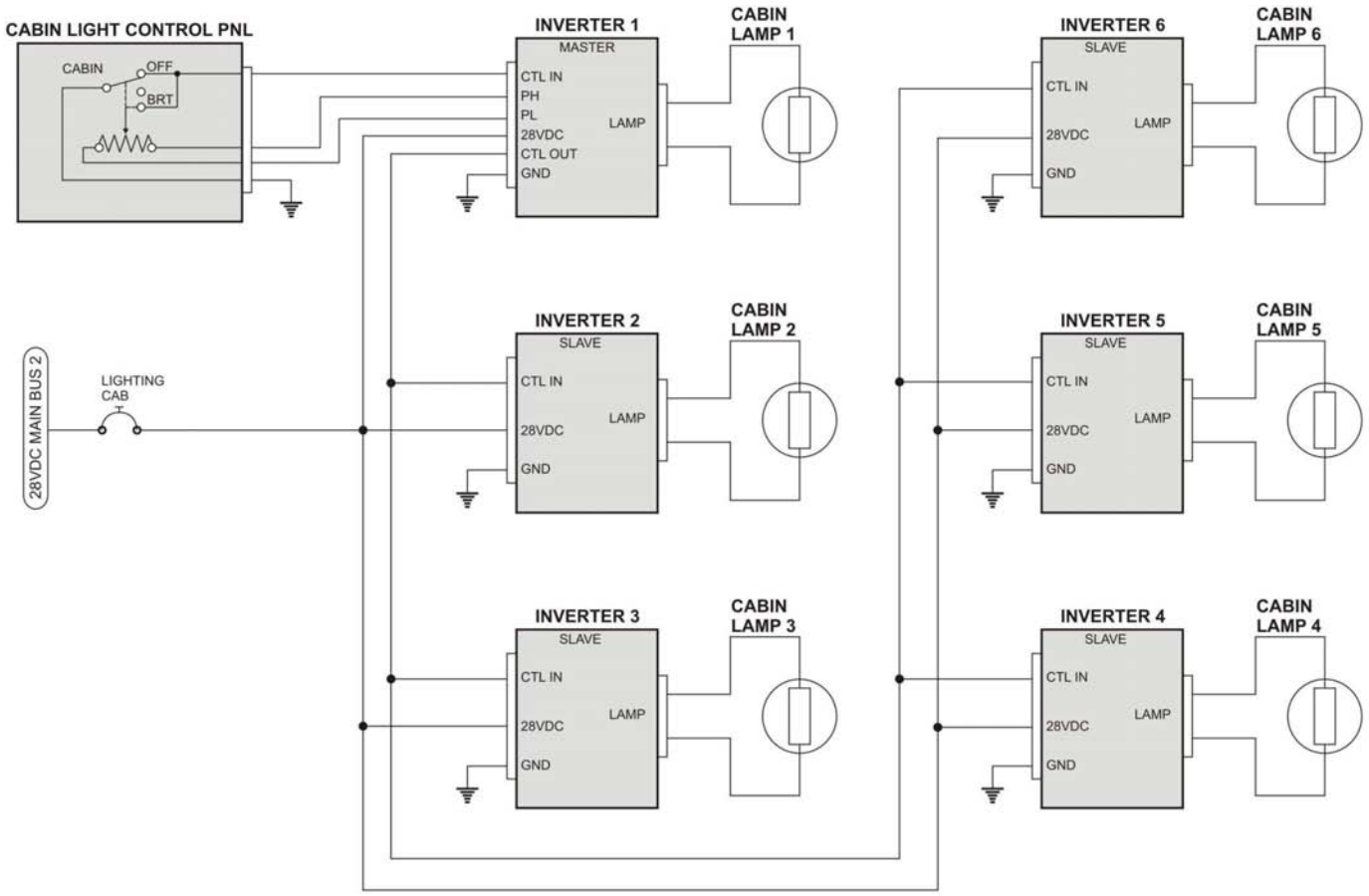


ICN-39-A-332000-G-A0126-02994-A-01-1

LIGHT CONTROL PANEL – CONTROLS AND INDICATORS

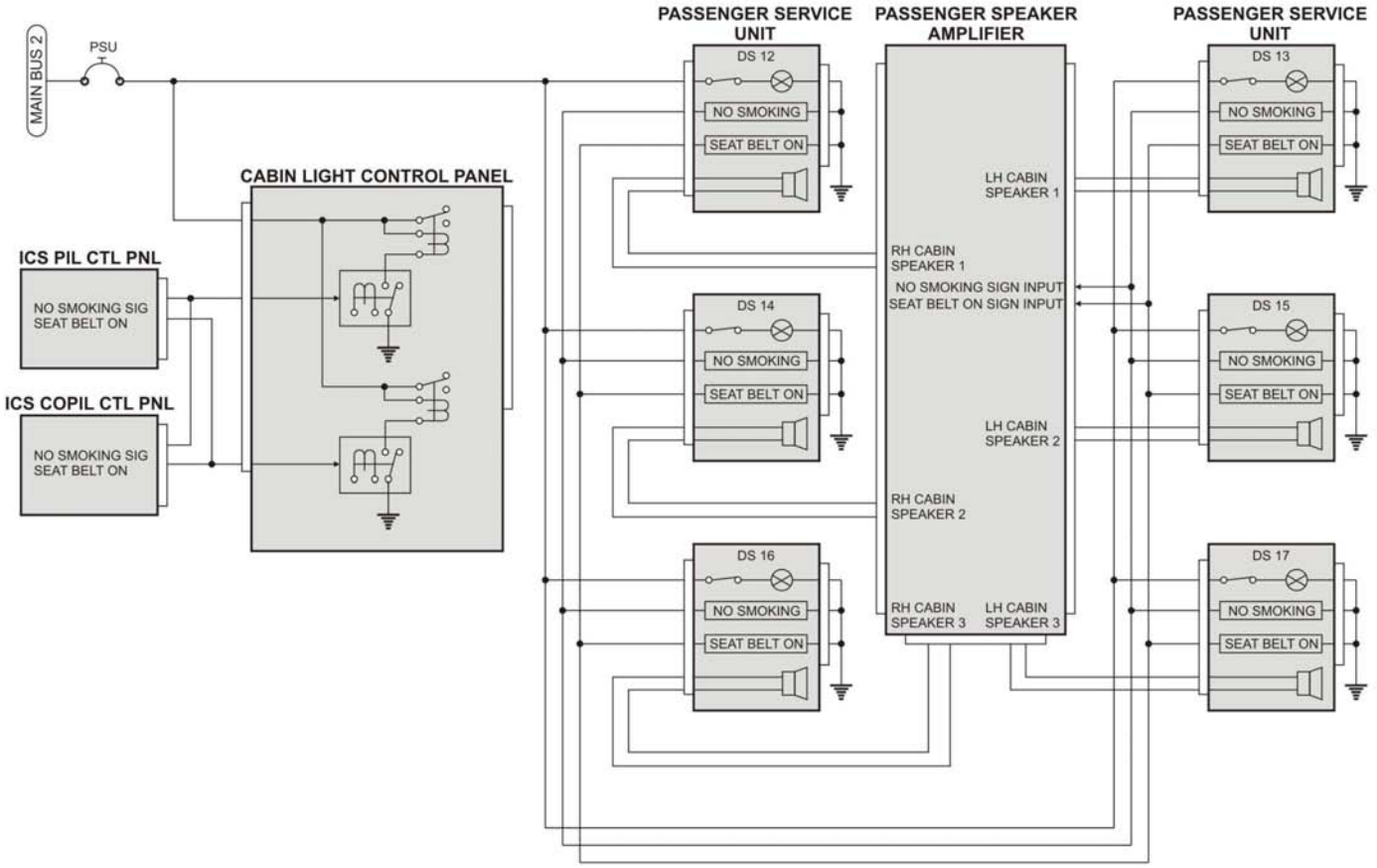
LEADING PARTICULARS

- Inverter:
  - Supply voltage..... 28 VDC (MAIN BUS 2)
  - Current consumption..... 0.85 A
- Passenger service unit:
  - Supply voltage..... 28 VDC (MAIN BUS 2)



ICN-39-A-332000-G-A0126-01426-A-01-1

CABIN LIGHTS ELECTRICAL SCHEMATIC



ICN-39-A-332000-G-A0126-01427-A-01-1

PASSENGER/ ADVISORY LIGHTS ELECTRICAL SCHEMATIC

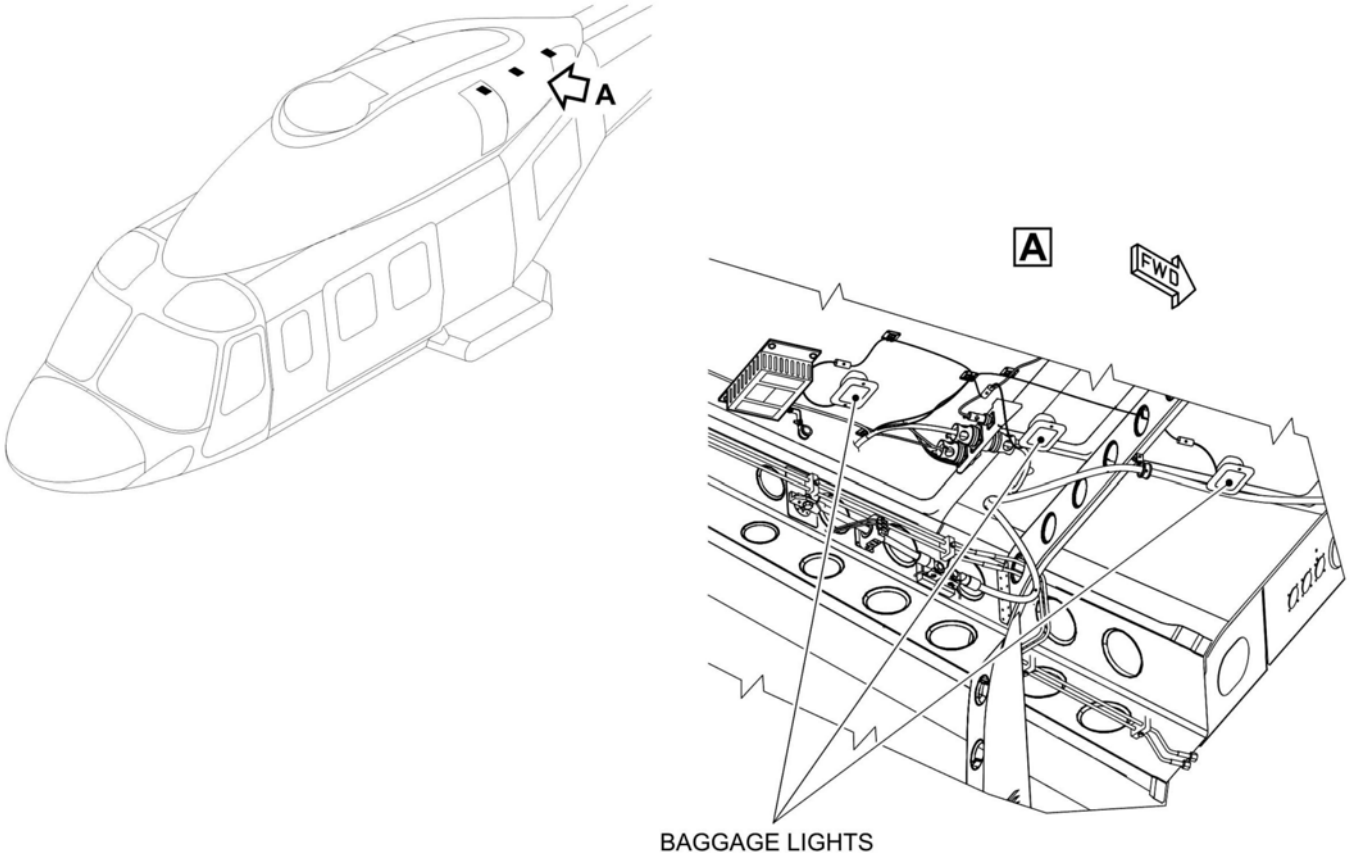
SECTION 30 - CARGO AND SERVICE COMPARTMENTS

DESCRIPTION

The cargo and service compartment lights comprise three lamps that are located on the baggage roof compartment.
They operate when one of the two baggage compartment access door is opened.

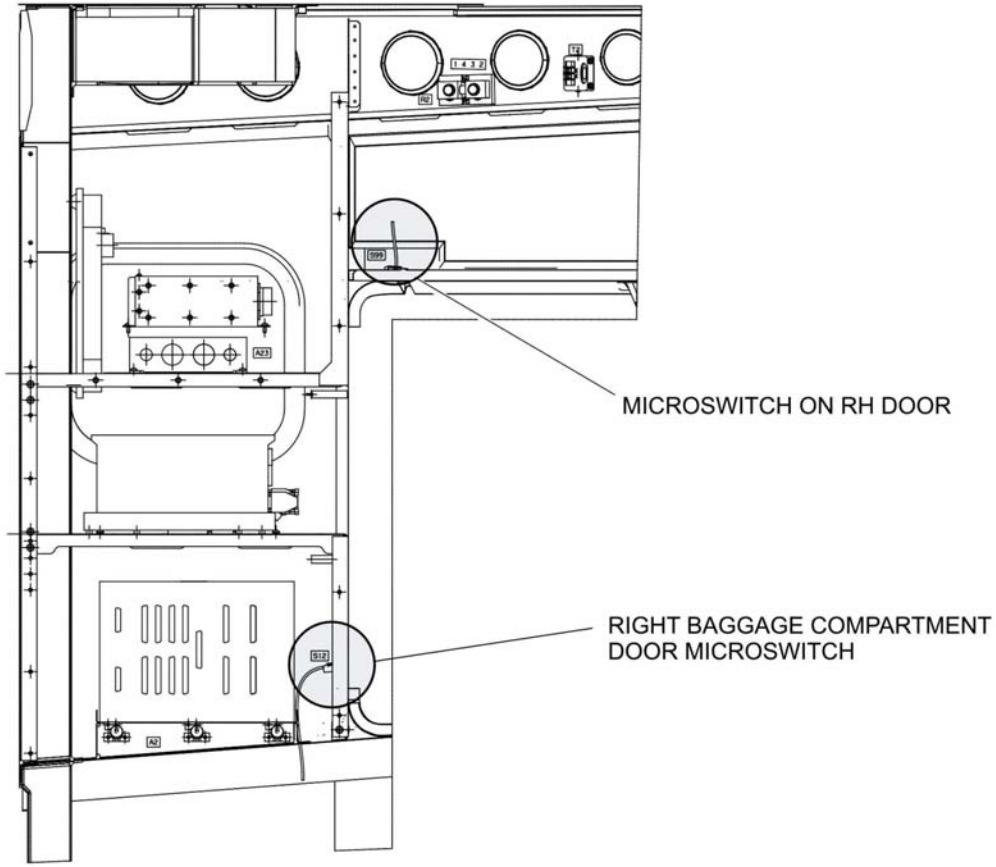
OPERATION

By means of the relative microswitches, on actuated (closed) by the right access door and the other one by the left access door, the three baggage compartment lamps light.



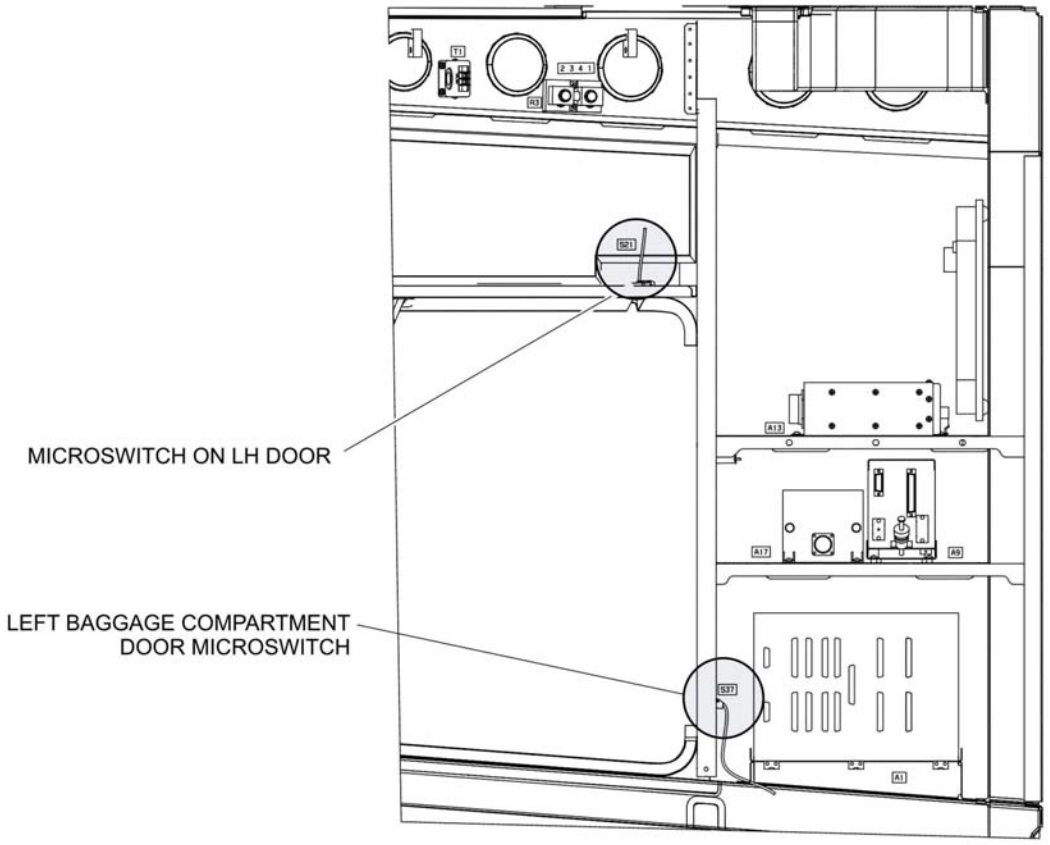
ICN-39-A-333000-G-A0126-01428-A-01-1

CARGO AND SERVICE COMPARTMENT LIGHT COMPONENT LOCATION (SHEET 1 OF 3)



ICN-39-A-333000-G-A0126-01429-A-01-1

CARGO AND SERVICE COMPARTMENT LIGHT COMPONENT LOCATION (SHEET 2 OF 3)



ICN-39-A-333000-G-A0126-01430-A-01-1

CARGO AND SERVICE COMPARTMENT LIGHT COMPONENT LOCATION (SHEET 3 OF 3)

CONTROLS AND INDICATORS

Baggage compartment light controls:

- Baggage compartment access doors closed:
  - Lights OFF.
- Either baggage compartment access door open:
  - Lights ON.

CAPTIONS

The caution provided in the CAS message window is as follows:

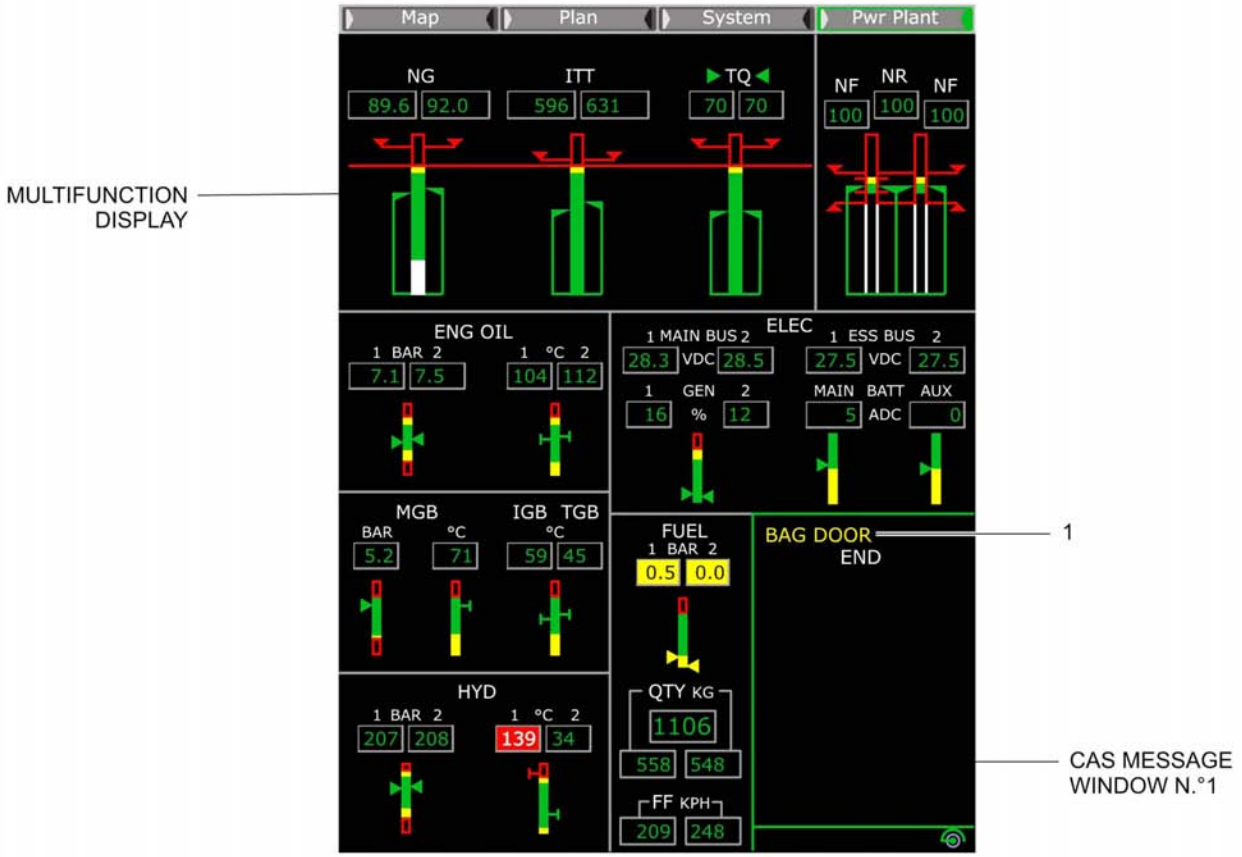
1 BAG DOOR: ..... when either baggage compartment access door is open.

NOTE A

Further details about the two microswitches that are involved with the above caution are reported on Chapter 52 Doors – Section 70 Door warning.

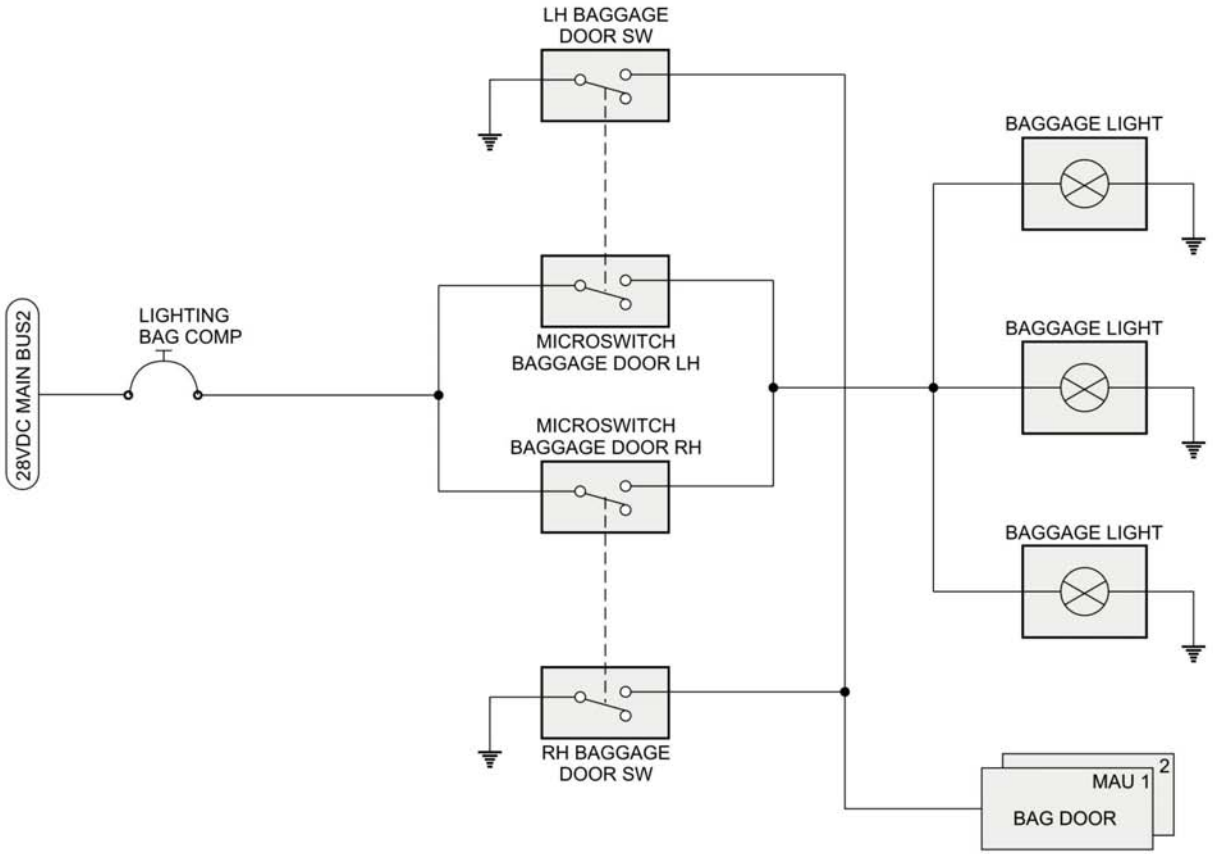
LEADING PARTICULARS

- Baggage lamp (3 each):
  - Nominal supply voltage..... 28 VDC (MAIN BUS2)
  - Current consumption..... 0.7 A each



ICN-39-A-333000-G-A0126-01431-A-01-1

CARGO AND SERVICE COMPARTMENT LIGHTS – CAPTIONS



ICN-39-A-333000-G-A0126-01432-A-01-1

BAGGAGE LIGHT/ DOOR ELECTRICAL SCHEMATIC

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SECTION 40 - EXTERIOR

DESCRIPTION

The external lights consist of:

- the anti-collision light
- the position lights
- the landing lights (2 each)
- secondary landing light (or landing light n. 2)

The anti-collision light is a flashing (45 ± 5 flashes per minute) red light and provides a signal to enhance the visibility of the helicopter at great distances.
The light is installed on the tail in the way to be seen in the correct angle.
The position lights provide, to the other a/c's pilots, the position and direction of flight of the helicopter.
The position lights consist of three lights of three different colours.
The rear position light (white) is installed on the top of the tail fin, the green on the aft part of the right sponson and the red on the aft part of the left sponson.
The two landing lights provide a high intensity light external source suitable for the night during landing and taxi operations.
They are installed on the right and left side sponsons.
The secondary landing light has been designed to provide a mobile and versatile high intensity light source for aerial reconnaissance at nightime by helicopter for rescue, search and surveillance operations.
It is installed under the cockpit (forward-left side) in order to minimise glare to the pilot.

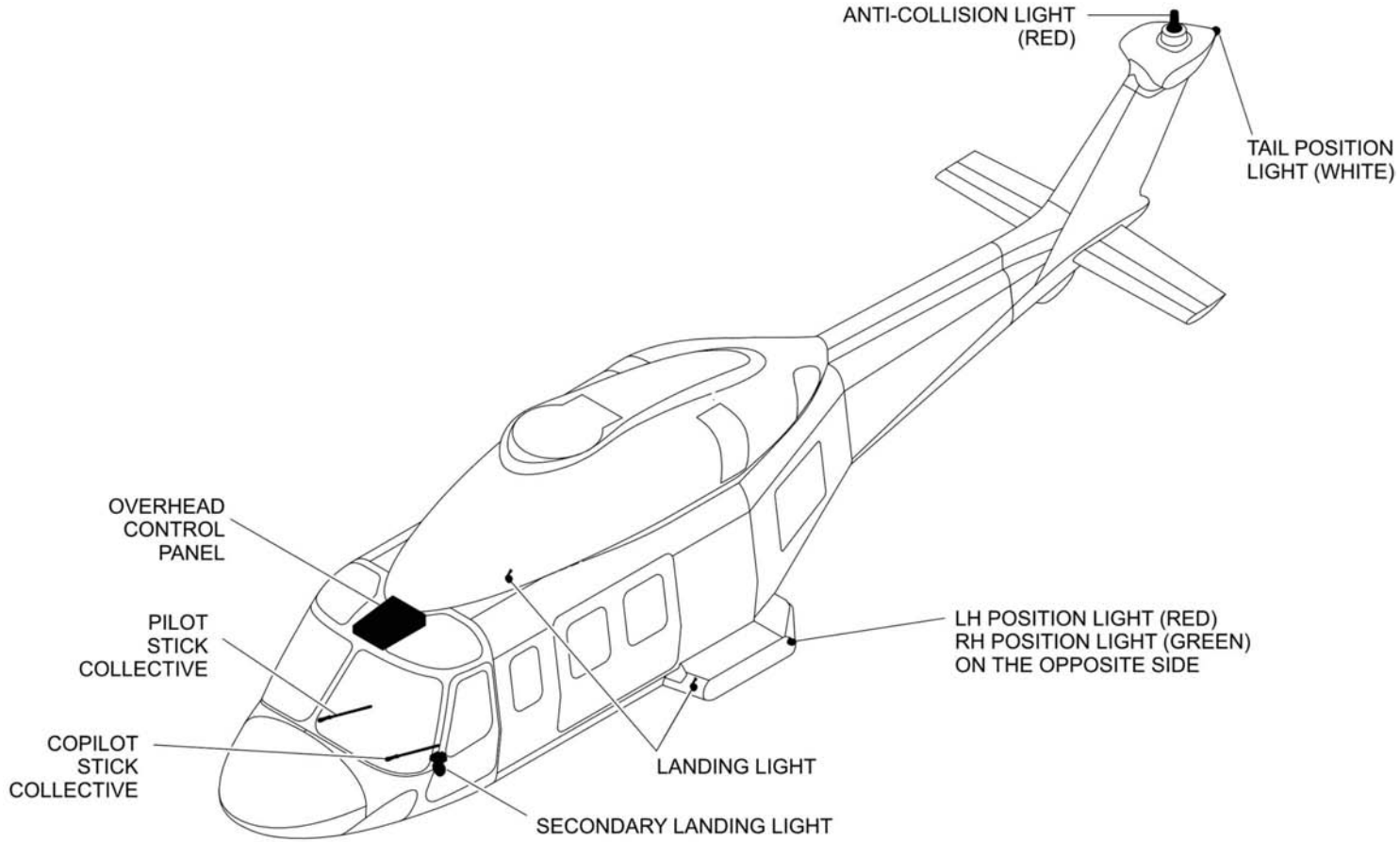
OPERATION

The anti-collision light is controlled by ANTI-COLL switch installed on the circuit breaker panel on the overhead panel. The switch can be set to OFF/ON.

The position lights are controlled by the POSITION switch installed on the circuit breaker panel on the overhead panel. The switch can be set to OFF/ ON.
The two landing lights are controlled by the LDG LT 3 position momentary toggle switch installed on the pilot/copilot collective grips. The switch is a three position type:

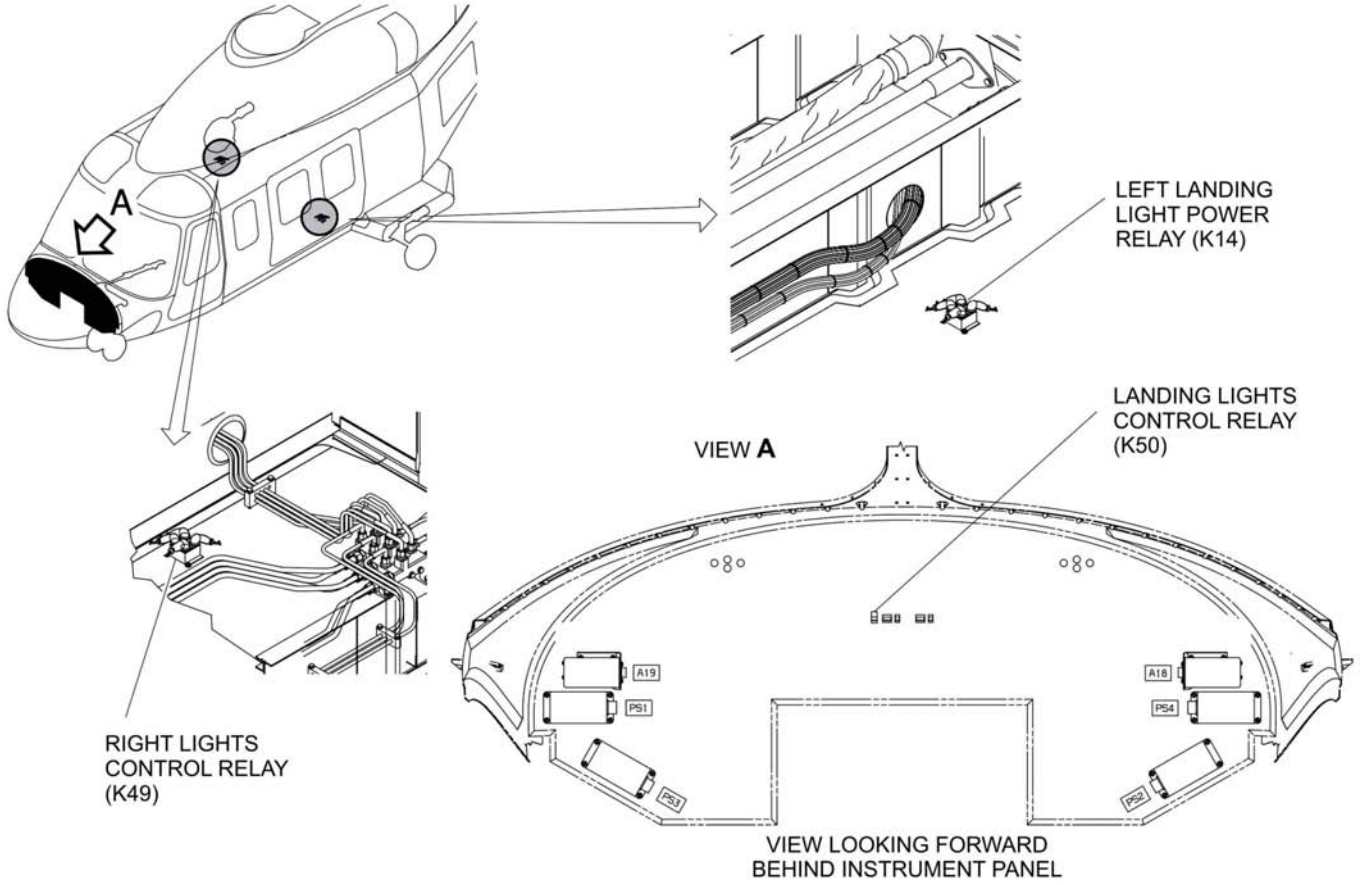
- central
- ON
- OFF

When the ON position is set, the lamps are lit and an advisory message is generated, when OFF is set the lamps extinguish and the advisory message removed.
The secondary landing light is controlled by the LDG LT2 coolie hat with four way switch and a central momentary contact installed on the pilot/copilot collective grips and can be rotated in either direction.



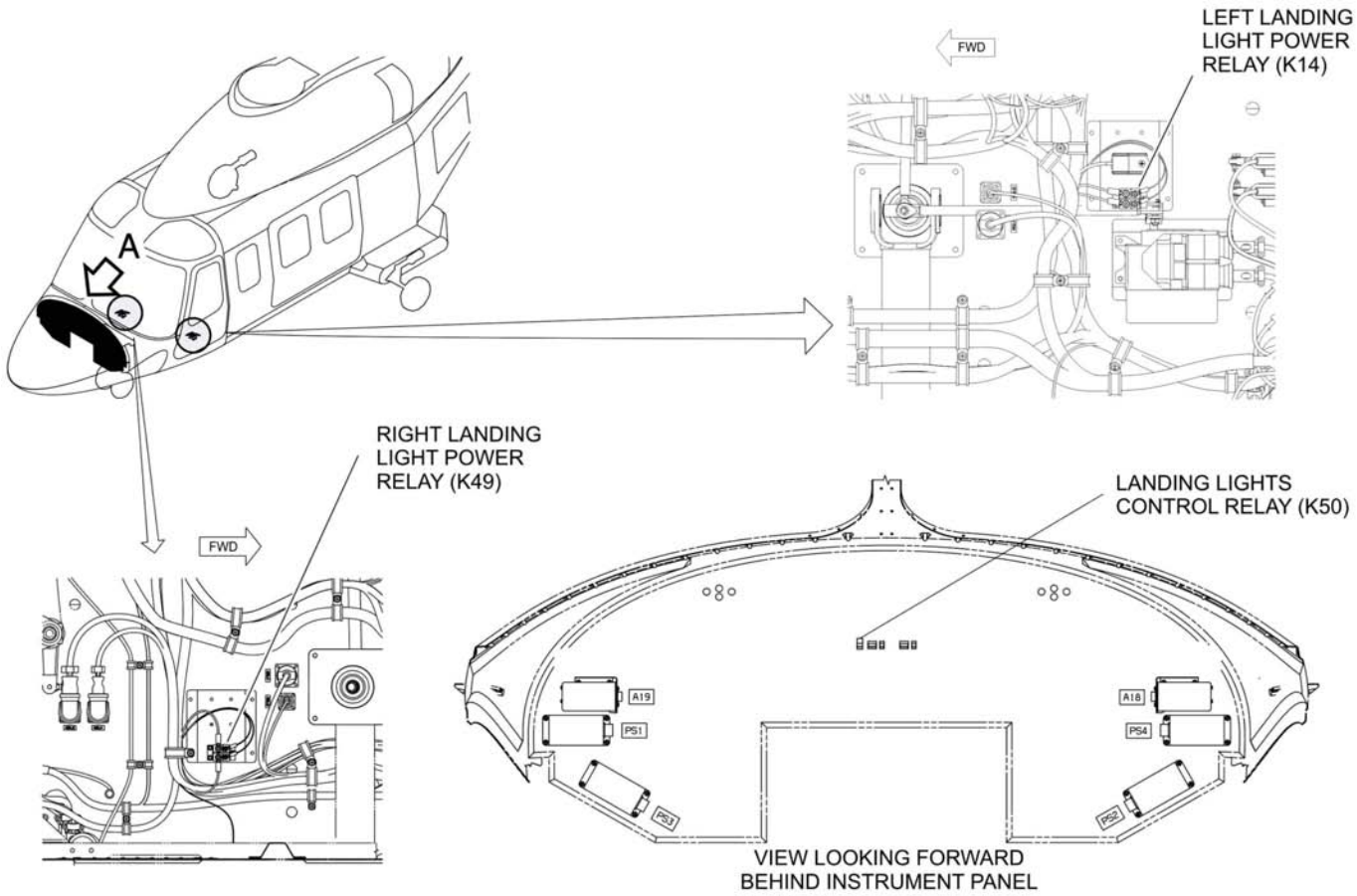
ICN-39-A-334000-G-A0126-01433-A-01-1

EXTERNAL LIGHTS INSTALLATION



ICN-39-A-334000-G-A0126-01434-A-01-1

EXTERNAL LIGHTS COMPONENTS LOCATION (STANDARD NOSE)



ICN-39-A-334000-G-A0126-01435-A-01-1

EXTERNAL LIGHTS COMPONENTS LOCATION (LONG NOSE)

EXTERNAL LIGHTS – CONTROLS

The external lights have the controls housed on the overhead panel and on the pilot/copilot collective grips.
On the overhead panel the following functions are provided:

1 ANTI COLL switch:
- OFF/ON: ..... allows the anti-collision light operational condition.

2 POSITION switch:
- OFF/ON: ..... allows the position lights operational condition.

On the pilot/copilot collective sticks the following functions are provided:

3 LDG LT spring loaded momentary toggle switch:
- Central ..... inoperative
- ON (spring loaded) ..... allows to light on the two lamps
- OFF (spring loaded) ..... allows to light off the two lamps

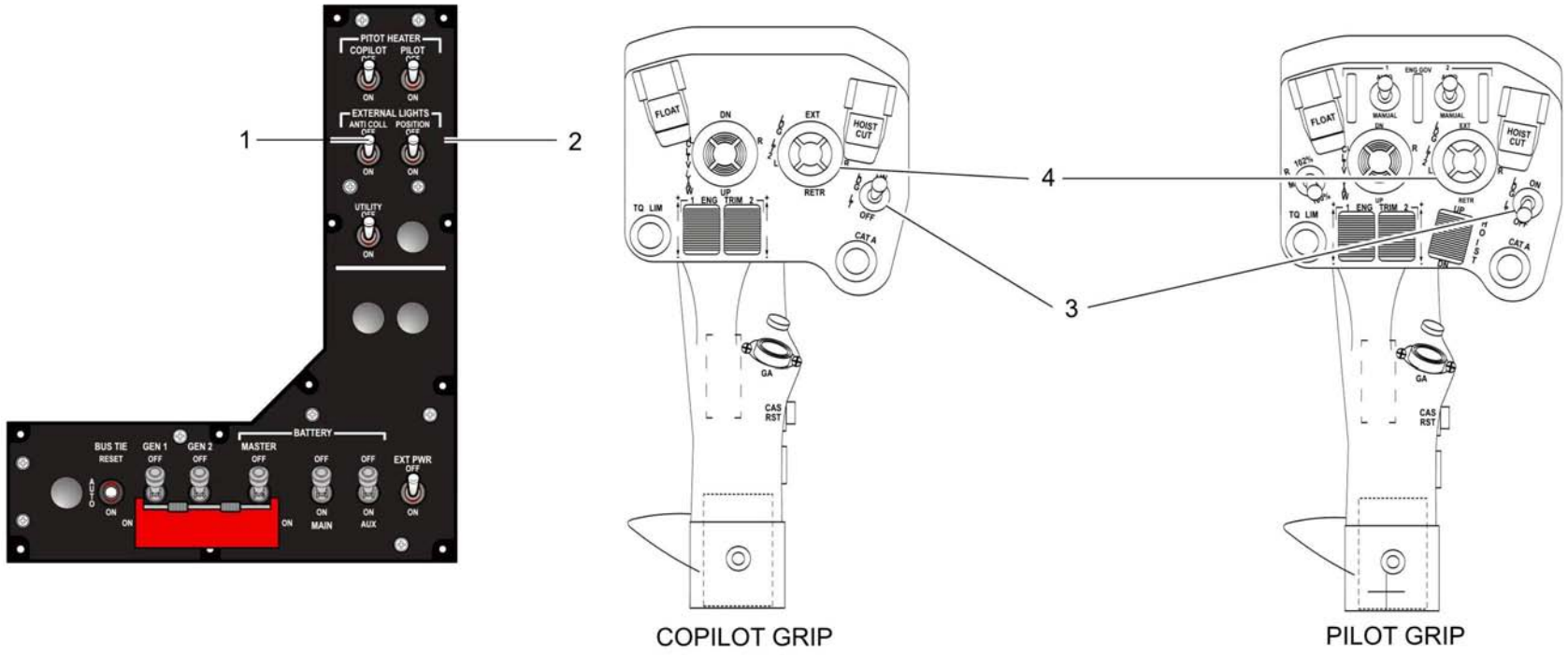
4 LDG LT2 coolie hat with four way switch and a central momentary contact:
- Pressed: ..... allows to light ON/OFF the lamp
- EXT: ..... allows to extend the light if the lamp is ON only
- RETR: ..... allows to retract the light
- L: ..... allows to rotate the light to left
- R: ..... allows to rotate the light to right

SEC LANDING LIGHT OPERATIVE TABLE

EXTENDED ANGLE	ROTATION ANGLE	LAMP ILLUMINATION
0° to 50°	From 0° to 132.5° CW or From 0° to 227.5° or viceversa	YES
	From 132.5° to 227.5° or viceversa	NO
50° to 60°	TRANSITION ZONE	1
60° to 120°	360°	YES always

NOTE A

The transition zone is the range in the extended angle that, due to tolerances, may or may not cause the lamp to be extinguished beyond 132.5° rotation left or right.



ICN-39-A-334000-G-A0126-01436-A-01-1

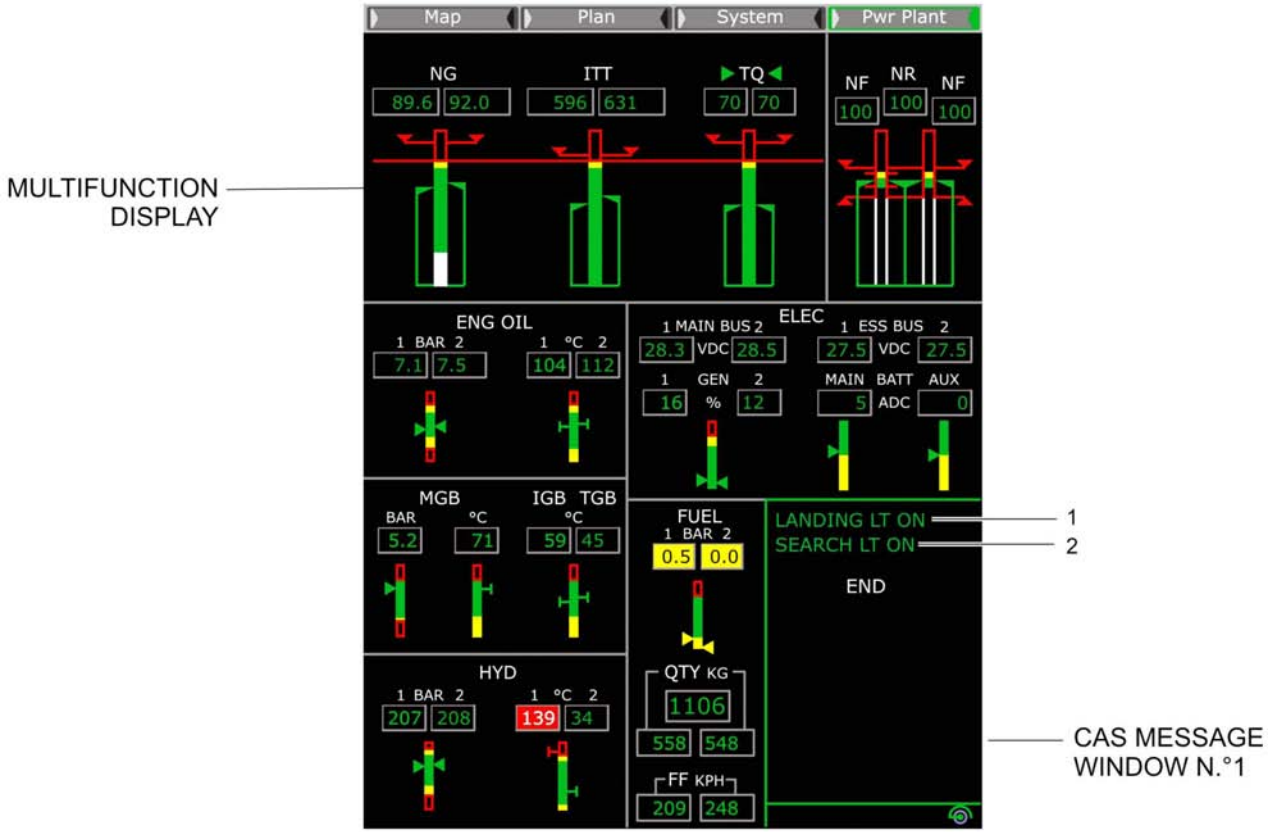
EXTERNAL LIGHTS CONTROL

EXTERNAL LIGHTS INDICATORS

Only advisory messages are involved with the external lights and displayed in the CAS message window.

1 LANDING LT ON: ..... when the landing lights are lit.

2 SEARCH LT ON: ..... when the secondary landing light is lit.

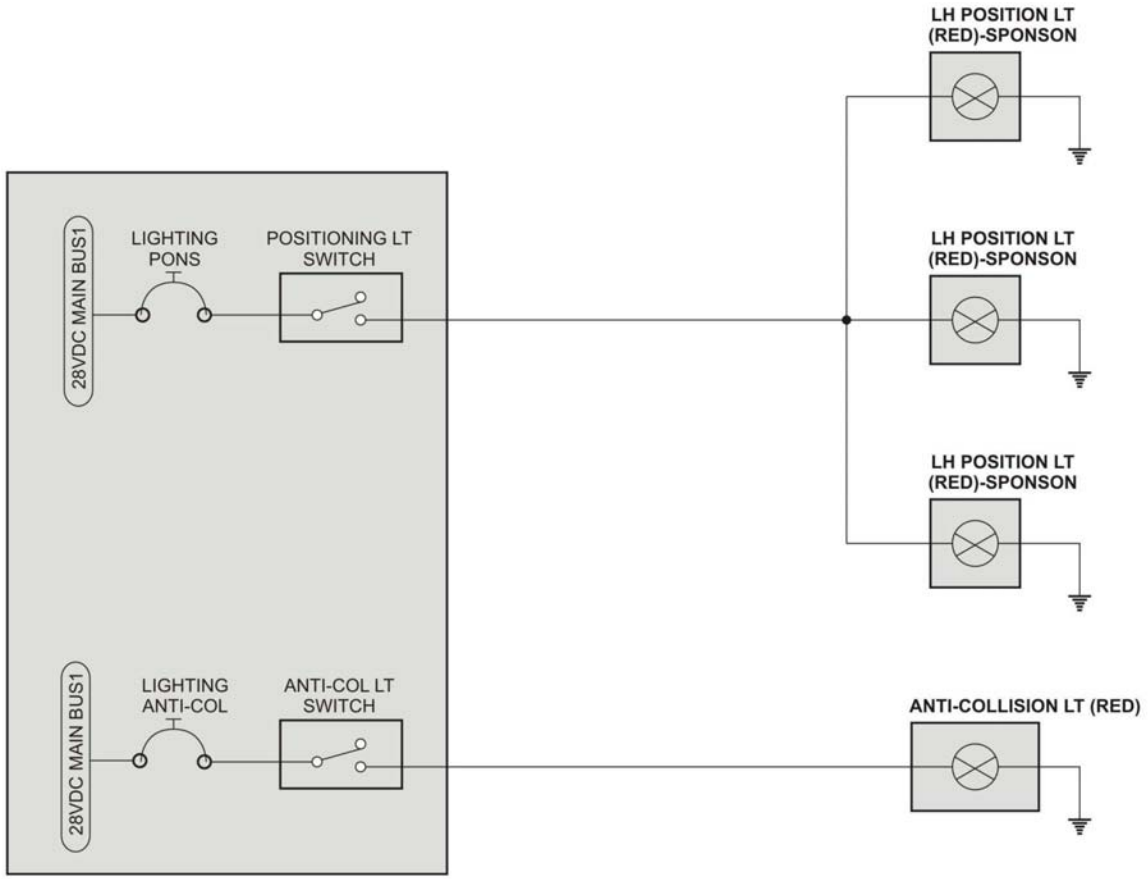


ICN-39-A-334000-G-A0126-01437-A-01-1

EXTERNAL LIGHTS – INDICATION

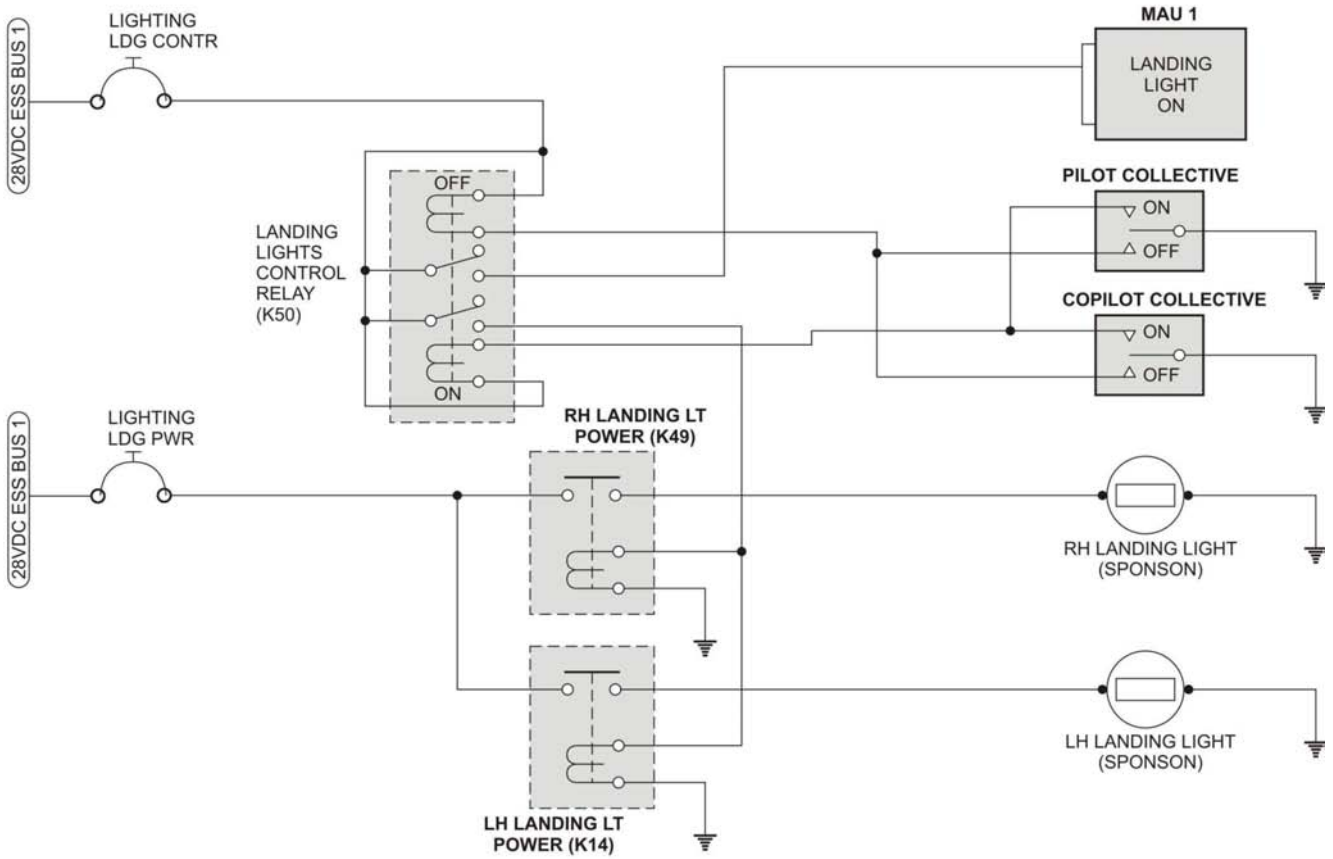
LEADING PARTICULARS

- Anti-collision light:
  - Nominal supply voltage..... 28 VDC (ESS BUS 1)
  - Power consumption..... 45 W max
- Position light:
  - Nominal supply voltage:..... 28 VDC (ESS BUS 1)
  - Power consumption:..... 26 W max (green light)
26 W max (red light)
28 W max (white light)
- Landing light (2 each):
  - Nominal supply voltage..... 28 VDC (ESS BUS 1)
  - Power consumption..... 250 W each
- Secondary landing light:
  - Nominal supply voltage..... 28 VDC (MAIN BUS 2)
  - Power consumption..... 450 W



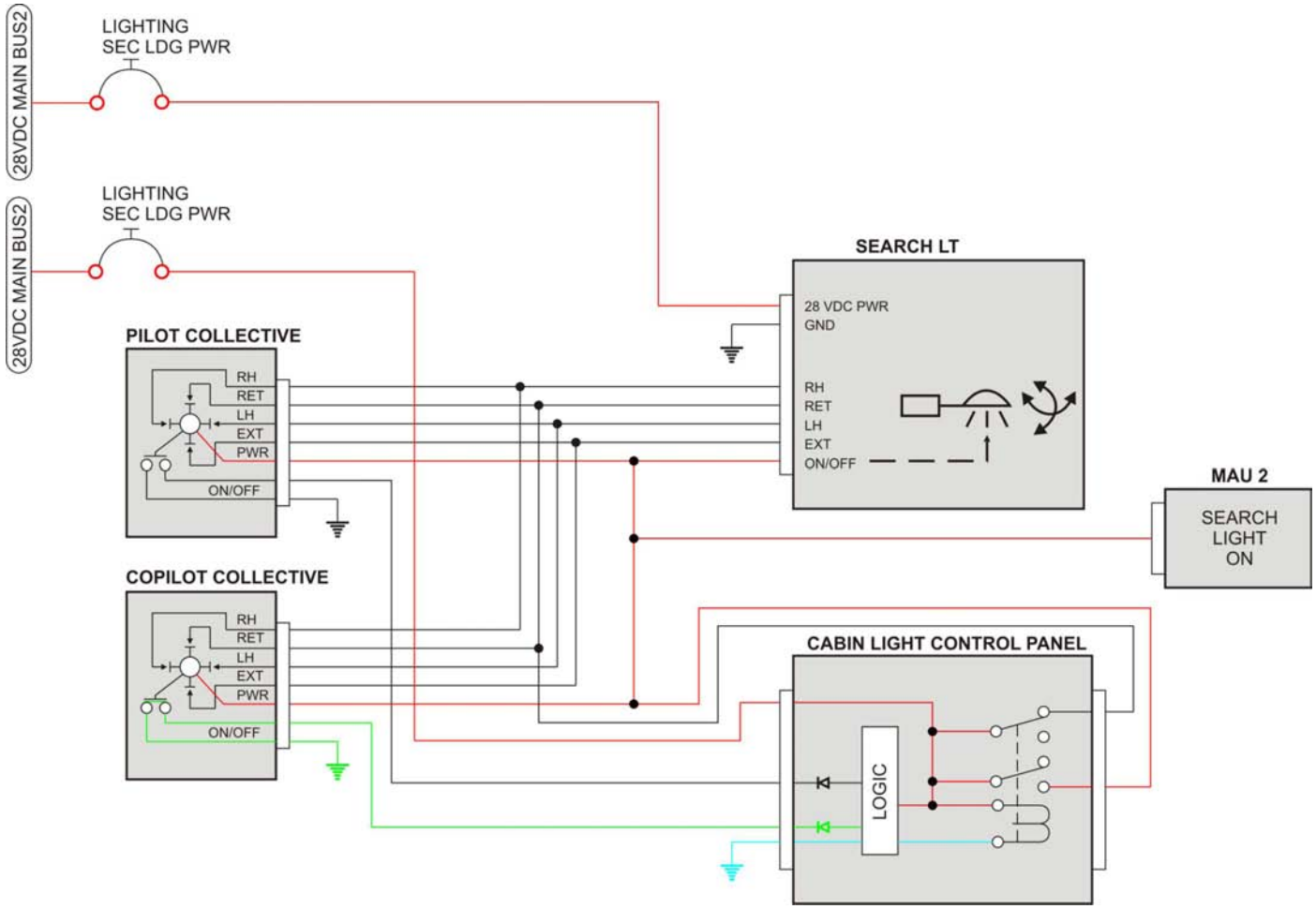
ICN-39-A-334000-G-A0126-01438-A-01-1

POSITION/ANTICOLLISION LIGHTING ELECTRICAL SCHEMATIC



ICN-39-A-334000-G-A0126-01439-A-01-1

LANDING LIGHT ELECTRICAL SCHEMATIC



ICN-39-A-334000-G-A0126-01440-A-01-1

SECONDARY LANDING LIGHT ELECTRICAL SCHEMATIC

SECTION 50 - EMERGENCY LIGHTING

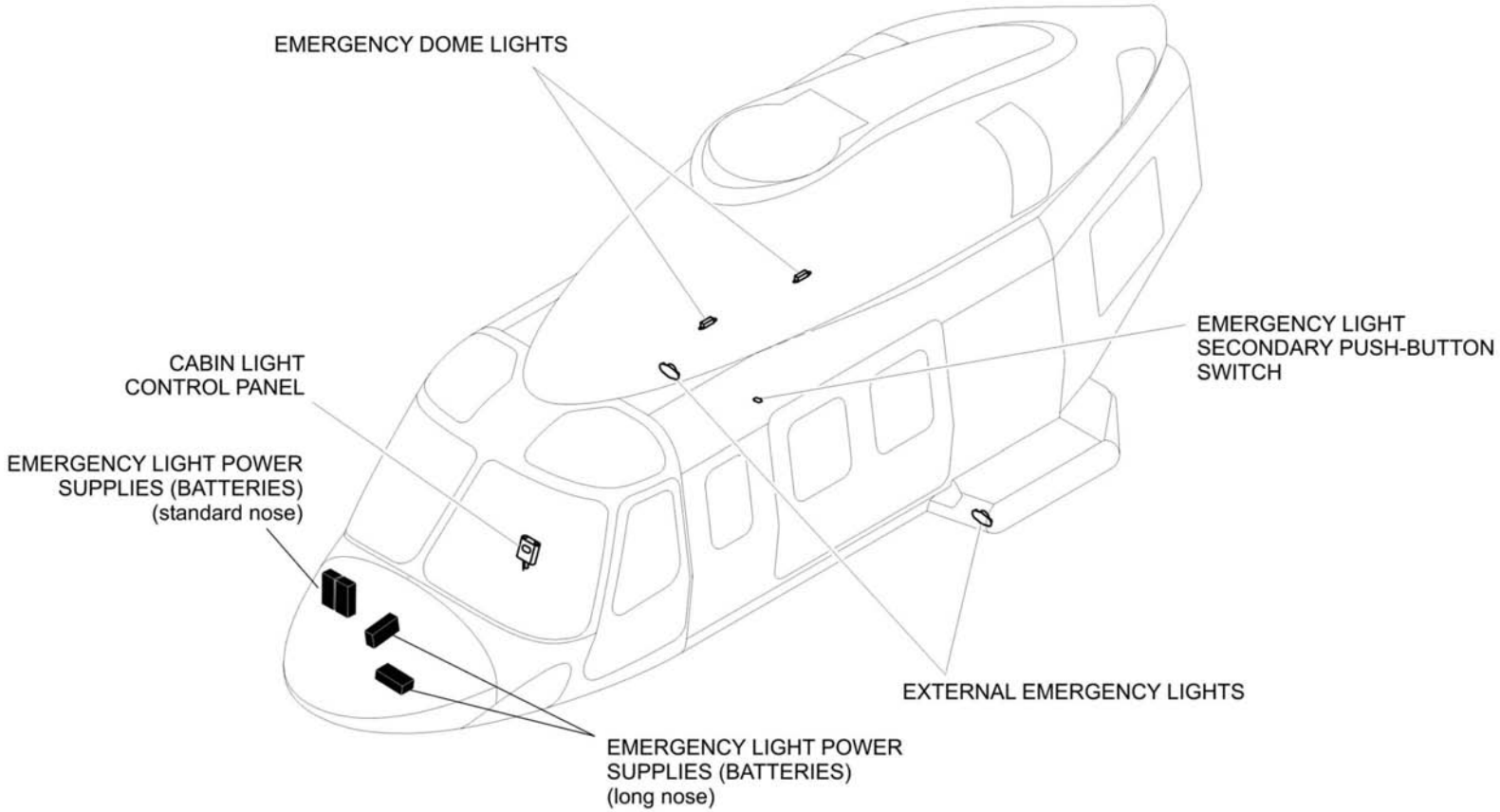
DESCRIPTION

The emergency lights system includes the following components:

- two battery packs rechargeable from the aircraft power supply 28VDC which supply the Emergency lights at 6 VDC
- two emergency dome lights installed on the cabin roof
- two external emergency lights installed in the LH and RH sponsons

OPERATION

The Internal and External Emergency lights are supplied by two rechargeable batteries installed in the RH nose bay.
The rechargeable battery (NiCad) power supply consumption is 28 VDC, 350 mA and the its output power supply is 6.1 VDC with 7 A.
In charge mode, the batteries are fully charged after 4 hours of continuous charging at 28 VDC. The batteries are then capable of supplying 7 A at 6.1 V to 6.5 V for at least 15 minutes.
The two internal emergency lights are installed in the cabin roof to provide enough general lighting in the passenger cabin. These lights are white LEDs.
The two external emergency lights are provided at each emergency exit when the evacuee is likely to make first contact with the ground outside the cabin.
The two emergency lights installed on the sponsons are incandescent halogen lamps.
All the emergency lights are controlled by a switch installed on the Cabin Light Control Panel and a secondary switch installed in the left side of the internal cabin.
This switch is a push button type, illuminated and controlled by the cabin crew.



ICN-39-A-335000-G-A0126-01441-A-01-1

EMERGENCY LIGHTING COMPONENT LOCATION

EMERGENCY LIGHTING CONTROLS

The emergency lights have the control housed on the Cabin Light Control Panel located on the interseat and through the secondary push button switch installed at the left cockpit door. The Cabin Light Control Panel provides the following functions:

1 EMERG switch

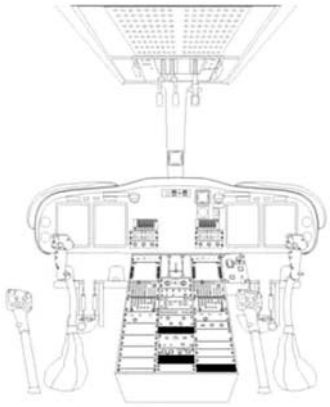
- OFF: ..... all the emergency lights are OFF and the two batteries are recharging.
- ARM: ..... all the emergency lights are OFF and the two batteries are recharging. The lights are lit automatically upon loss of the aircraft power.
- ON: ..... all the emergency lights are powered by the two batteries.

NOTE A

On each battery, there is a red push button that, when pressed, provides the lighting of the emergency lights.

The emergency light secondary push button switch provides the following functions:

- OFF:..... all the emergency lights are OFF and the batteries are recharging. This position gives system control to the switch on the Cabin Light Control Panel.
- ON:..... all the emergency lights are operated by the emergency light power supplies (batteries) regardless of the switch on the Cabin Light Control Panel and independent of the aircraft power supply.



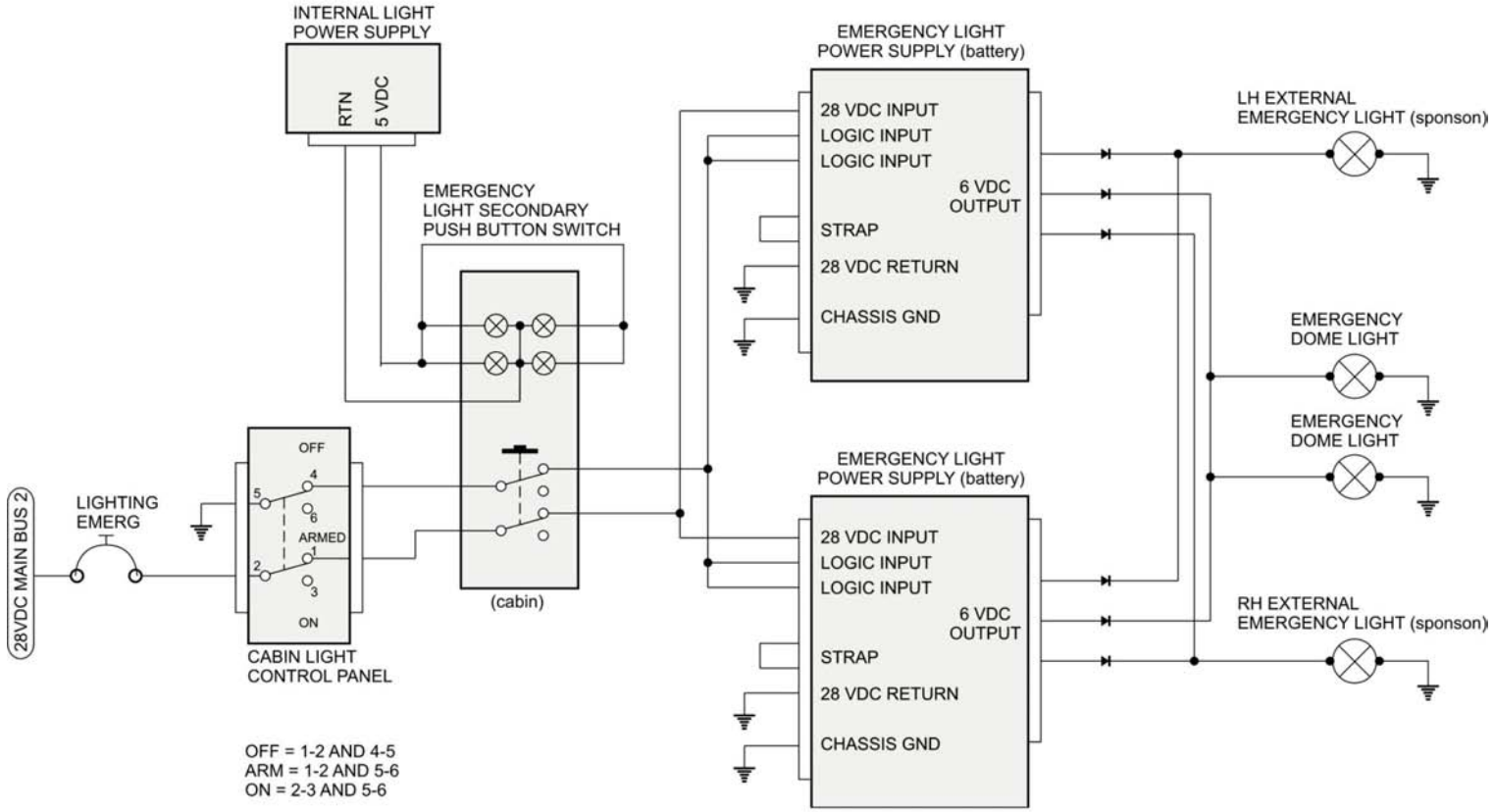


ICN-39-A-335000-G-A0126-02995-A-01-1

EMERGENCY LIGHTING CONTROLS

LEADING PARTICULARS

- Emergency light power supplies (2 nichel-cadmium batteries):
  - Nominal voltage supply..... 28 VDC (ESS BUS 1)
  - Input current consumption..... 350 mA
  - Output voltage supply..... 6.1 VDC to 6.5 VDC
  - Output current consumption..... 7 A
- Emergency dome light (2 each):
  - Nominal voltage supply..... 6 VDC each
  - Current consumption..... 0.6 A each
- External emergency light (sponson) (2 each):
  - Nominal voltage supply..... 6 VDC each
  - Current consumption..... 0.7 A each



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EMERGENCY LIGHTING ELECTRICAL SCHEMATIC

CHAPTER
34
NAVIGATION

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SECTION 00 - NAVIGATION

DESCRIPTION

The navigation system supplies the aircraft navigation data. It includes instrument flight, landing and other navigation components.
The navigation system has the systems as follow:

- Flight Environment Data
- Attitude and Direction
- Landing and Taxiing Aids
- Independent Position Determining
- Dependent Position Determining
- Flight Management Computing

FLIGHT ENVIRONMENT DATA

Flight environment data systems use sensors to measure the different environmental conditions, such as the conditions that follow:

- Air pressure
- Airspeed
- Altitude
- Pitot pressure
- Static pressure
- Outside air temperature.

This data has an effect on navigation.

ATTITUDE AND DIRECTION

Attitude and direction systems use magnetic or inertial forces to measure and give the attitude and direction of the aircraft.

LANDING AND TAXIING AIDS

Landing and taxiing aids systems help the aircraft safety approach, make a landing and taxi to an airport.

INDEPENDENT POSITION DETERMINING

Independent position determining systems use the equipment, other than ground station and/or orbital satellites, to identify the position of the aircraft.

DEPENDENT POSITION DETERMINING

Dependent position determining systems use ground stations and/or orbital satellites to determine the position and velocity of the aircraft.

FLIGHT MANAGEMENT COMPUTING

Flight management computing systems use the navigation data to calculate or control the position or flight path of the aircraft.

NAVIGATION SYSTEM	
34-10	**FLIGHT ENVIRONMENT DATA**
	Pitot and Static System
	Air Data System (ADS)
34-20	**ATTITUDE AND DIRECTION**
	Attitude and Heading Reference System (AHRS)
34-30	**LANDING AND TAXIING AIDS**
	VOR / ILS Data Link (VIDIL) System
34-40	**INDIPENDENT POSITIONING DETERMINING**
	Radio Altimeter System (RAD ALT)
	Weather Radar (WXR) (Options)
	TCAS (Options)
	EGPWS (Options)
34-50	**DEPENDENT POSITION DETERMINING**
	Distance Measuring Equipment (DME)
	Air Traffic Control Transponder (ATC XPDR)
	Automatic Director Finder (ADF)
	Global Position System

NAVIGATION SYSTEM	
34-60	**FLIGHT MANAGEMENT SYSTEM (FMS)**
	Flight Management System (FMS)

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SECTION 10 - FLIGHT ENVIRONMENT DATA

DESCRIPTION

Flight environment data has the system that follow:

- Pitot and Static System
- Air Data System (ADS)

The Flight environment data system permits the crew to select the Barometric Reference Pressure and its purpose is to supplies the crew with the display of the:

- Airspeed
- Vertical Speed
- Barometric Altitude
- Outside Air Temperature.

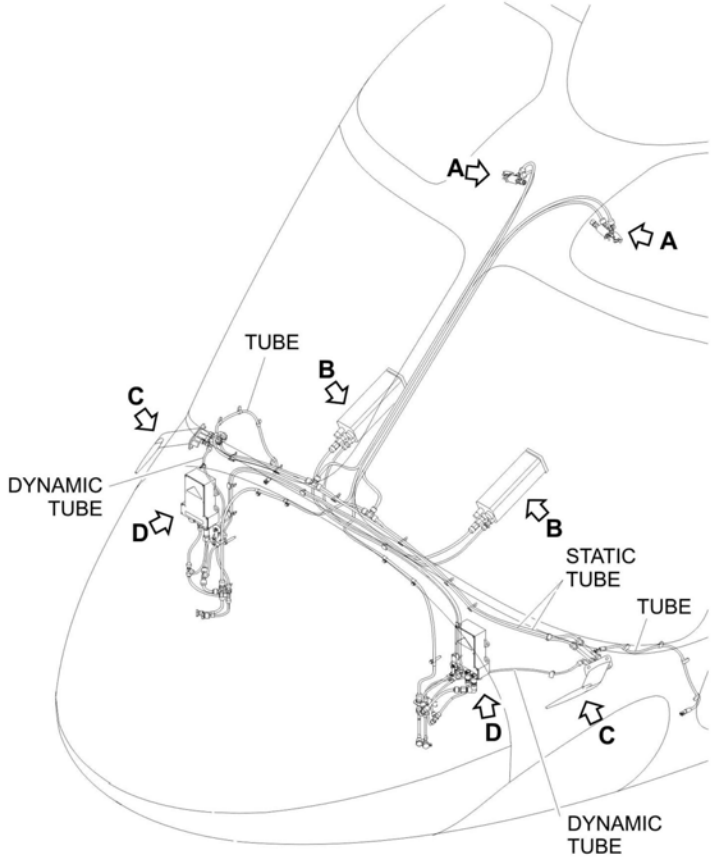
PITOT AND STATIC SYSTEM

The Pitot and static system has the primary components that follow:

- Number 1 Pitot – static probe
- Number 2 Pitot – static probe
- Number 1 alternate static – source valve
- Number 2 alternate static – source valve.

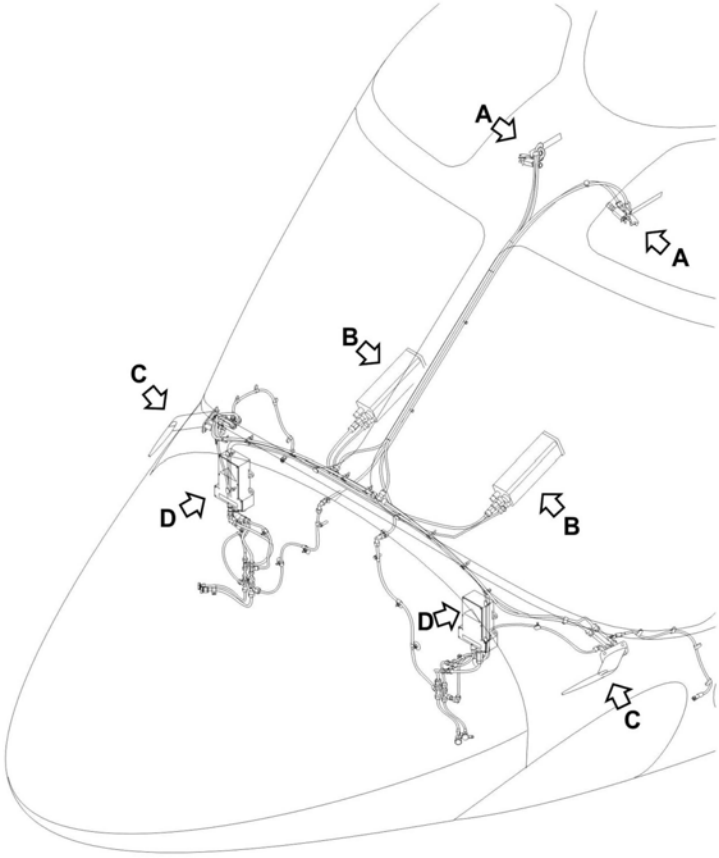
The number 1 and the number 2 Pitot – static probes are installed respectively on the left and on the right side of the helicopter nose (STA 2110).
Four screws attach the Pitot – static probe to the structure and a set of nylon tubes connects the Pitot – static tubes with the air-data management system of the helicopter.

The Pitot – static probes sense the total and static pressure and have three output ports; two output ports are for the static pressure (S1 and S2) and one for the total pressure (P).
The static port has a left-to-right cross-link with the “Tee – fitting”.
There are four drain points in the system to collect the moisture and thus prevent incorrect indications.
The electrical connector HR5P1 connects the number 1 Pitot – static probe (HR5) to the left current monitor (A19) and the connector HR6P1 connects the number 2 Pitot – static probe (HR6) to the right current monitor (A18).
The number 1 and the number 2 alternate static source valves are installed respectively on the left and on the right side of the lateral structure of the overhead console.
The static-source valve has two fittings that connects the IN static tube and the OUT static tube and there is a toggle switch, that is used to do the selection of the alternate static source valve and to cancel it. The toggle switch is usually set at NORM, but must be moved to the ALTN position if the system indications become incorrect. When the toggle switch is at ALTN, the source of the static pressure becomes the alternate static source-valve. The switch has a red guard for protection from inadvertent operation.



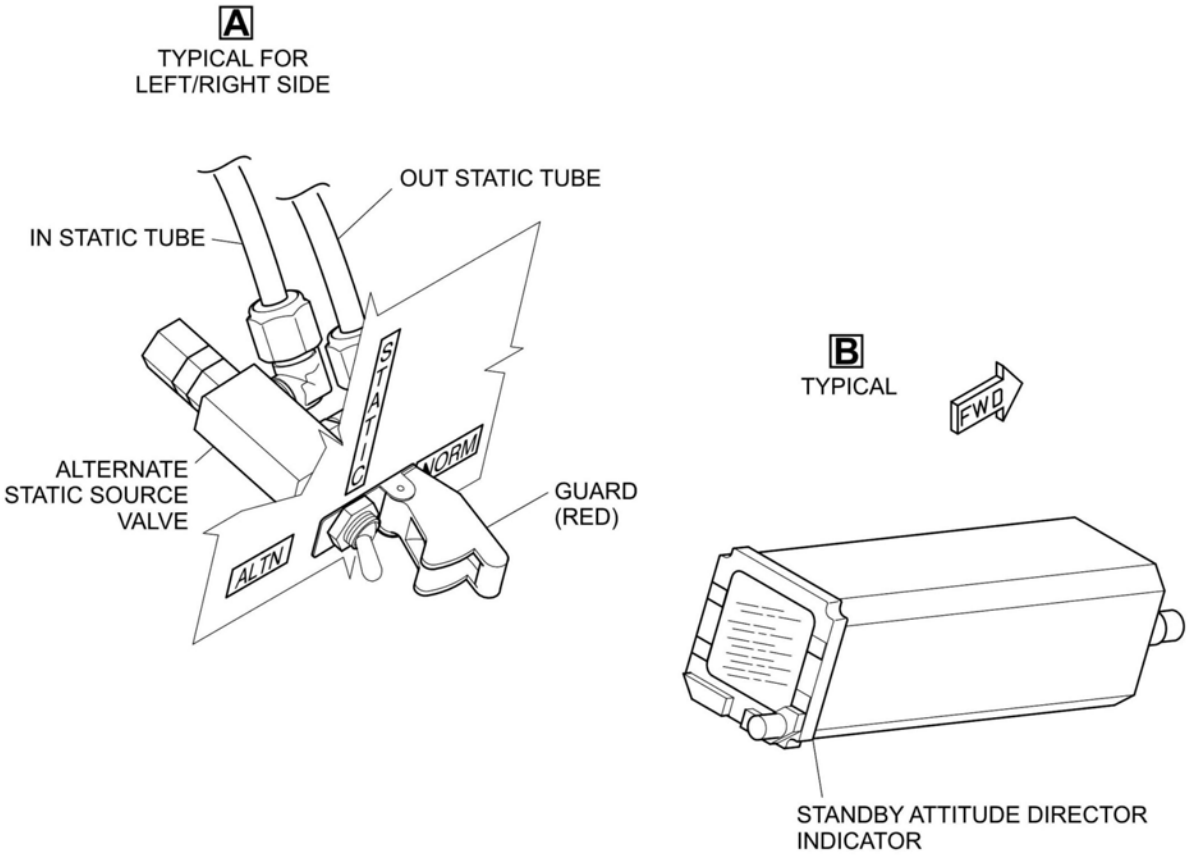
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STANDARD NOSE HELICOPTER CONFIGURATION



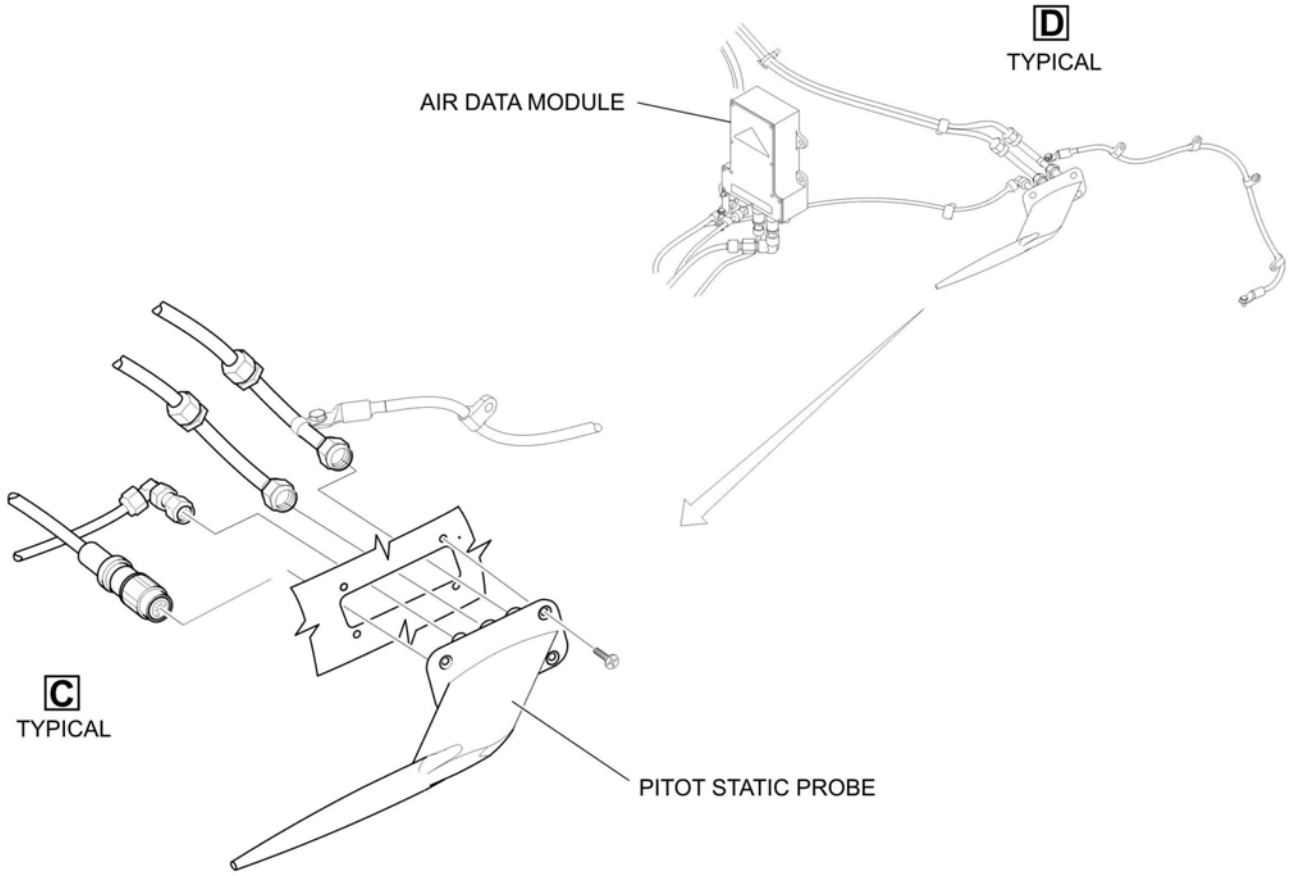
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LONG NOSE HELICOPTER CONFIGURATION



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SYSTEM COMPONENTS (SHEET 1 OF 2)

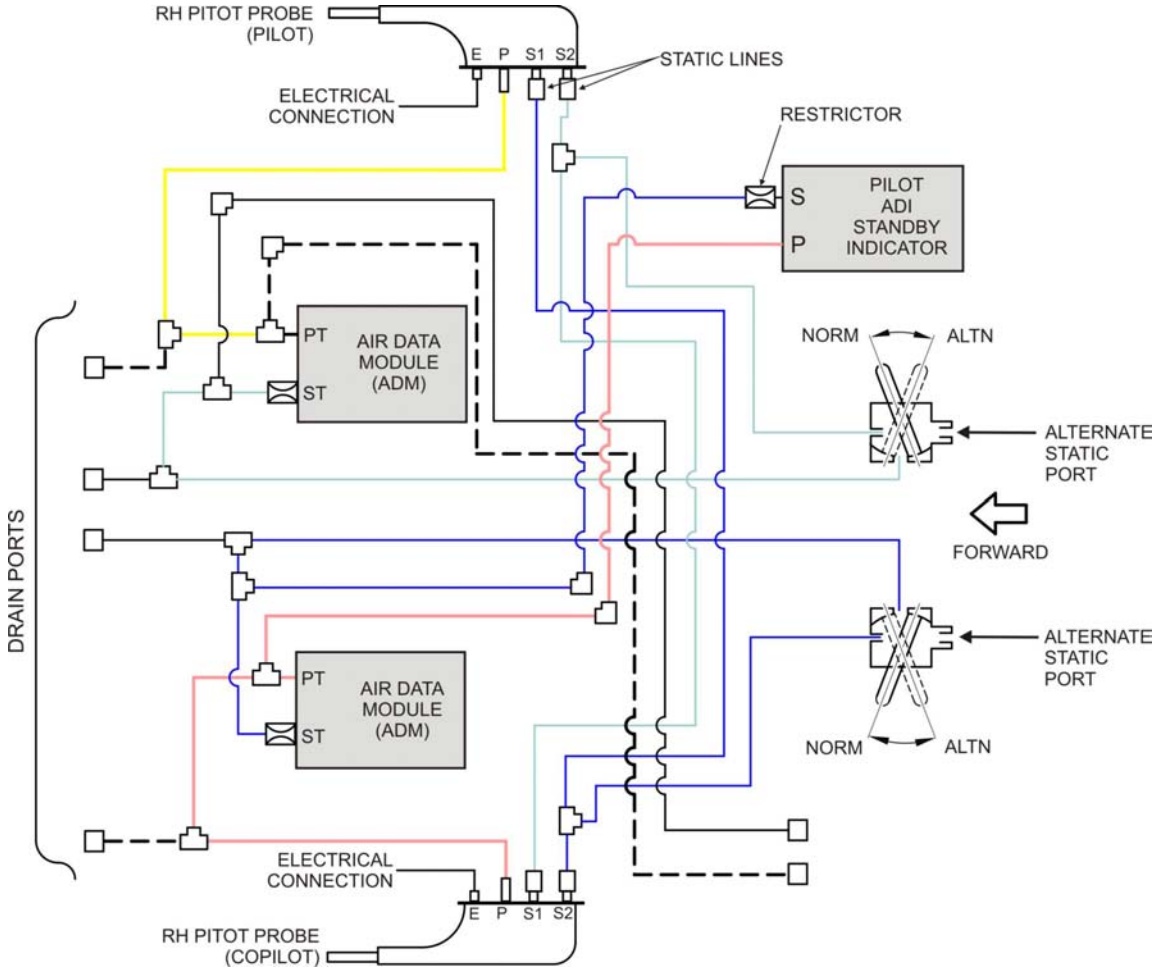


ICN-39-A-341000-G-A0126-01251-A-01-1

SYSTEM COMPONENTS (SHEET 2 OF 2)

OPERATION

The Pitot and static system senses the air data (total and static pressure) necessary to show the related indications on the instruments in the cockpit. The Pitot and static system sends, trough the Pitot – static tubes, the sensed data to the two air data modules (ADMs) and the two standby instruments. The ADMs change the data received from the Pitot and static system into digital signals. These signals are then transmitted to the helicopter computers (MAUs) for the calculation of the related flight data (airspeed, altitude and rate of climb or descent). These data are then shown on the primary flight displays (PFDs).
In the Pitot – static probe, the total pressure output port (P) transmits the pressure signal to the on side Air data module (ADM) and to the cross side stand-by instrument.
In the left Pitot – static probe the static pressure output port S1 is left-to-right cross – link with the “Tee – fitting” to the S2 port, in the right Pitot – static probe and trough the number 1 alternate static source valve, they transmit the static pressure signal to the ADM2 and to the number 1 standby instrument (optional).
In the left Pitot – static probe the static pressure output port S2 is left-to-right cross-link with the “Tee – fitting” to the S1 port, in the right Pitot – static probe and trough the number 2 alternate static source valve, they transmit the static pressure signal to the ADM1 and to the number 2 standby instrument.
The two standby instruments are the backup for the PFDs indications and they also show airspeed, altitude and rate of climb or descent (vertical speed).



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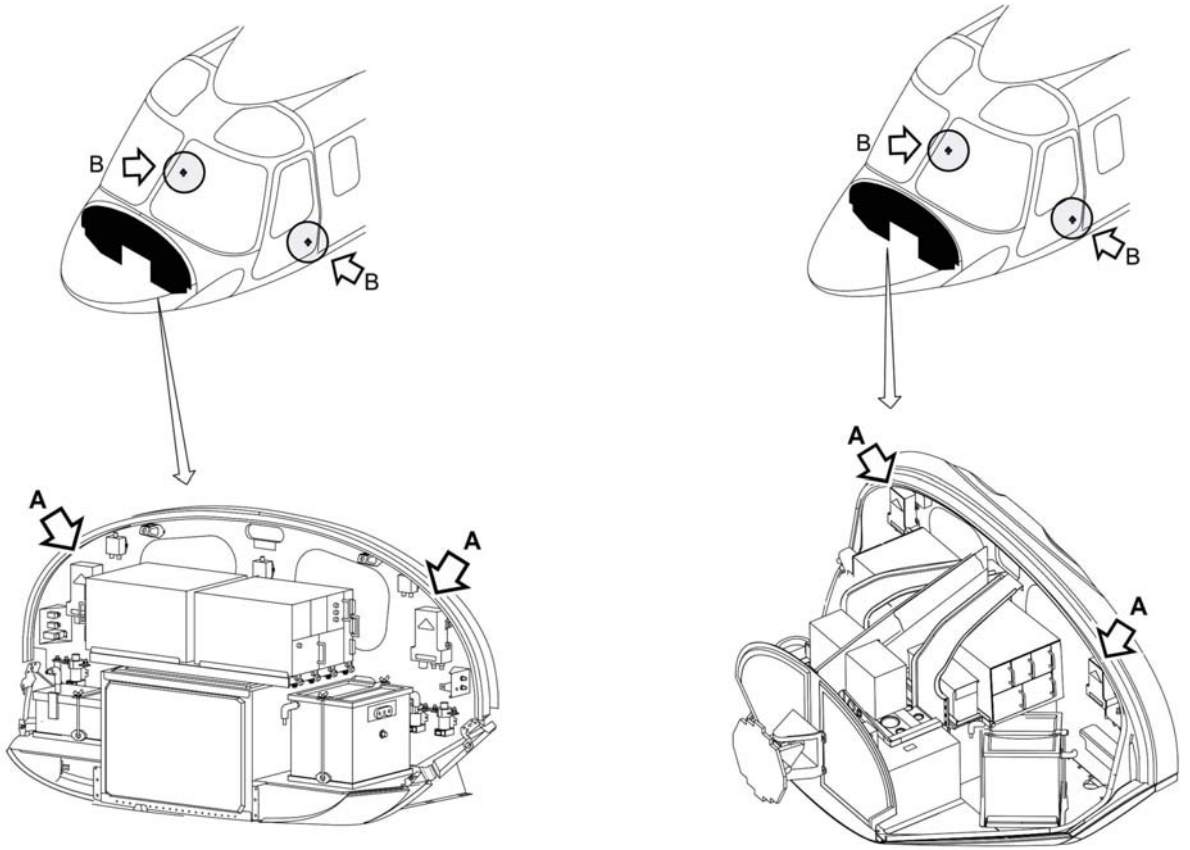
AIR DATA SYSTEM SCHEMATIC DIAGRAM

AIR DATA SYSTEM (ADS)

The primary components of the air data system are the:

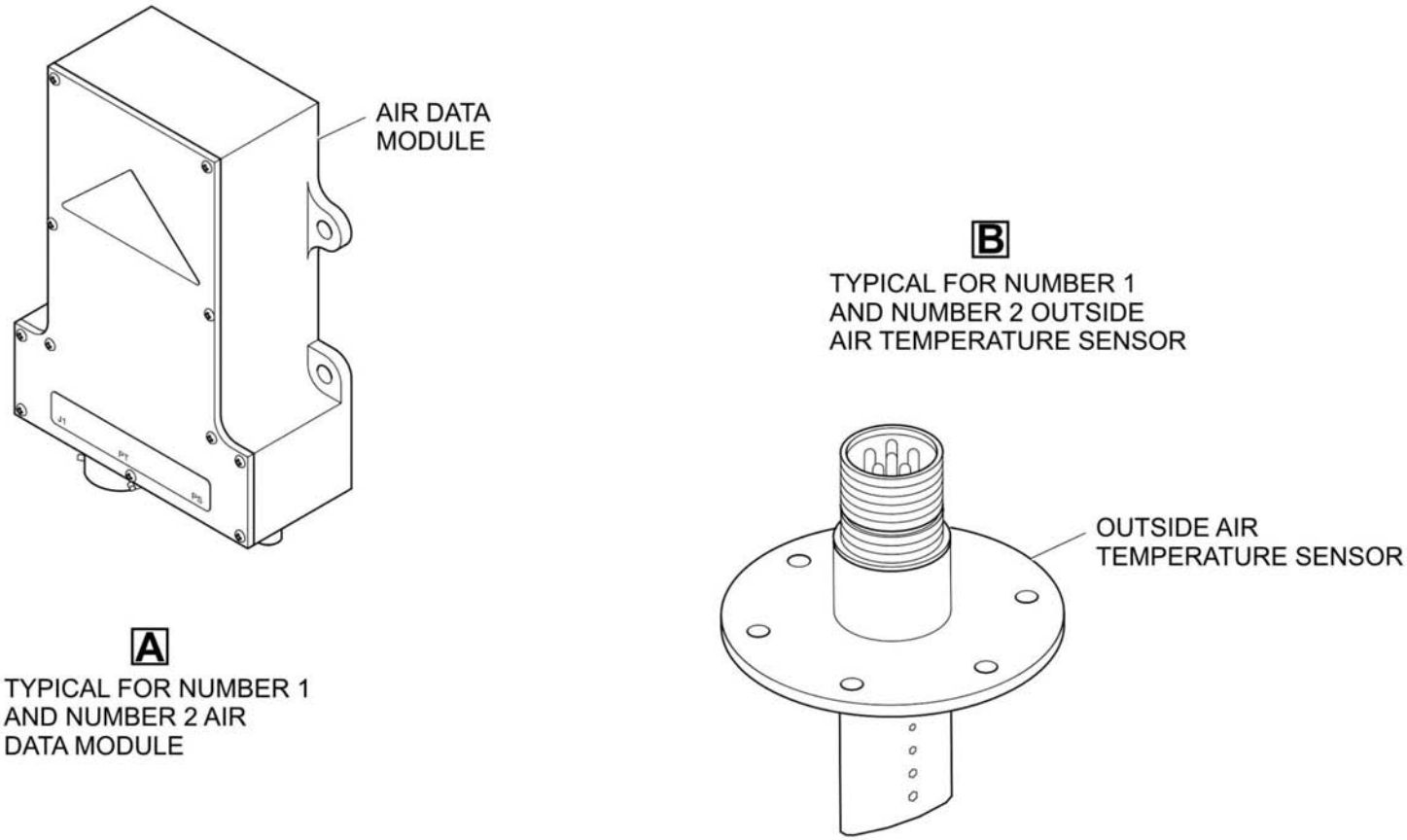
- Number 1 air data module (ADM1)
- Number 2 air data module (ADM2)
- Number 1 outside air temperature sensor (OATS1)
- Nmber 2 outside air temperature sensor (OATS2)

The number 1 and the number 2 air data modules (ADMs) are installed respectively on the left and on the right side of the nose.
Four screws and four washers attach each ADM to the nose bulkhead.
The connector A5P1 connects the ADM1 (A5) to the circuit breaker ADM1 (CB39) and to the MAU1 (A1); the connector A6P1 connects the AIDM2 (A6) to the circuit breaker ADM 2 (CB38) and to the MAU2 (A2).
Each ADM has the total pressure (PT) and the static pressure (PS) input ports. PT is connected to the total pressure output port P of the pitot-static probe on side.
In the ADM1 PS is connected to the static pressure output ports S1-S2 (left-to-right) of the pitot-static probes and in the ADM2 the PS is connected to the static pressure output ports S2-S1 (left-to-right) of the pitot-static probes.
The number 1 and the number 2 outside temperature sensors (OATSs) are installed respectively on the left and on the right bottom side of the fuselage.
Six screws attach each OATS on the bottom fuselage structure.
The ADM is supplied with +28 V DC electrical power from the helicopter.
The connector RT7P1 connects the OATS1 (RT7) to the number 1 Modular Avionic Unit (MAU1) (A1) and to the Heating Control Box (HR4) and a connector RT7P2 connects the OATS2 (RT8) to the number 2 Modular Avionic Unit (MAU2) (A2).
The custom input/output (CSIO) module supplies the sensors excitation. The ADS has two sensors that have different excitation and are read separately. The OAT sensor channels connect to on side MAU.
The display controllers, which have the BARO correction input knobs, transmit a count that is related to knob movement to the control input/output (CIO) modules. Each display controller connects to on side MAU.



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AIR DATA SYSTEM (SHEET 1 OF 2)



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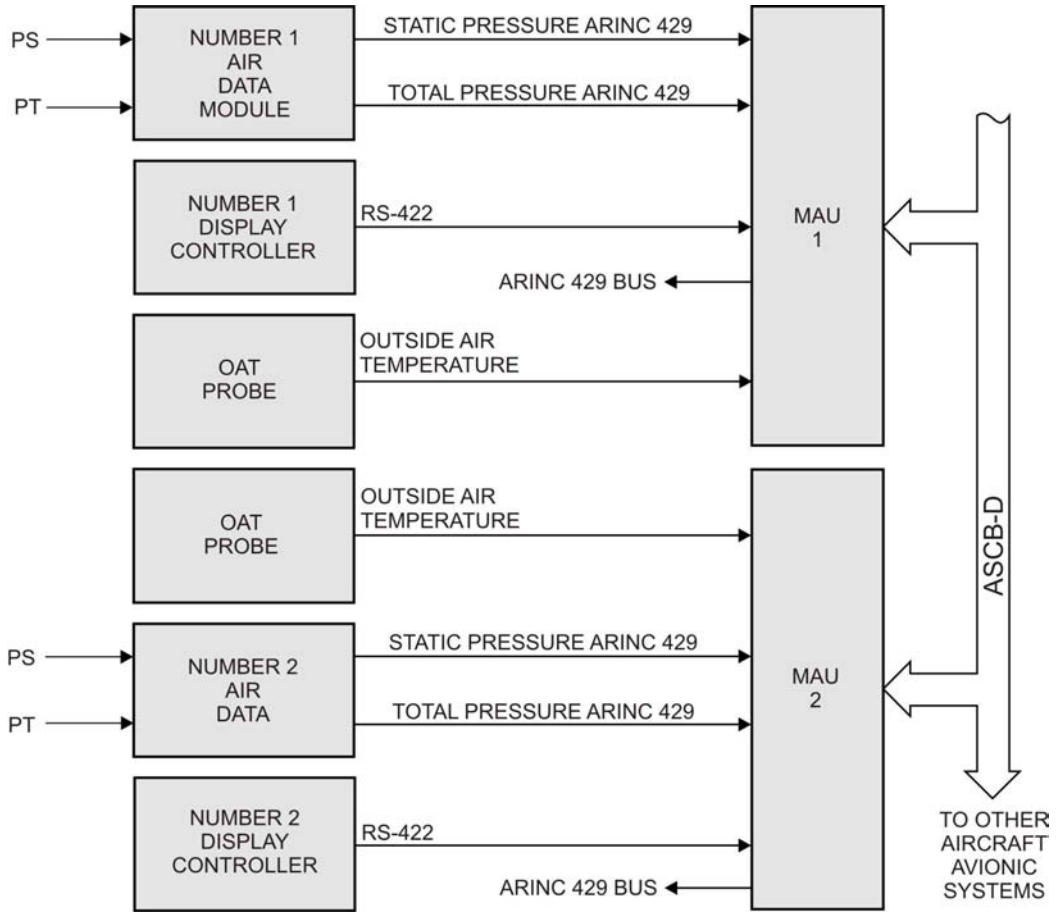
AIR DATA SYSTEM (SHEET 2 OF 2)

The ADS is a dual system that operates with two modular avionics units (MAUs) to shows on the primary flight displays (PFDs) altitude, airspeed, vertical speed, outside air temperature and to supply air data for other avionic systems. The pilot and copilot each have an ADS. Each pilot uses their onside ADS application.
Each ADS has the inputs, settings and software that follow:

- sensor inputs that include pressure signals from the air data module (ADM)
- sensor inputs that include data from external air temperature (OAT) probes
- barometric (BARO) correction settings
- air data application (ADA) software that is installed in the actuator input/ output with processor (AIOP) module in each MAU.

The ADA uses data inputs to calculate the aircraft altitude, airspeed, and vertical speed.
The other systems that also use this data follow:

- The Traffic Alert and Collision Avoidance System (TCAS)
- The Automatic Flight Control System (AFCS)
- The Flight Data Recorder (FDR)
- The Flight Management System (FMS)
- The radar system.

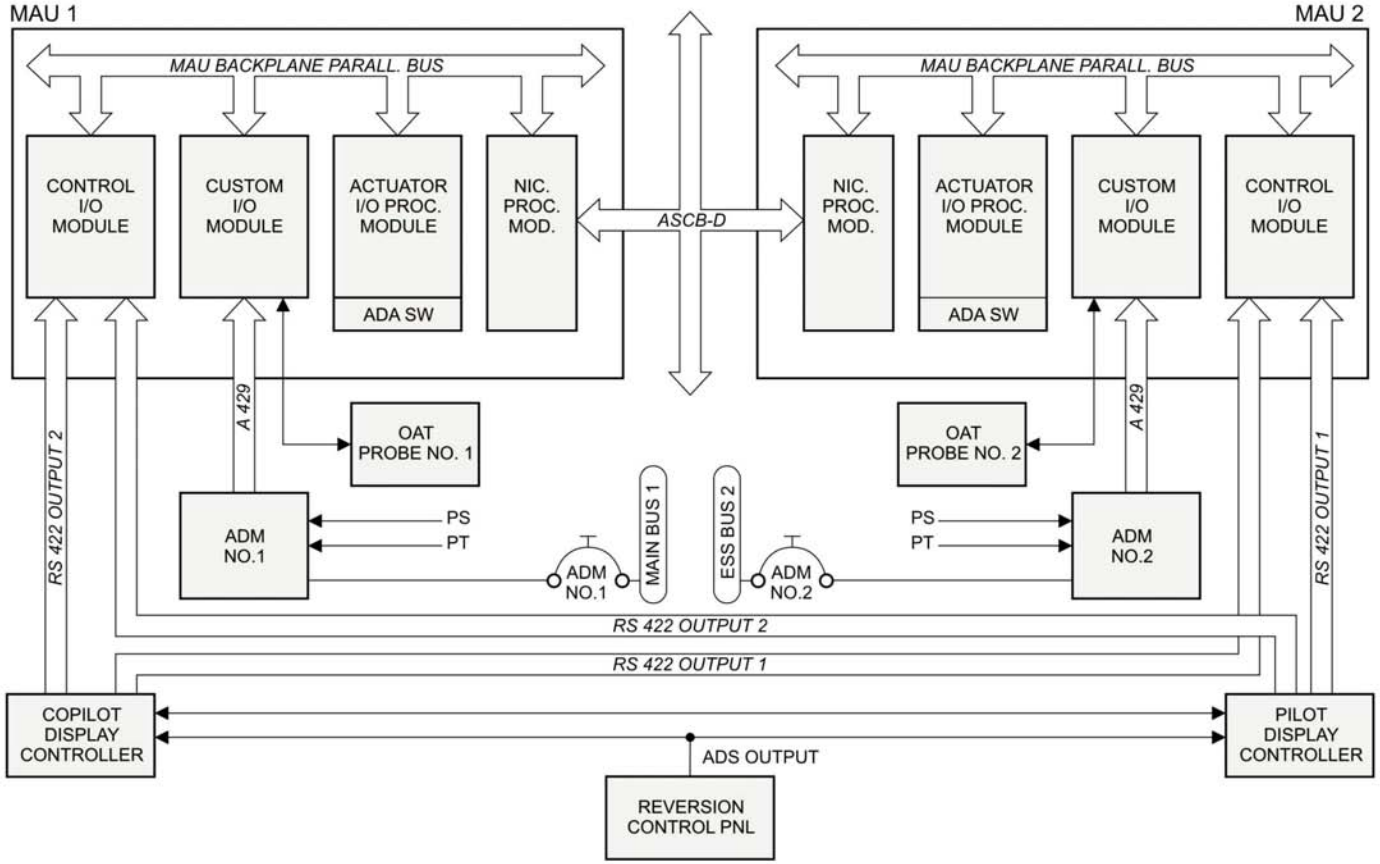


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AIR DATA SYSTEM – OVERVIEW

OPERATION

Each ADM senses and calculates the absolute pressure applied to the PS and PT input ports and the low-speed ARINC 429 data buses transmit the pressure data to the CSIO module in the MAU. The CSIO module also receives data from the OAT probe.
The CIO module in the MAU receives the barometric correction value from the barometric set control on the display controller (DC). The CSIO/CIO modules send the data through the MAU back plane to the air data application (ADA) software that is installed on the actuator input/output with processor (AIOP). The ADA software uses these signals, along with OAT, BARO setting and discrete data to calculate the air data parameters of the aircraft altitude, airspeed and vertical speed. The AIOP module sends the air data through the MAU back plane to the Network Interface Controller with Processor (NIC/PROC). The NIC/PROC transmits the air data on the ASCB-D data bus to Display Units (DUs) and other avionics systems that use air data to operate. The BARO correction supplies adjustment of the altitude calculation for local weather effects on the static pressure measurement and supplies the barometric correction inputs for the ADS. The input is entered by the pilots related to the instructions from air traffic controller (ATC) or from Automatic Terminal Information Service (ATIS). The CIO module counts to the ADA are relative position data of a quadrature knob. The counts are changed to inches-of-mercury and millibars for transmission. There is one barometric set knob for each ADS. The ADS includes a VNE (Velocity Never Exceed) curve that shows the maximum operation airspeed for the AW – 139 as a function of pressure altitude and temperature; the CIO module changes these input discretes to a value related to the position of the barometric set control and then sends the correction code through the MAU back plane to the AIOP module.
The two display controllers are cross-wired so that each display controller transmits both pilot and copilot barometric count data. The cross-wiring lets operation to continue even if there is a failure of a single display controller. If there is a failure of both display controllers, the barometric correction output shown for ADA 1 and ADA 2 stay unchanged at the last value.



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ADMS / OATS ELECTRICAL SCHEMATIC

CONTROLS AND INDICATORS

The Primary Flight Display provides the information of the airspeed, vertical speed, barometric altitude and OAT.

Outside Air Temperature

The digital readout range is -55°C to +70° C. Loss of information from the MAU results the data removal and replacement by dashes.



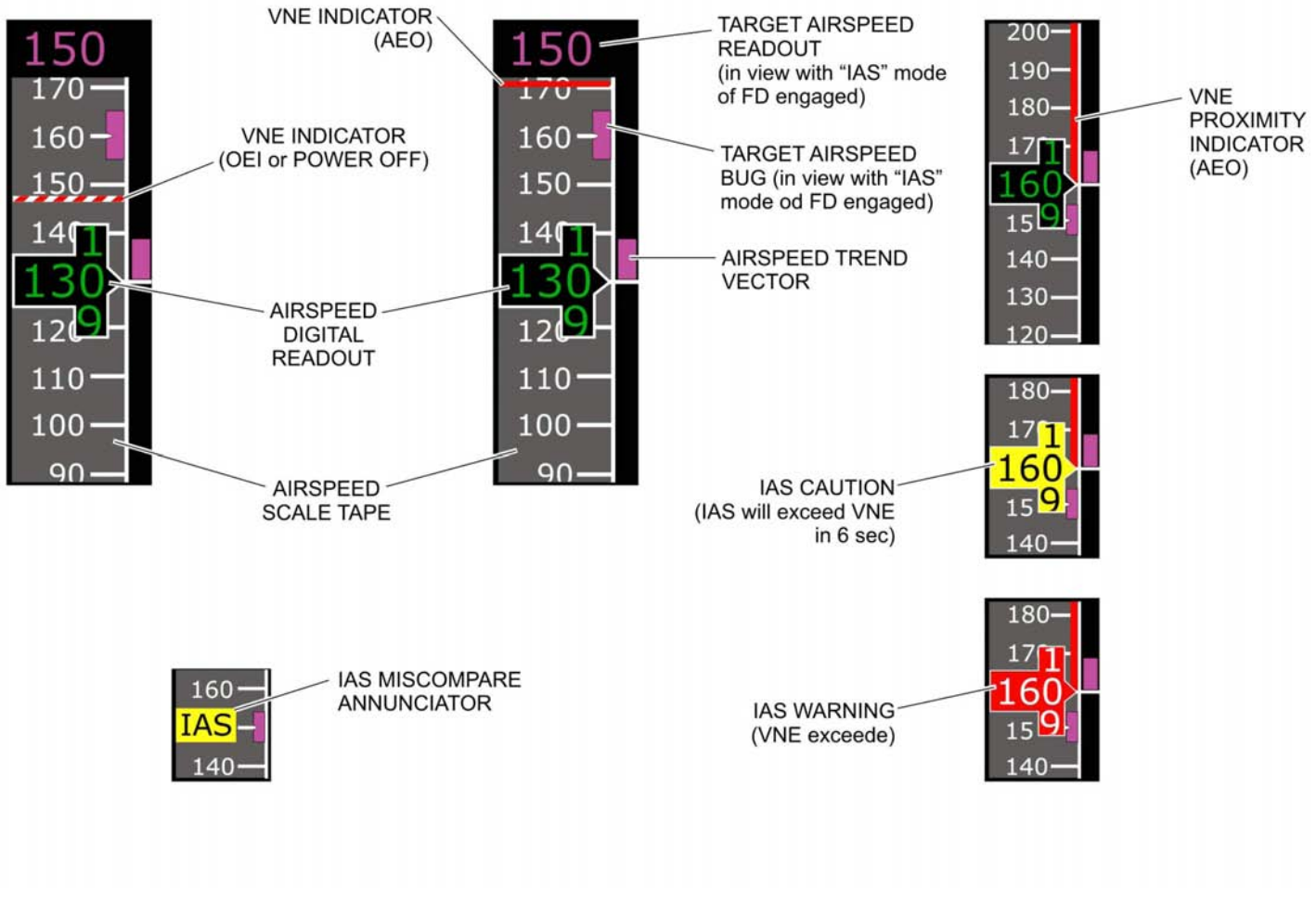
PRIMARY FLYGHT DISPLAY

ICN-39-A-341000-G-A0126-01257-A-01-1

CONTROLS AND INDICATORS

Airspeed Tape

It is selected from either the on-side or cross-side ADS. The tape continuously displays an 80 knot window. The tape is graduated in 10 knot increments. A magenta selected airspeed bug corresponds to the selected speed target displayed in the digital readout. The current airspeed window is positioned in the center of the tape. A magenta rectangular bar traveling vertically on the right outside edge of the tape represents the airspeed trend vector indicator. The airspeed tape displays a red bar vertically along the right inside of the tape at 5 knots prior to exceeding Vne (at AEO). A barber pole symbol is instead displayed along the right inside of the tape at 5 knots prior to exceeding Vne (at OEI or AUTOROTATION.) Loss of airspeed information form the ADM results in the removal of the digits in the window and replaces the airspeed tape with a red cross.

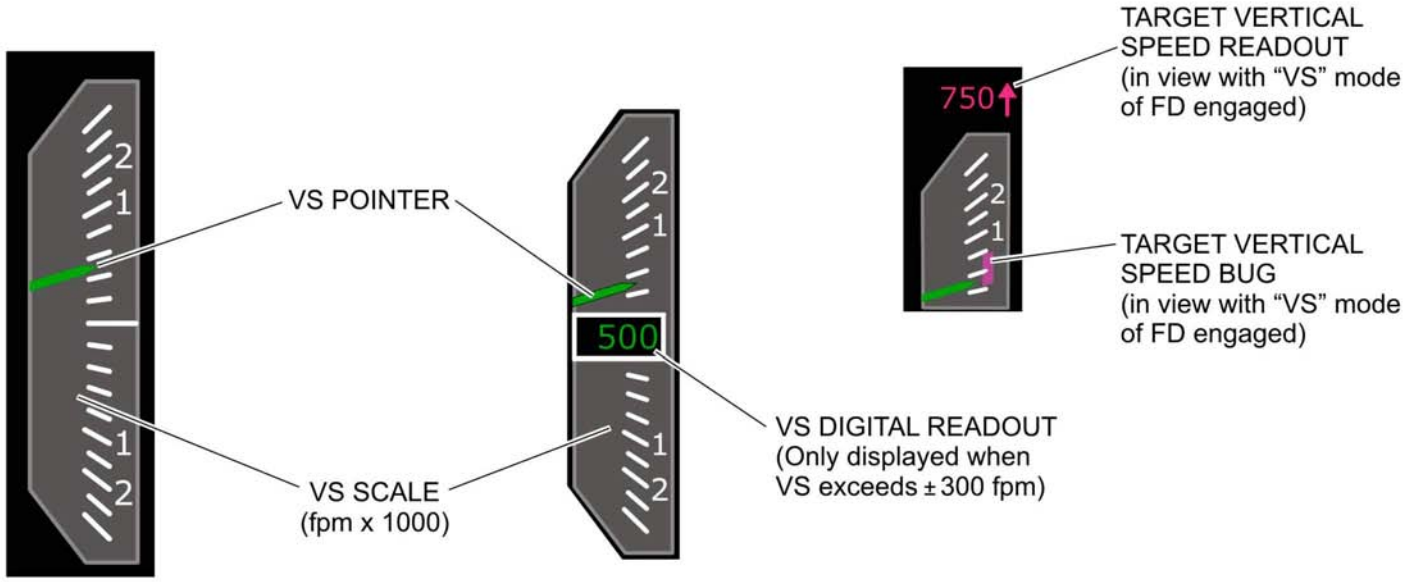


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NAVIGATION SYSTEM – IAS INDICATOR

Vertical Speed Tape

it is selected from either the onside or cross-side ADS. The VSI shows altitude rate of change in feet per minute and is comprised of a vertical speed scale, digital readout and target speed bug. The tape shows a vertical speed range between +/- 3000 fpm with a green pointer indicating the current vertical speed. The vertical speed digital readout (green) is removed from the view when the vertical speed is between -300 fpm and +300 fpm. Loss of VS information form the ADM results in the removal of the digits in the window and replaces the VS tape with a red cross.



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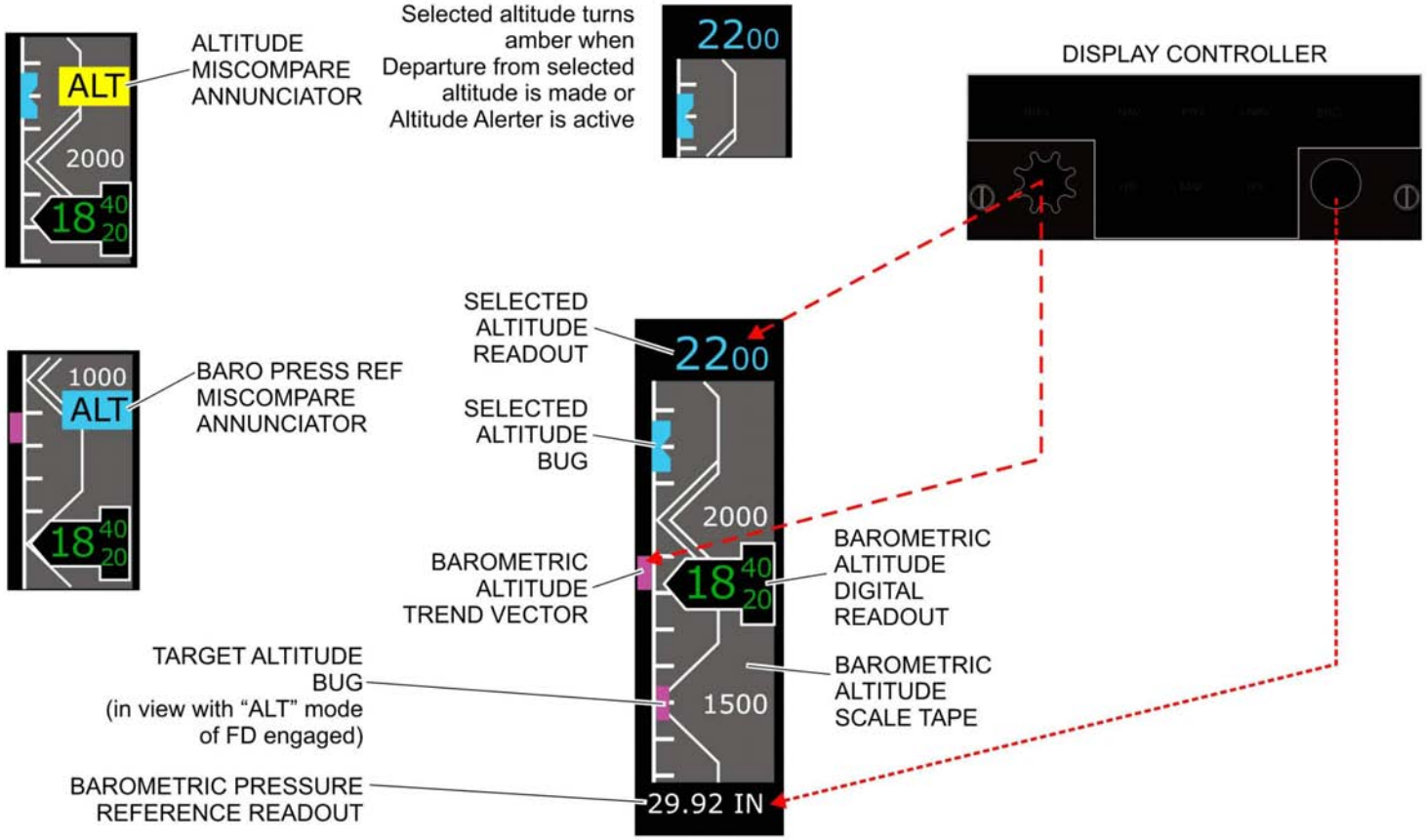
NAVIGATION SYSTEM – VS INDICATOR

Barometric Altimeter Tape

it displays the barometric altitude MSL (Mean Sea Level) from either the on-side or cross-side ADS. The tape is arranged so that a ± 550 ft window is visible all times. A double lined chevron marks each 1000 ft increment and a single lined chevron marks each 500 ft increments. A green digital readout of the current altitude is displayed in a box at the center of the tape. A magenta trend vector indicator that travels vertically along the left, outside of the altitude tape alerts the pilot to the estimated altitude the aircraft attains over a period of 6 sec given the existing rate of climb. On the tape, the low altitude awareness is also reported when the aircraft descends through 550 ft AGL. The altimeter settings are also placed in the window beneath the altitude tape.
Loss of altimeter information form the ADM results in the removal of the tape/ digits in the window and replaces the tape with a red cross. Also the altimeter setting is removed and replaced by dashes.
The Baro Set Knob is housed on the Display Controllers (on the instrument panel) and provides the following function:

- BARO:
  - turning cw the knob, the corrected barometric pressure setting on the PFDs increases
  - turning ccw the knob, the corrected barometric pressure setting on the PFDs decreases

The BARO correction supplies adjustment of the altitude calculation for local weather effects on the static pressure measurement and supplies the barometric correction inputs for the ADS.
The altitude preselect bug is a cyan notched rectangle indicator that indicates the preselected altitude set on each PFD. Rotating the ALT SEL knob on the display controller moves the bug vertically along the left inside edge of the altitude tape and changes the altitude preselect digital readout. The color of the bug matches the color of the altitude preselect digital readout at all times.



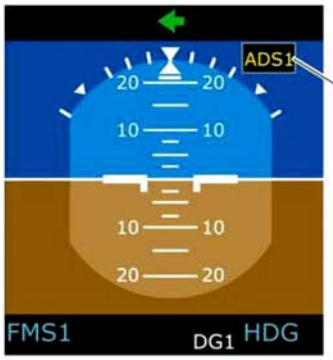
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NAVIGATION SYSTEM – BAROMETRIC ALTITUDE INDICATOR

Reversion Control Panel

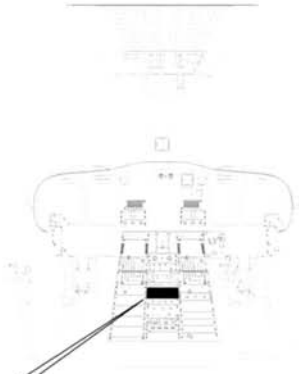
On the Reversion Control Panel (on the interseat), the ADS reversion switch provides the following function:

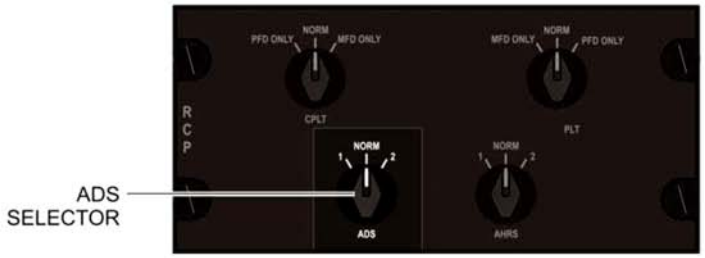
- NORM:..... the pilot and copilot displayed sources are the onside ADS.
- 1:..... the pilot and copilot displayed source is the ADS 1.
- 2:..... the pilot and copilot displayed source is the ADS2.



Air Data Source Annunciators - When displayed for the pilot and copilot are the same, ADS 1 or ADS 2 annunciates on both PFDs.

The ADS annunciator is not displayed when the displayed sources of the pilot and copilot are from the on-side ADS.





"ADS" SELECTOR
- NORM Each pilot's station displays the "On-Side" ADS data
- 1 Both pilot's stations display the ADS 1 data
- 2 Both pilot's stations display the ADS 2 data

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NAVIGATION SYSTEM – ADS REVERSION

CAPTIONS

The caution provided in the CAS message window is as follows:

1	1(2) ADS FAIL	when ADS fail and loss of the airspeed, altitude and vertical speed.
2	AVIONIC FAULT	when no Display Controller activity is detected.

ANNUNCIATORS

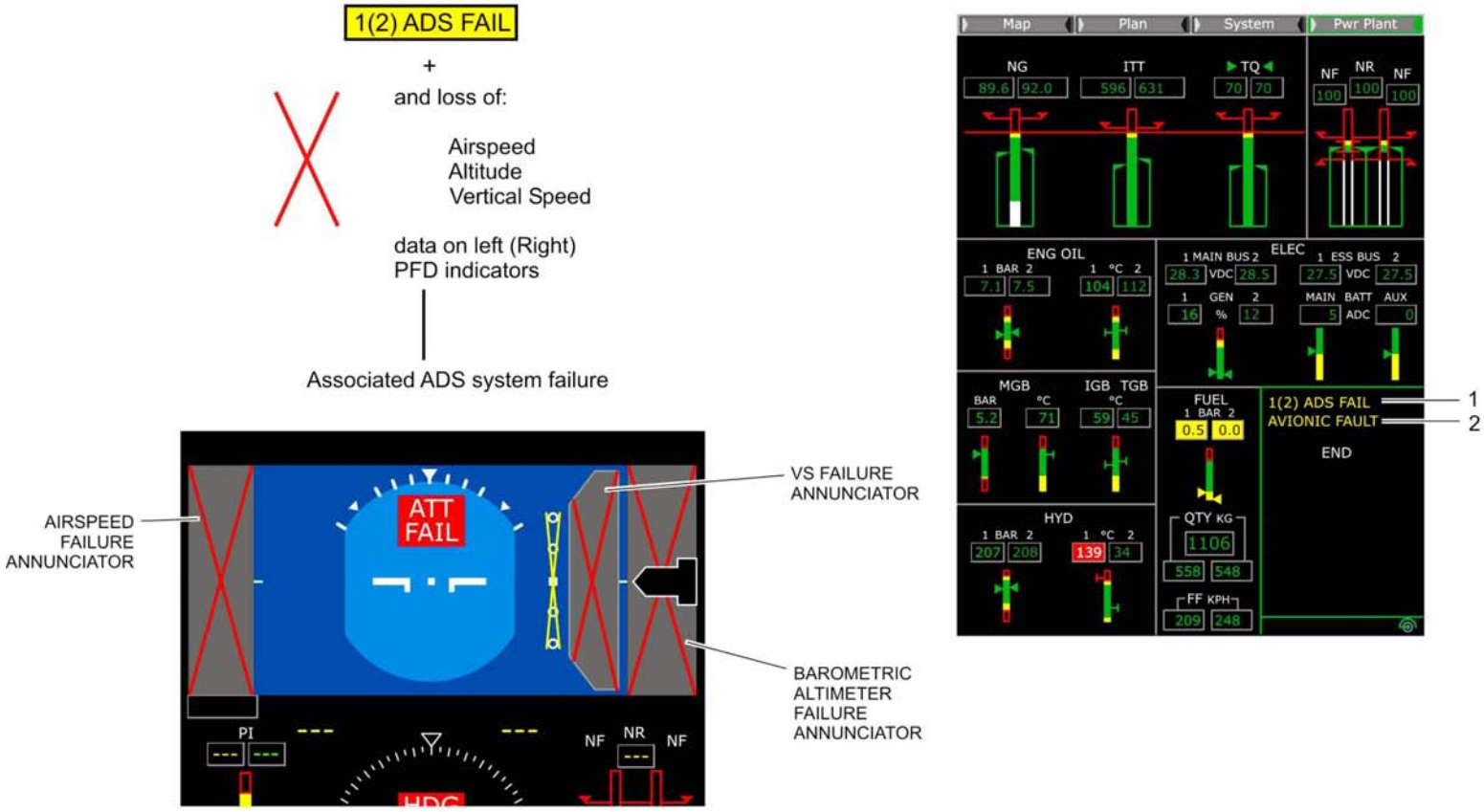
Beside, on the PFDs the following source annunciators are displayed as follows:

- Miscompare Annunciators (inverse video)
  - IAS when the indicated airspeed from the two ADS disagrees by ± 20 kts.
  - ALT when the barometric altitude from the two ADS disagrees by ± 200 feet.

NOTE A

The above annunciators flash in inverse video for 5 seconds before being displayed steady state.

- Source Annunciators
  - ADS 1 appear in amber only when the ADS reversion switch is set to 1. Both the pilot and copilot displayed source is the same.
  - ADS 2 appear in amber only when the ADS reversion switch is set to 2. Both the pilot and copilot displayed source is the same.



ICN-39-A-341000-G-A0126-01262-A-01-1

NAVIGATION SYSTEM – ADS: FAILURE INDICATIONS

LEADING PARTICULARS

- ADM NO. 1
  - Nominal voltage..... 28 VDC (MAIN BUS NO.1)
- ADM NO. 2
  - Nominal voltage..... 28 VDC (MAIN BUS NO.2)
- OAT PROBE NO.1
  - Nominal current..... 1.9 ma from the MAU NO.1
- OAT PROBE NO.2
  - Nominal current..... 1.9 ma from the MAU NO.2

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SECTION 20 - ATTITUDE AND DIRECTION

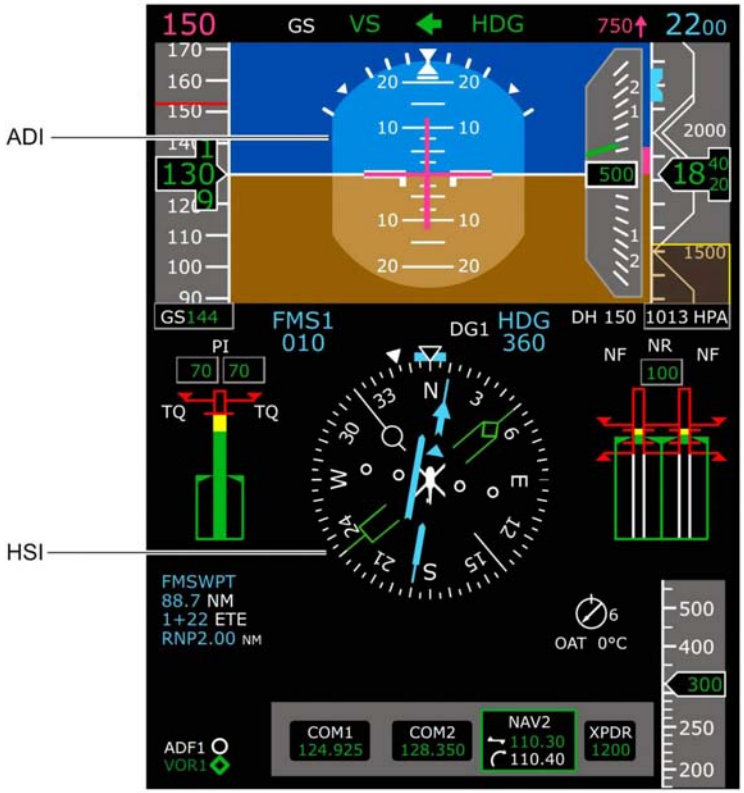
DESCRIPTION

The attitude and direction system supplies attitude heading reference data to the flight crew.
Attitude and direction has the system that follows:

- Attitude Heading Reference System (AHRS)
- Number 1 (optional) and 2 standby attitude director indicator (STBY ADI) system

ATTITUDE AND HEADING REFERENCE SYSTEM (AHRS)

The Attitude and Heading Reference System (AHRS) is an attitude inertial sensor system that supplies attitude, heading, and flight dynamics data to the Electronic Display System (EDS), Flight Management System (FMS) and Weather Radar (WXR) system. These data are then presented on the Primary Flight Displays (PFD) and on the Multifunction Displays (MFDs). The AHRS uses rate based Fiber Optic Gyros (FOGs) and micromechanical accelerometers that are aligned to the primary aircraft axes to supply an inertial measurement system. A digital computer integrates the angular rate data to supply heading, pitch, and roll. A flux valve supplies long term heading reference. The air data system (ADS) supplies true airspeed (TAS) to increase the attitude performance.
The AHRS supplies inertial altitude and vertical speed if pressure altitude is received from the air data system. The AHRS also supplies ground speed data if it is connected to a global positioning system (GPS) receiver. The AHRS receives the GPS data through a ARINC 429 interface.



PRIMARY FLYGHT DISPLAY

ICN-39-A-342000-G-A0126-01263-A-01-1

ATTITUDE AND HEADING REFERENCE SYSTEM (AHRS)

ATTITUDE HEADING REFERENCE SYSTEM (AHRS)

The AHRS has the components that follow:

- Number 1 and 2 attitude and heading reference units (AHRUs) (A9 and A10).
- Number 1 and 2 flux valves (M5 and M4).
- AHRU Calibration Programmable Read-Only Memory (CAL PROM).
- Number 1 and 2 compass control panel (PL3 and PL20).

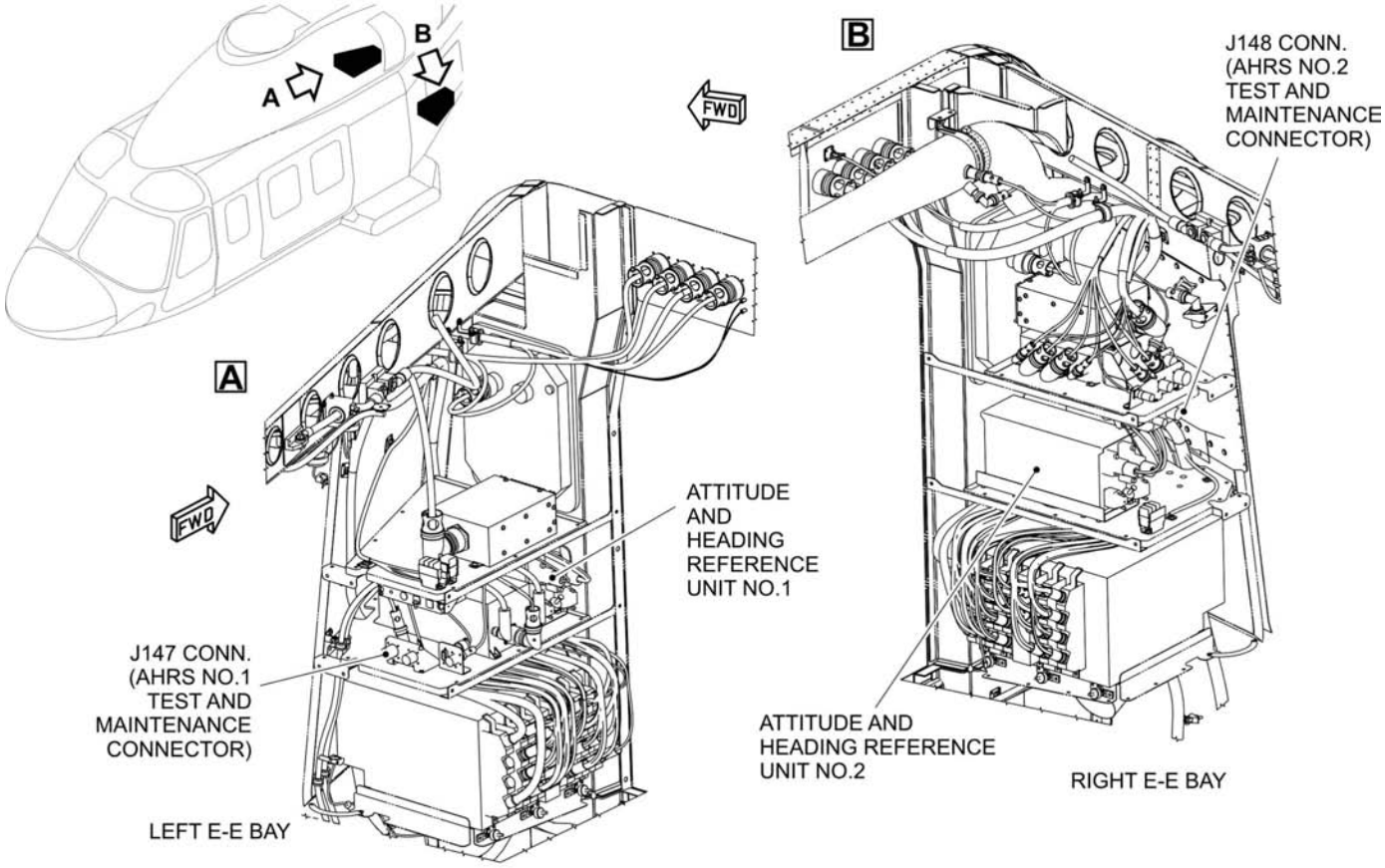
The AHRS uses rate based Fiber Optic Gyros (FOGs) and micromechanical accelerometers that are aligned to the primary aircraft axes to supply an inertial measurement system. Omission of the mechanical platform and use of FOGs eliminates any moving part within the system and thus enables a significant improvement of reliability.

AHRU

The AHRU is the main AHRS component, containing the sensors (FOGs and accelerometers), processor, power supply, and interface to the users. There is no requirement for the pilot access to the unit. It is located in the avionics compartment.
The AHRU1 (A9) and the AHRU2 (A10) are installed respectively in the left and right avionic bay (helicopters [1J]), or in left and right nose compartment (helicopters [1L]). Two hold-down knobs hold the AHRU to its mounting tray. The mounting tray includes the fan. The AHRU has the CAL PROM.
Two connectors A9P1 and A9P4 connect the AHRU1 to the MAU1, MAU2, circuit breakers CB49 and CB51, flux valve M5 and number 2 compass control panel (PL20).
Two connectors A10P1 and A10P4 connect the AHRU2 to the MAU1, MAU2, circuit breakers CB48 and CB50, number 2 flux valve M4 and number 1 compass control panel (PL3).

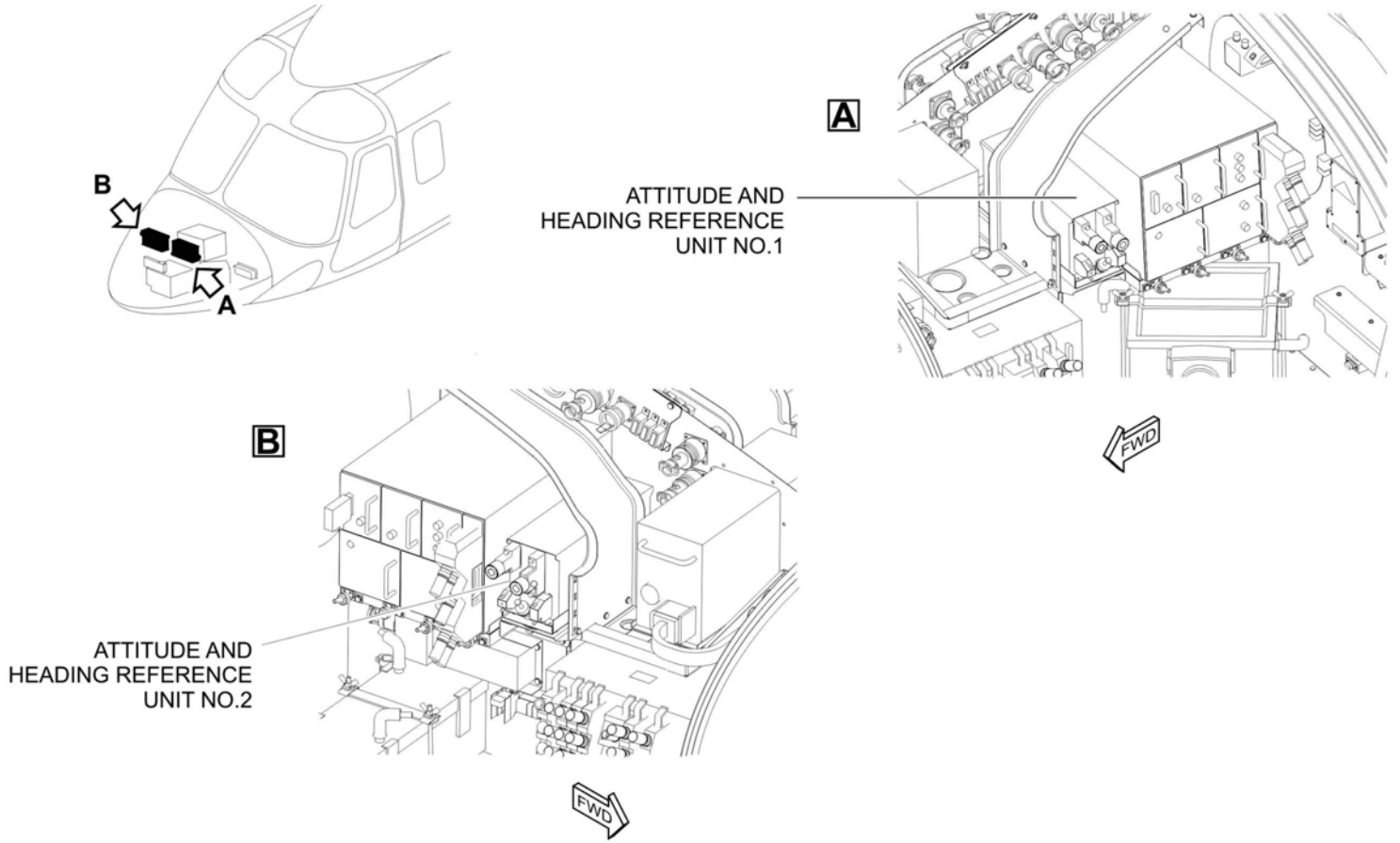
CALIBRATION PROGRAMMABLE READ-ONLY MEMORY (CAL PROM)

The Calibration Programmable Read-Only Memory (CAL PROM) is an external electronic detachable device, located at the AHRU front plate. It stores the aircraft specific information necessary to compensate the flux valve errors and the misalignment of the tray. CAL PROM stores MSU calibration data, thus recalibration is required after flux valve replacement. Two screws attach the CAL PROM to the mounting tray. Two captive screws attach the CAL PROM to the AHRU connector.



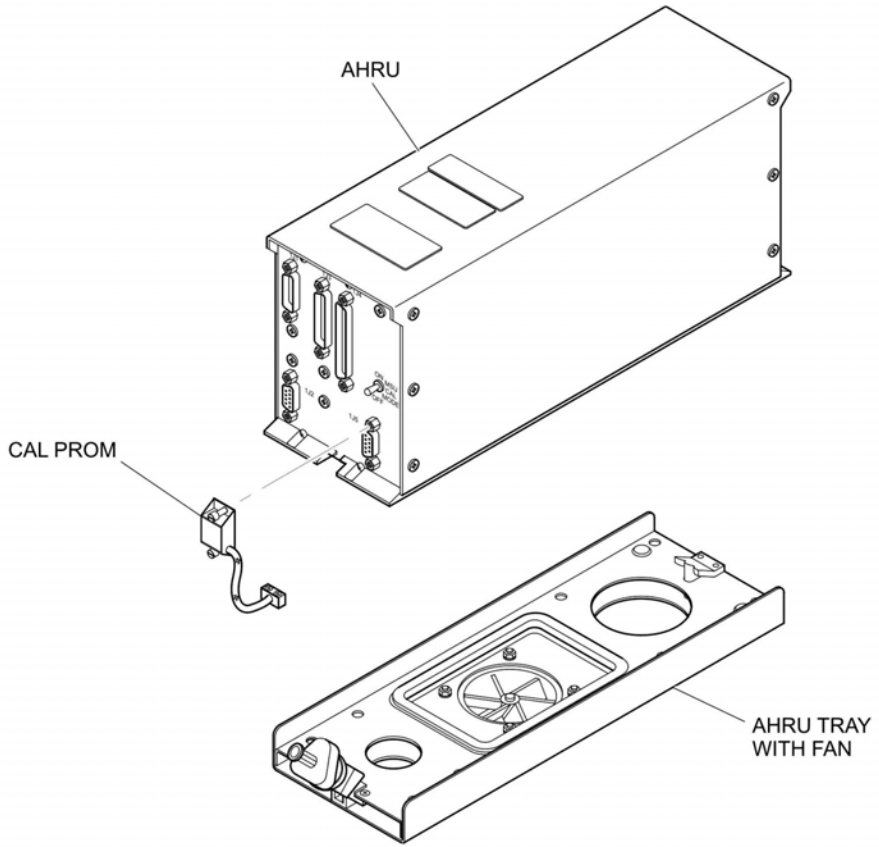
ICN-39-A-342000-G-A0126-01264-A-01-1

AHRS UNIT LOCATION (STANDARD NOSE HELICOPTER)



ICN-39-A-342000-G-A0126-01265-A-01-1

AHRS UNIT LOCATION (LONG NOSE HELICOPTER)

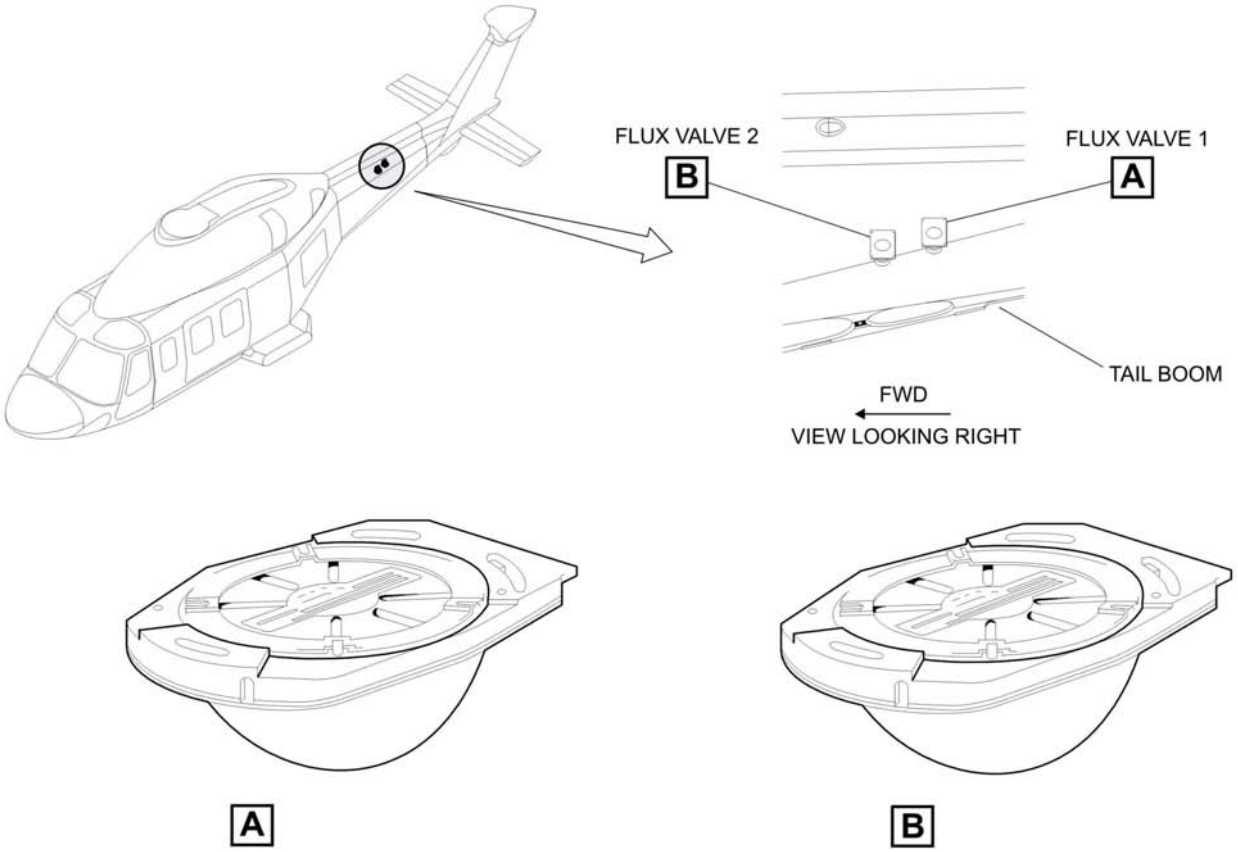


ICN-39-A-342000-G-A0126-01266-A-01-1

ATTITUTDE AND DIRECTION

FLUX VALVE

The flux valve is a magnetic sensor unit (MSU) that transmits information to the AHRU relative to the earth's magnetic field (magnetic north). It is located in the tail boom assembly. This area is selected to minimize interface by ferromagnetic materials and magnetic fields generated by the helicopter.
Number 1 and 2 flux valves are installed, respectively in the after and forward position on the left side of the rear fuselage. Three screws and three washers attach the each flux valve to its support. Connector connects the flux valves to the on side AHRU.

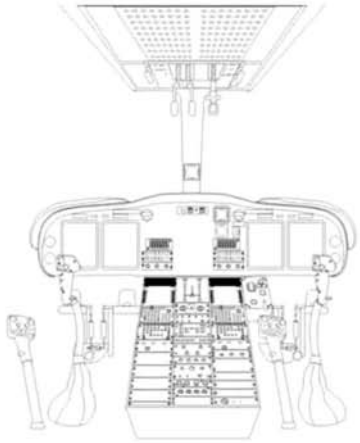


ICN-39-A-342000-G-A0126-01267-A-01-1

FLUX VALVES LOCATION

COMPASS CONTROL PANEL (CCP)

The compass control panel (CCP) enables the pilot to control the operation of the AHRS.
The Number 1 and 2 compass control panels (PL3 and PL20) are installed, respectively, in the left and right side of the interseat console in the cockpit. Two quick-release fasteners hold each compass panel in its correct position on the interseat console. The connector PL3P1 connects the Number 1 compass control panel (PL3) to the AHRU1.
The connector PL20P1 connects the compass control panel (PL20) to the AHRU2.





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AHRS CONTROL PANEL

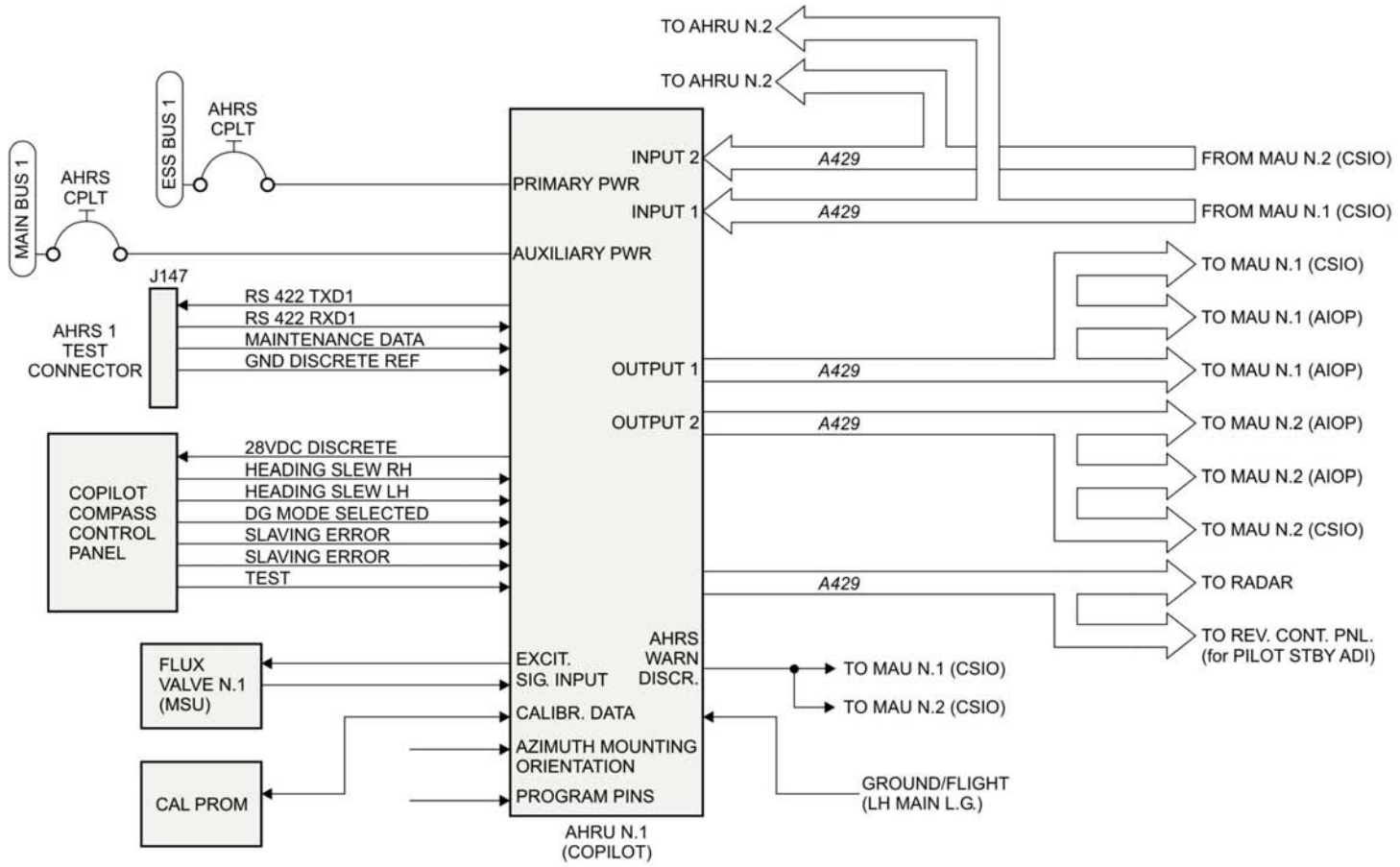
OPERATION

The AHRS uses the magnetic (MAG) mode as the standard operation mode for heading. In the MAG, or slaved mode, the AHRU is aligned with the magnetic heading output of the flux valve. If there is a difference between the heading gyro and the flux valve output, then the heading gyro is aligned with the flux valve.
In the directional gyro (DG) mode, the AHRU operates as a free directional gyro. The flux valve output to the AHRS is turned off. The AHRS has an attitude accuracy of 0.5° and a free directional gyro (DG) drift rate of 5° per hour. The magnetic heading accuracy is 2°.
Operation of the AHRS is automatic upon the application of power to the system. No operator inputs are required unless the DG mode is selected and setting of the helicopter heading is required. After application of power, the system performs an alignment, during which orientation is determined with respect to the local vertical, magnetic north and performs self-test functions. If the ambient temperature is less than +15 ˚C or more than +35 ˚C, a temperature stabilization phase of the FOGs starts. It continues for 25 seconds maximum.
As the system's sensor (FOGs and accelerometers) are used during the alignment phase to determine the earth's gravity vector and rotation, the helicopter should remain stationary during this phase in order to allow successful completion of alignment. However, helicopter motion due to the wind buffeting and cargo or passenger loading does not affect the alignment.
After the nominal 30 second alignment is finished, the AHRS enters normal operation, ATT and HDG warning flags on the cockpit displays disappear.
Depending on the dynamic conditions of the helicopter at the application of power the AHRS will enter into either an alignment on ground or an alignment in the air.
Depending of the input from Air Data System (ADS) the AHRS will enter either the Normal mode or the Basic mode.
Depending on the heading mode selection the system will go either to MAG or DG mode.
The AHRS can operate in the following operational modes:

- Alignment on the ground
- Alignment in the air
- Normal mode
- MAG mode
- DG mode
- Basic mode

ALIGNMENT ON THE GROUND

Following initial start up, the AHRS performs an alignment, which lasts 30 seconds. During the alignment ATT and HDG warning flags are in view at the cockpit displays and disappear when the alignment is finished. At this time the helicopter may be moved. In DG mode a refinement of the estimated earth's rate and free drift compensation will continue on for an additional 90 seconds as long as the helicopter is stationary.



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AHRS NO.1 ELECTRICAL SCHEMATIC

A new alignment can be initialized by pulling and resetting the AHRS circuit breakers (C/Bs) and ensuring a power interrupt of more than 0.5 seconds.

ALIGNMENT IN THE AIR

In order to perform an inflight AHRS alignment, the helicopter has to be flown straight and level for a minimum of 30 seconds for ATT and HDG to become valid. Flight straight and level means Max Pitch +/- 10°, Max Roll +/- 5°, Max Yaw Rate 1° per second, Max Lateral and Longitudinal Acceleration 0.05 g.

NORMAL MODE

Normal mode is the operational mode that the AHRS enters after the alignment is finished and if valid True Air Speed (TAS) data from the Air Data System (ADS) is available. The AHRS is capable of staying in Normal mode for up to 5 minutes after loss of valid TAS data. Both MAG and DG modes belong to the Normal mode category.

MAG MODE

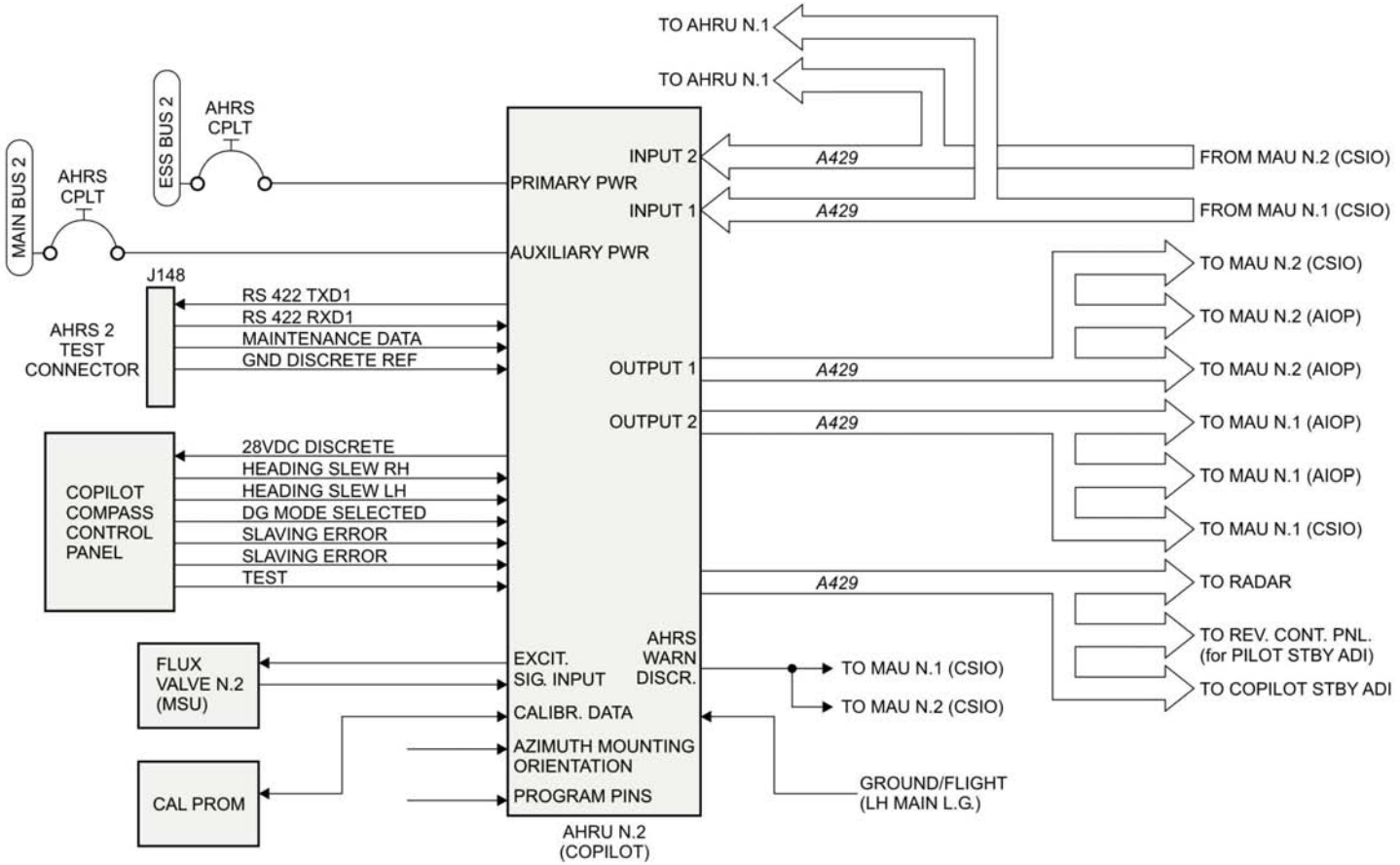
The AHRS operates in MAG mode as long as it receives valid input data from the flux valve and it is not switched manually by means of the compass control panel (CCP) into the DG mode. In this mode the AHRS is synchronized to the data from the flux valve, thus providing gyro stabilized magnetic heading information.

DG MODE

The AHRS operates in DG mode if the flux valve fails or if it is switched manually by means of the CCP into the DG mode. In this mode the AHRS operates as a free directional gyro and is required to set the desired heading into the system by means of the slew switch. The DG heading is subject to a drift, specified to be max 5° per hour.

BASIC MODE

The AHRS enters the Basic mode in case of no valid True Air Speed (TAS) data available. The attitude accuracy in this mode is better than 1°. Either MAG or DG modes may be used in the heading axis.



ICN-39-A-342000-G-A0126-01270-A-01-1

AHRS NO.2 ELECTRICAL SCHEMATIC

CONTROLS AND INDICATORS

The AHRS system control is housed on the Compass Control Panel (CCP) located on the interseat.

The CCP provides the following functions:

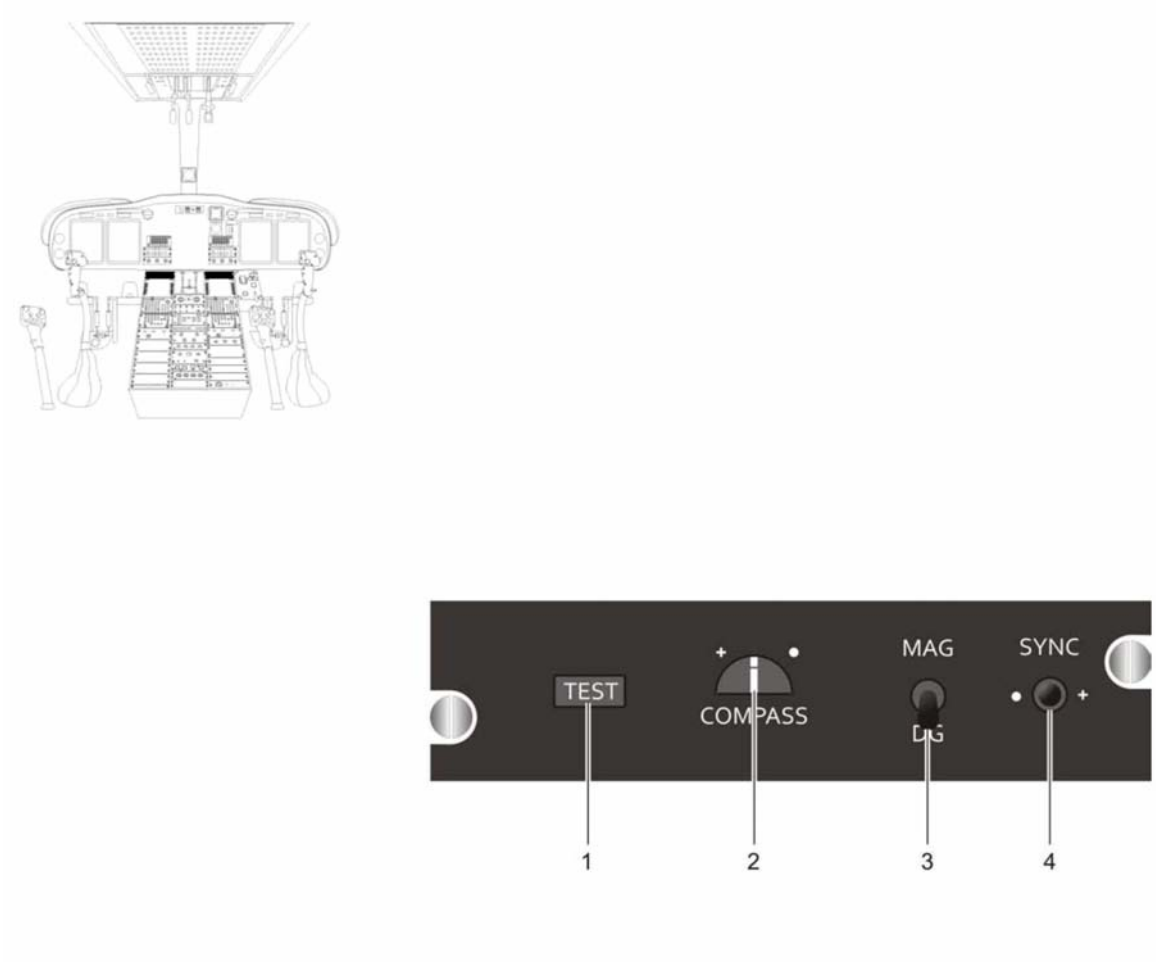
1 TEST pushbutton: ..... when pressed enables the proper AHRS 1 or 2 to activate the system self test.

2 COMPASS annunciator: ..... indicates a misalignment between the gyro output and the flux valve signal when the needle is not centered.

3 MAG/DG switch:

- MAG: ..... lift to MAG, the sub mode change the heading indicator from being driven by the directional gyro to being referenced to local magnetic north;
- DG: ..... lift to DG, the heading indicator is slaved to the directional gyro.

4 SYNC switch: ..... toggle the spring loaded switch in the direction indicated by the COMPASS annunciator, either (+) or (-) to align the gyrocompass.



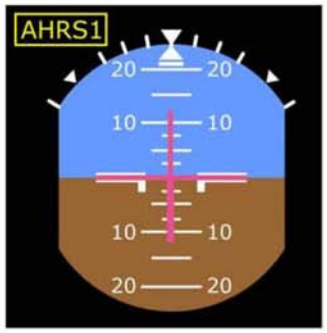
ICN-39-A-342000-G-A0126-02998-A-01-1

AHRS CONTROL PANEL

Reversion Control Panel

On the Reversion Control Panel (RCP), the AHRS reversion switch provides the following functions:

- NORMAL:..... pilot and copilot displayed sources are the onside AHRS.
- 1:..... pilot and copilot displayed source is the AHRS 1.
- 2:..... pilot and copilot displayed source is the AHRS 2.

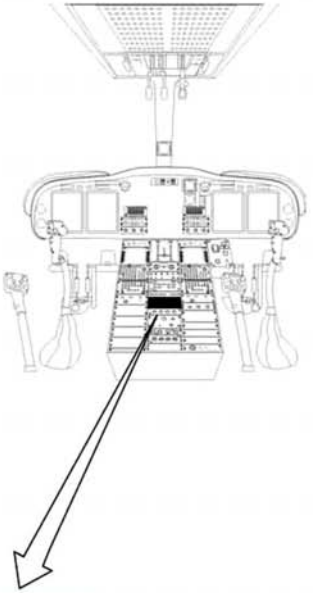


Attitude Source Annunciators - The source of attitude information, either AHRS1 or AHRS2 is annunciated at the top of the ADI left of the zero index mark on the linear roll scale.

The annunciator is not displayed when both PFD are displaying information from the on-side AHRS.

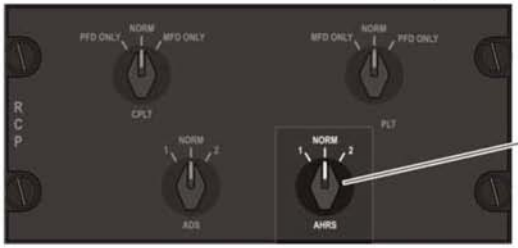
When the pilot and copilot PFDs are displaying information from the same source AHRS1 or AHRS2 (as appropriate) annunciates in amber on each PFD.

When there is a sensor failure, AHRS1 and AHRS2 are not removed.



AHRS SELECTOR

- NORM Each pilot's station displays the "On-side" AHRS data
- 1 Both pilot's stations displays the AHRS 1 data
- 2 Both pilot's stations displays the AHRS 2 data



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AHRS REVERSION

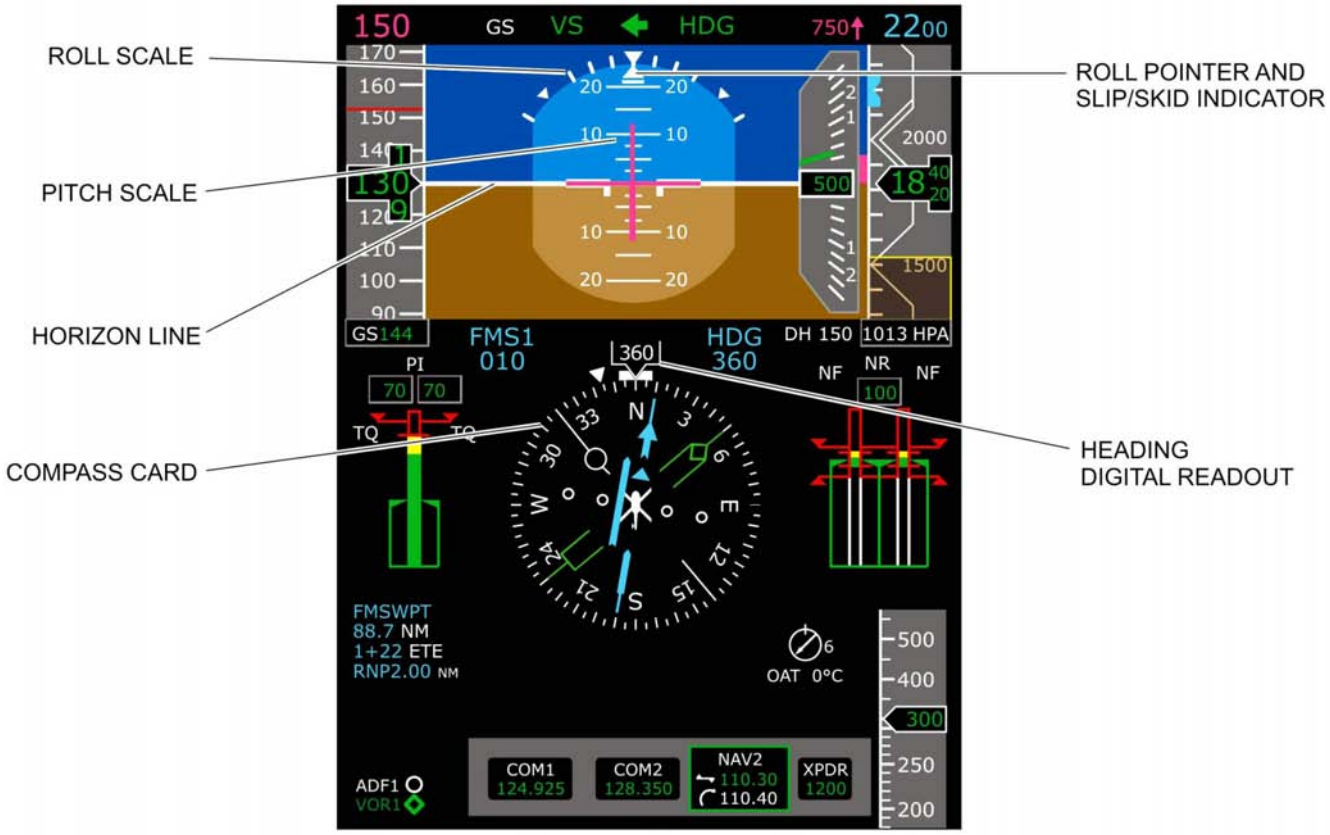
The Primary Flight Display (PFD) provides the following informations.
These data are received from either the on-side or crossside AHRS.

Attitude Direction Indicator (Adi)

ADI is depicted artificially on the attitude display of the PFD as a white, horizontal line that joins the bottom edge of the cyan field with the top edge of the brown field. The cyan field represents the sky and the brown field represents the surface of the earth. Pitching the aircraft nose up increases the area of the cyan field, cuing the pilot to a nose high attitude. The rate the cyan field increases in area is proportional to the rate of pitch change. Pitching the aircraft nose down increases the area of the brown field, cuing the pilot to a nose low attitude. The rate the brown field increases in area is proportional to the rate of pitch change. The position of the aircraft reference symbol in relation to the attitude pitch tape affords instant recognition of the number of degrees the aircraft is nose high or nose low. The aircraft reference symbol also affords bank angle changes, the magnitude of change in number of degrees and the rate at which bank angle is changing. The attitude pitch tape is between +/-20° while the roll scale is graduated with marking 10°, 20°, 30° and 60°. An inverted triangle marks 0° and 45° angle of bank positions. The bank angle and roll rate are identified on the ADI by means of a white segmented triangle referred to as the roll pointer. The bottom segment of the roll pointer is the slip/skid indicator. It indicates a slip or skid by displacing left or right of the top segment of the roll pointer. As long as the aircraft is flown in trim, the roll pointer and slip/skid indicator form a triangle. If a slip or skid is induced, the slip/skid indicator moves sideways.

Horizontal Situation Indicator (Hsi)

HSI displays the aircraft heading on the compass rose located below the attitude sphere.
The compass rose can be displayed as either a full compass or as a 45° segment of the full compass rose referred to as a compass arc, pressing HSI button on the DC. A fixed helicopter symbol is displayed at the center of the compass as an aid to visualizing the actual position of the aircraft in relation to horizontal navigation information being presented on the display. The aircraft heading is indicated by movement of the compass rose with respect to the lubber mark at the top of the compass. The aircraft heading is numerically displayed inside the heading indicator.



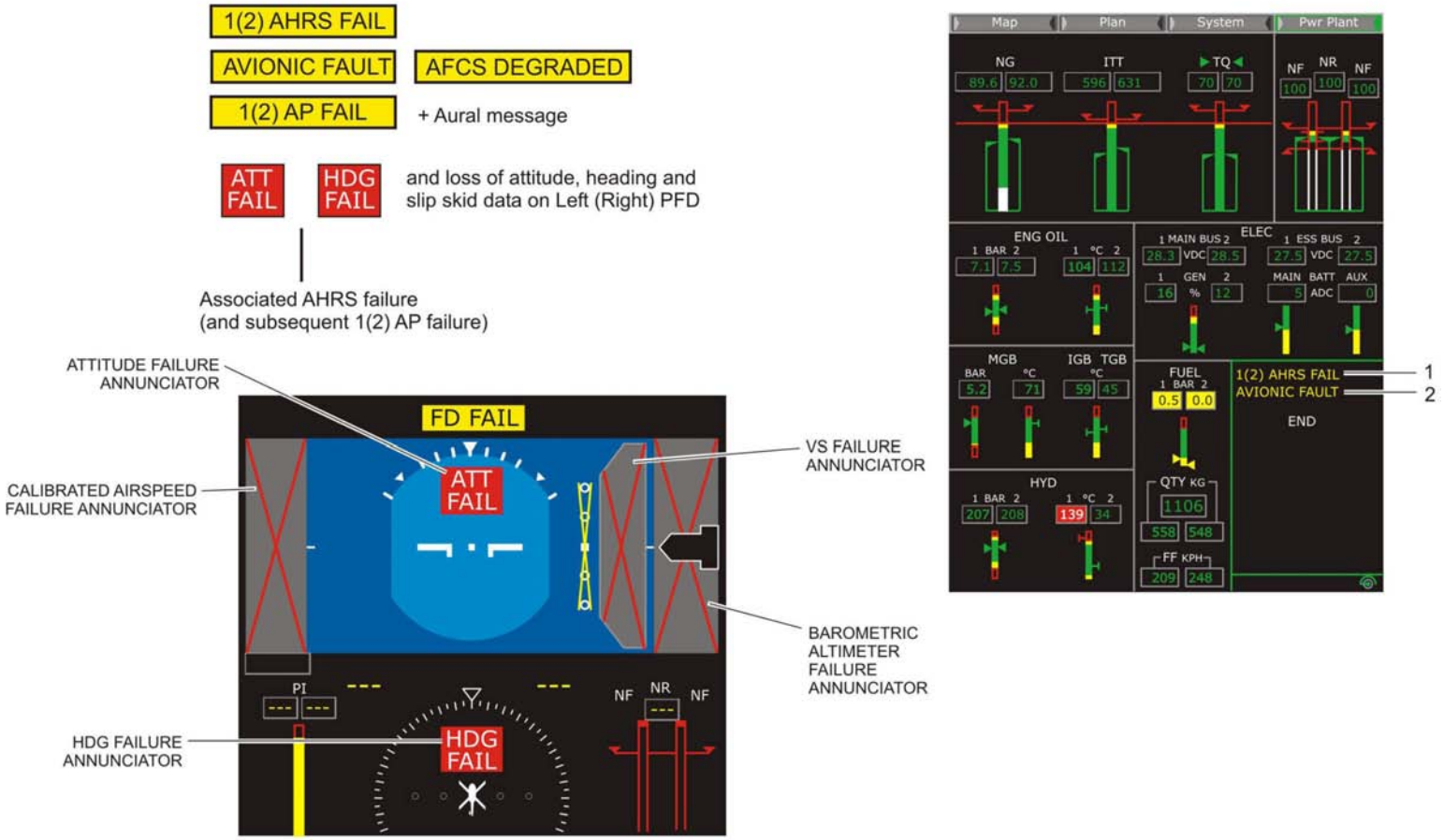
ICN-39-A-342000-G-A0126-01273-A-01-1

PFD-AHRS INDICATIONS

CAPTIONS

The caution provided in the CAS message window is as follows:

1	1(2) AHRS FAIL:	AHRS failure associated with AVIONIC FAULT; AFCS DEGRATED; 1(2) AP FAIL Cautions; Aural message; ATT FAIL and HDG FAIL Flags and loss of attitude, heading and slip skid data on PFD.
2	AVIONIC FAULT:	when the loss of communications to a single MAU is detected for the AHRS (the maintenance message in the CMC is A429/RS 422 BUS).



ICN-39-A-342000-G-A0126-01274-A-01-1

NAVIGATION SYSTEM-CAPTIONS

FLAG AND ANNUNCIATORS

The PDFs also annunciate either ATT FAIL or HDG FAIL in inverse video.

Beside, on the PFDs the following Miscompare annunciators (inverse video) are displayed:

- PITCH:..... pitch attitude from the two AHRS disagrees by ±5°.
- ROLL:..... roll attitude from the two AHRS disagrees by ±6°.
- ATT:..... pitch and roll from the two AHRS threshold exceed.
- HDG:..... heading from the two AHRS disagrees by ± 10°.

NOTE A

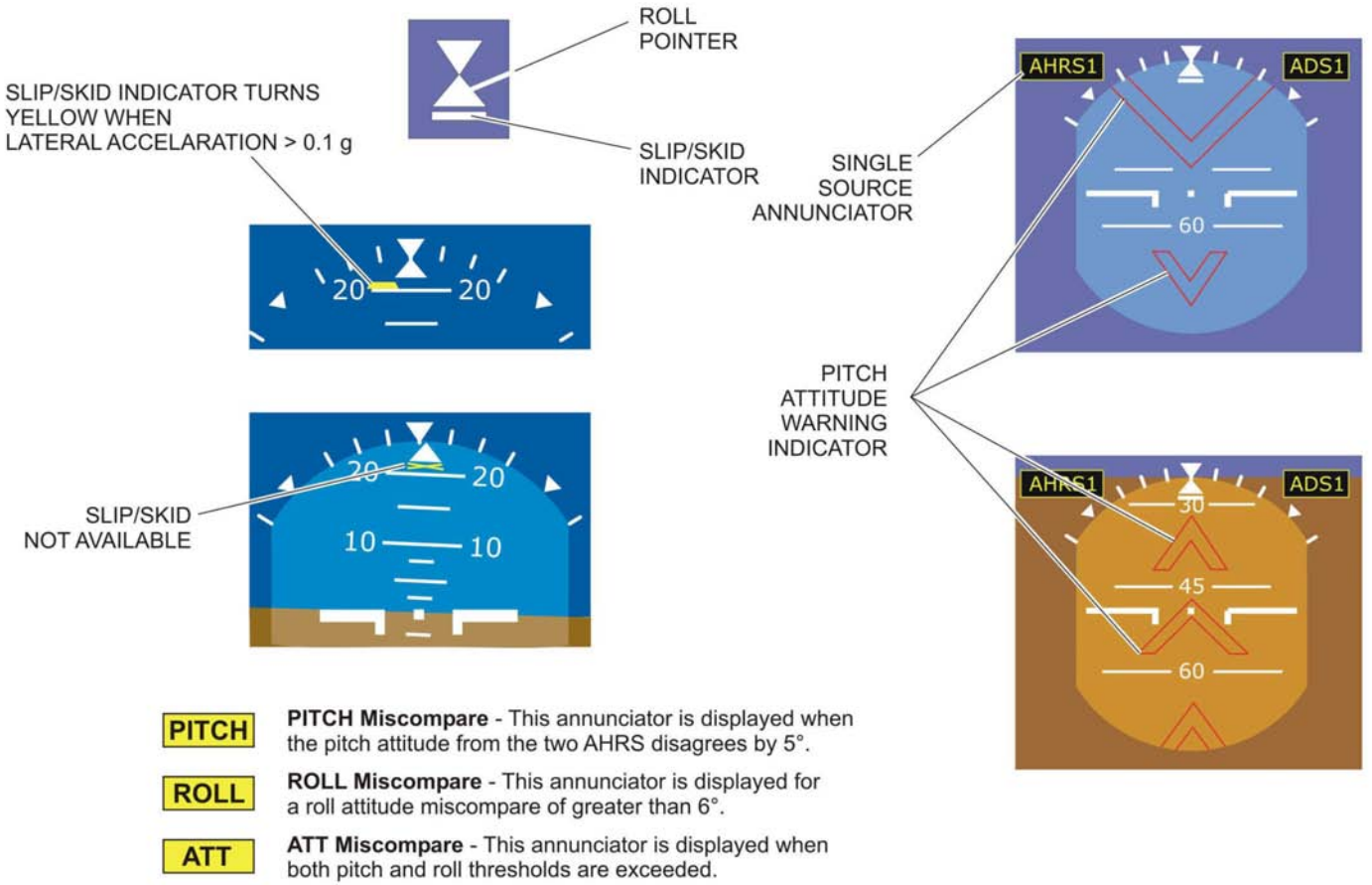
The above annunciators flash in inverse video fro 5 seconds before being displayed steady state.

Source Annunciators

- AHRS 1:..... appears in amber on both PFDs only when the AHRS reversion switch is set to 1. Both the pilot and copilot displayed sources are the same.
- AHRS 2:..... appears in amber on both PFDs only when the AHRS reversion switch is set to 2. Both the pilot and copilot displayed sources are the same.
- MAG 1:..... appears in amber only when the pilots Compass Control Panel MAG/DG switch has been selected to MAG and when the pilot and copilot heading source are the same.
- MAG 2:..... appears in amber only when the pilots Compass Control Panel MAG/DG switch has been selected to MAG and when the pilot and copilot heading source are the same.
- DG 1:..... appears in white only when the copilot Compass Control Panel MAG/DG switch has been selected to DG.
- DG 2:..... appears in white only when the pilot Compass Control Panel MAG/DG switch has been selected to DG.

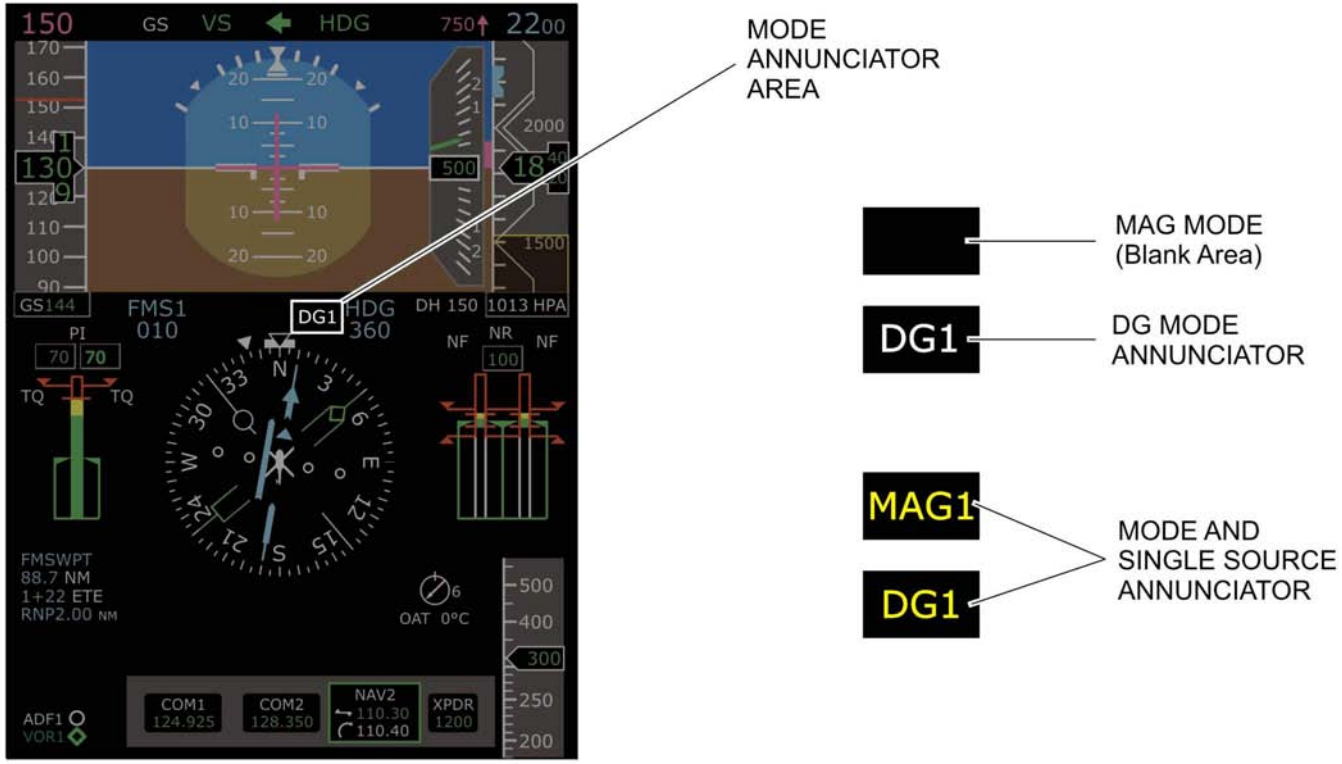
NOTE B

The DG 1 and 2 source annunciators appear in amber when the pilot and copilot heading source are the same.



ICN-39-A-342000-G-A0126-01275-A-01-1

SLIP/SKID INDICATOR AND ATTITUDE



ICN-39-A-342000-G-A0126-01276-A-01-1

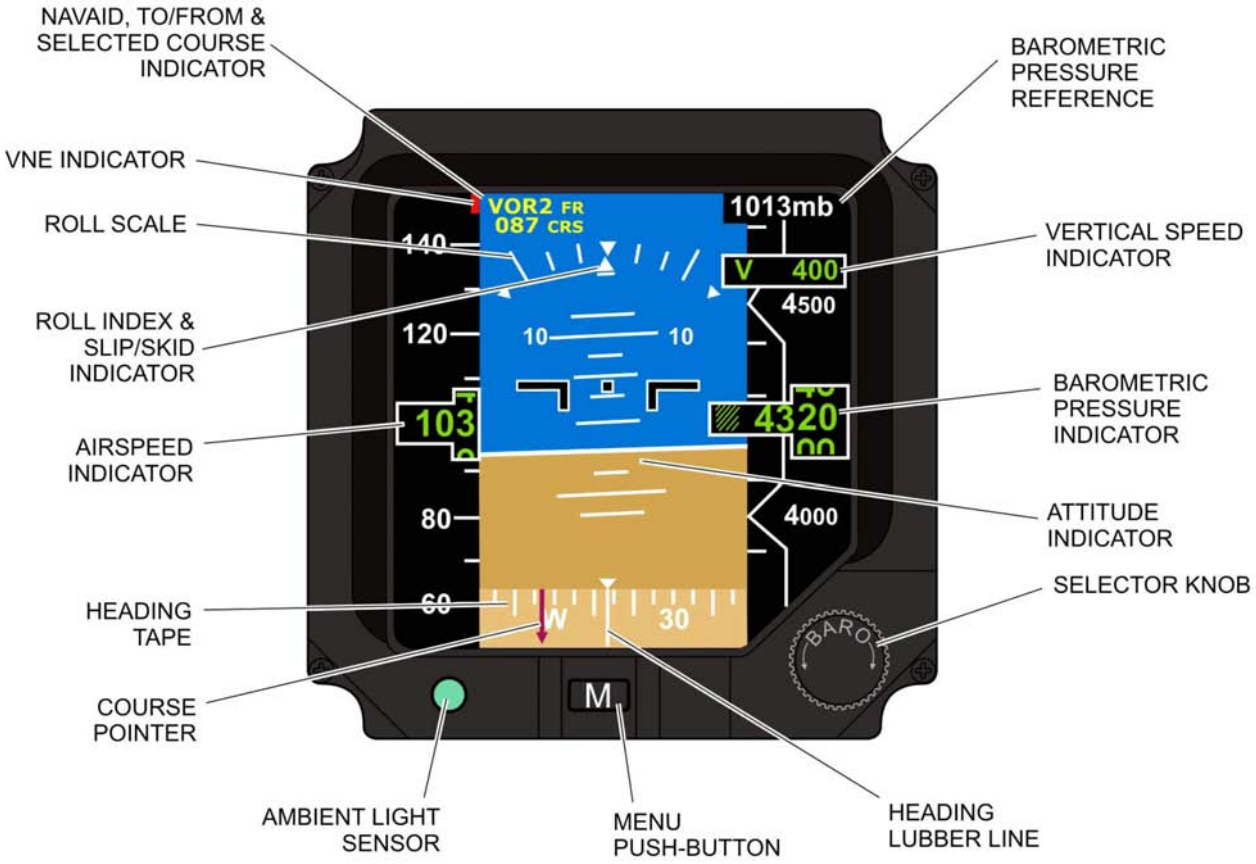
NAVIGATION SYSTEM-HEADING ANNUNCIATORS

STANDBY ATTITUDE DIRECTION INDICATOR (STBY ADI)

The attitude and direction system also comprises two standby attitude director indicators (STBY ADIs), used as backup.
The STBY ADI provides backup pitch, roll, and slip/skid information, and backup Air Data information (airspeed, altitude and vertical speed) using internal sensor systems. Also, the system contains interface capabilities that provide Heading and Navigation information.
The STBY ADI contains an inertial measurement cluster that eliminates the need for an external gyro system. It has absolutely no moving parts. It could replace all the AHRS gyro functions in case of failure. Its high-performance provides outstanding features such as precision attitude, heading, airspeed, vertical speed and altitude; reduced pilot workload; and improved flight safety. The indicator also features a full-color, active matrix LCD; dimmable fluorescent backlighting and a full range of navigation interface capabilities.
The Standby ADI has the component that follows:

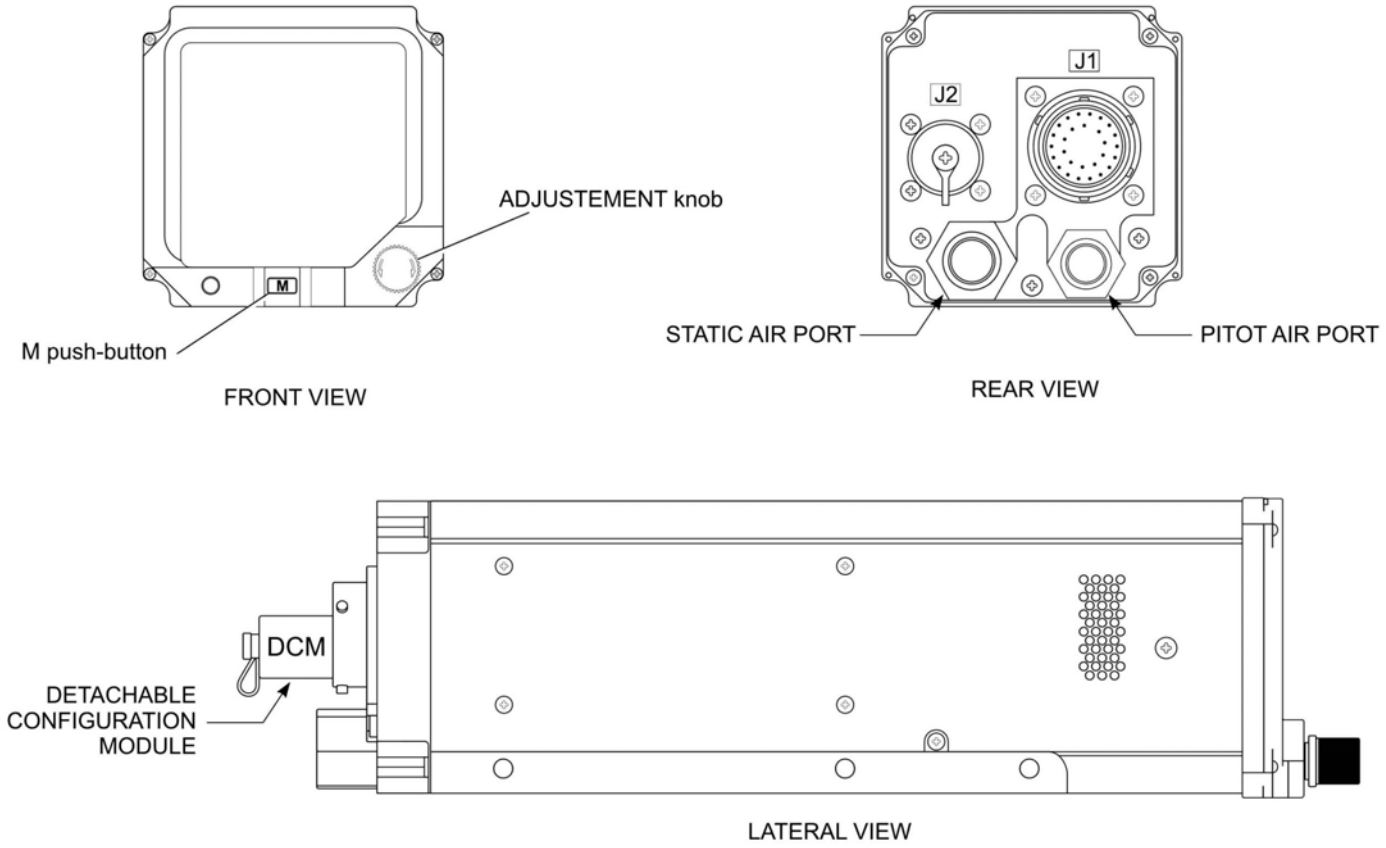
- the Detachable Configuration Module (DCM), which stores the display's format configuration.

The DCM contains sufficient memory to retain information specific to the hardware and software configuration for each installation, such as panel angle, navigation interface, aircraft heading calibration, and display format. When the instrument is removed from the aircraft, the DCM remains in the aircraft, attached to the aircraft wiring harness that mate with the indicator eliminating the need to reconfigure a replaced one.



ICN-39-A-342000-G-A0126-01277-A-01-1

STANDBY INSTRUMENT



ICN-39-A-342000-G-A0126-01278-A-01-1

STANDBY ADI

ADI STANDBY OPERATION

Power application to the STBY ADI starts an automatic process of self diagnostics prior to normal operations. After power application the display screen is blank approximately 15 seconds while the tests are performed. If no failures are detected during the power ON self test mode, the identification screen appears. If a failure is detected, the identification screen appears with either a clearly stated error message or an error code. If no failure is detected, the unit displays the ATT FAIL indication with the message ALIGNING and a completion timer/counter below the aircraft symbol. The sensor alignment reaches the normal operation mode within three minutes of applying power. During abnormal condition, such as motion during the sensor alignment mode, the indicator will reset and attempt to reach the normal operation mode within six minute of applying power. If sensor alignment is unsuccessful, the message will change to ALIGNMENT FAIL and the system will not enter the operational mode. While the indicator is operating normally, the system continues to perform diagnostic self tests to assure the crew of accurate information.

While the STBY ADI is operating normally, the system continues to perform diagnostic self-tests to assure the crew of accurate information. Invalidities are display indications that reveal a loss of information from their associated sources. The invalidities are detected during the STBY ADI Power ON Self-Test or by the continuous Background Self-Test. They may appear as either of the examples offered here dependent on unit configuration.

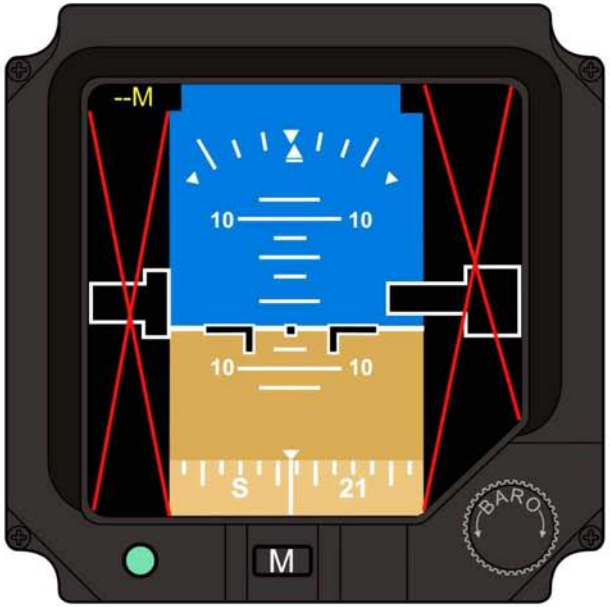
Air Data information will revert to normal display when invalidity conditions no longer exist. The STBY ADI will continue to operate, displaying information from sources that are still valid.

The Attitude Failure indication will appear as ATT FAIL in the center of the Display Screen during the system start-up process and/or when the Background Self-Test detect erroneous attitude information. Heading Tape Invalidity and Attitude Failure will appear as either of the examples offered here dependent upon unit configuration. The Heading Tape will always appear invalid simultaneously with ATT FAIL. Heading and Attitude Invalidities (internal sensor source) must be reported. Air Data and Navigation information will continue to display if those sources are still valid.



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STANDBY ADI CONFIGURATION PAGES



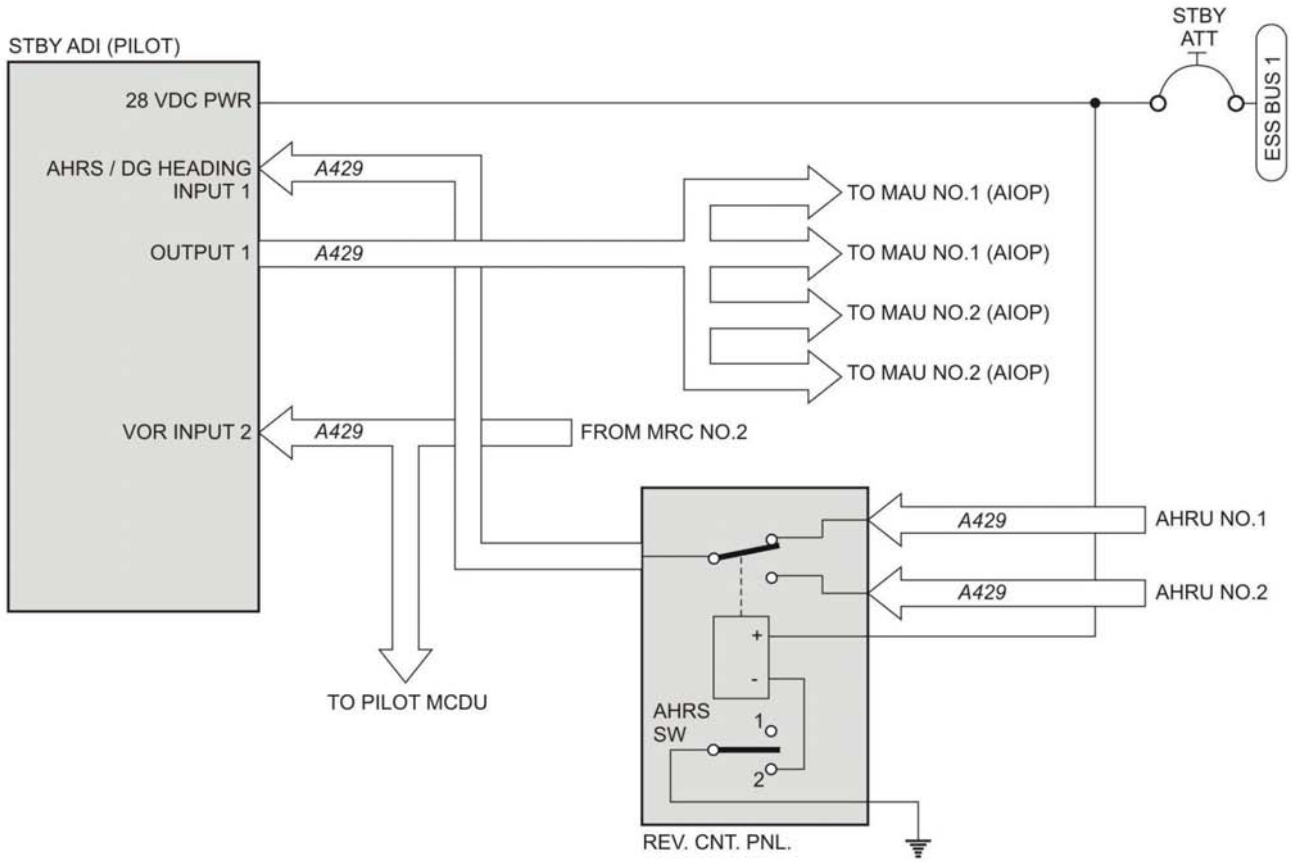
AIRSPEED AND
ALTITUDE INVALIDITY



ATTITUDE FAILURE
AND HEADING INVALIDITY

ICN-39-A-342000-G-A0126-01280-A-01-1

STANDBY ADI-AIRSPEED, ALTITUDE, ATTITUDE AND HEADING FAILURE



ICN-39-A-342000-G-A0126-01281-A-01-1

PILOT STBY ADI ELECTRICAL SCHEMATIC

STAND BY ADI CONTROLS

The STBY ADI controls are as follows:

- M push button..... when pressed, the menu is available along the bottom portion of the display.
- Adjustment knob..... when rotate, it allows the scroll through the submenu item and highlight the item.

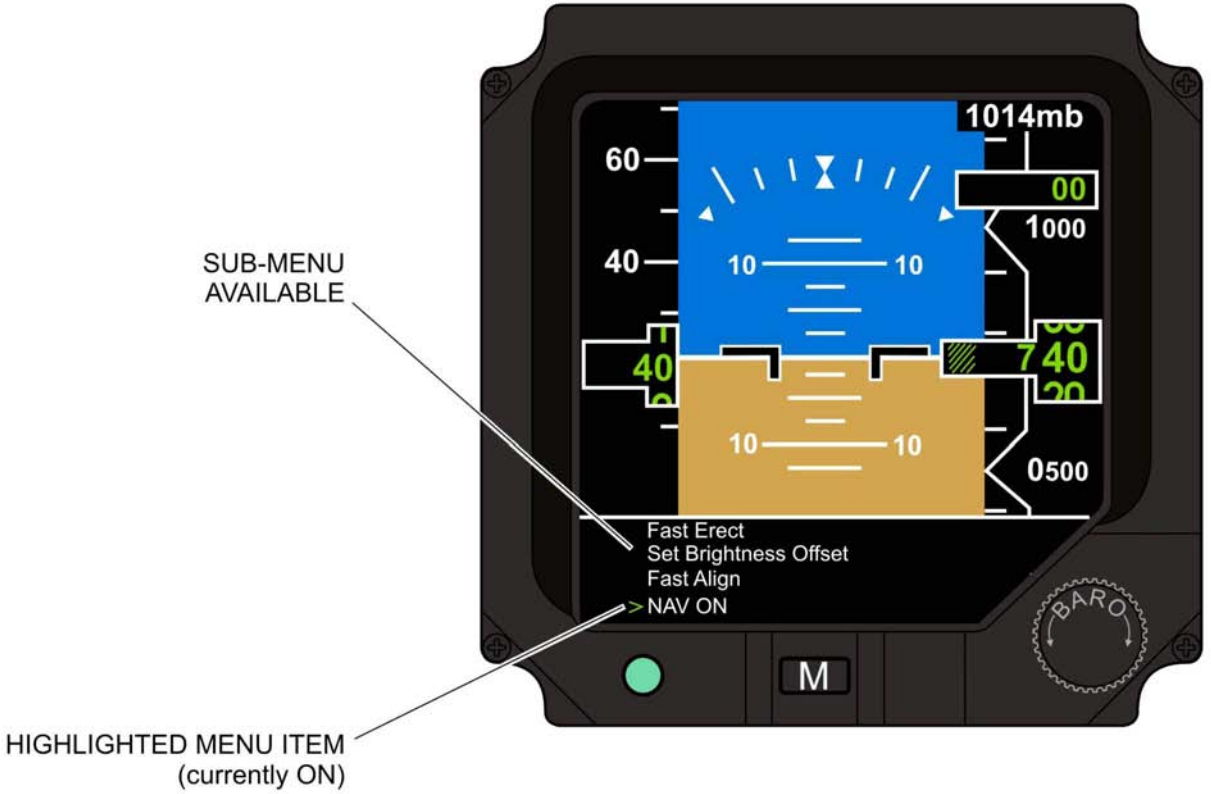
Menu item: FAST ERECT, SET BRIGHTNESS OFFSET, FAST ALIGN, NAV (ON/OFF), NAV MODE, SET CRS, CRS AUTO CENTER, BARO TYPE.

- FAST ERECT:..... press knob to initiate;
- SET BRIGHTNESS OFFSET: ..... press knob for submenu, rotate knob adjust, press knob to finish;
- FAST ALIGN:..... press knob to initiate;
- NAV (ON/OFF):..... press knob to toggle for opposite of current condition;
- NAV MODE:..... press knob for sub-menu, rotate knob to select mode, press knob to finish;
- SET CRS:..... press knob for sub-menu, rotate knob to set course, press knob to finish;
- CRS AUTO CENTER:..... press to initiate;
- BARO TYPE:..... press knob for sub-menu, rotate knob to select type, press knob to finish.

The AHRS provides continuously heading information with respect to the earth magnetic field.

NOTE C

When on the RCP, the AHRS reversion switch is set to 2, the AHRS NO.2 is connected to the PILOT STBY ADI.



ICN-39-A-342000-G-A0126-01282-A-01-1

STANDBY ADI – CONTROLS

LEADING PARTICULARS

AHRU NO. 1

- Nominal voltage (primary)..... 28VDC (ESS. BUS NO.1)
- Nominal voltage (auxiliary)..... 28VDC (MAIN BUS NO.1)
- Maximum nominal current..... 2.5 A

AHRU NO.2

- Nominal voltage (primary)..... 28VDC (ESS. BUS NO.2)
- Nominal voltage (auxiliary)..... 28VDC (MAIN BUS NO.2)
- Maximum nominal current..... 2.5 A

FLUX VALVE NO.1

- Nominal voltage..... 12 VRMS / 400 Hz from AHRU NO. 1

FLUX VALVE NO.2

- Nominal voltage..... 12 VRMS / 400 Hz from AHRU NO. 1

PILOT ADI STBY

- Nominal voltage..... 28VDC (ESS. BUS NO. 1)
- Maximum power consumption..... 50W warm up for the first 5 min of operation
- Normal running consumption..... Not exceed 23,5 W (with heaters, fan and thermal electrical cooler off)

REVERSION CONTROL PANEL

- Nominal voltage..... 28VDC (ESS. BUS NO. 1)

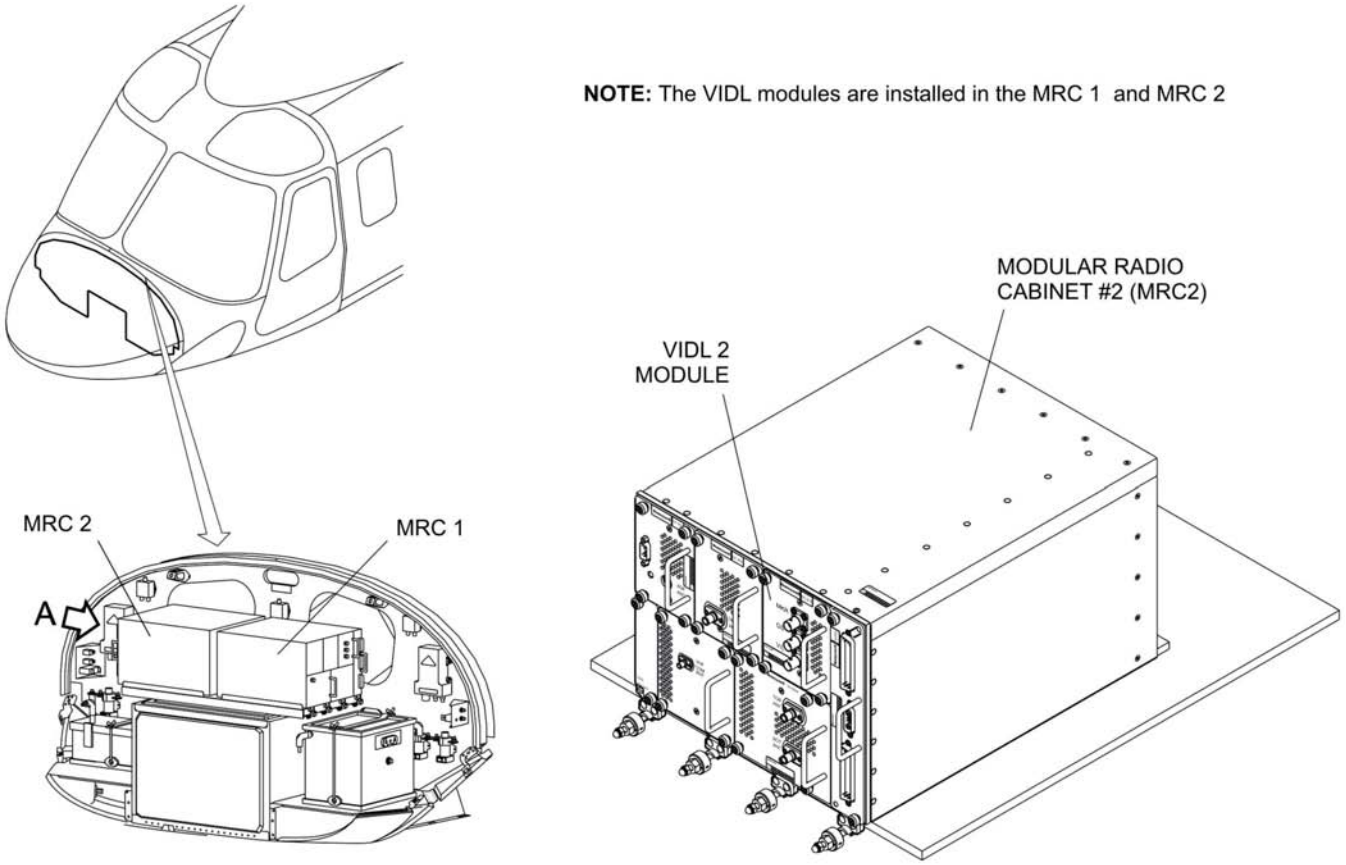
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SECTION 30 - LANDING AND TAXIING AIDS

VOR/ILS DATA LINK (VIDL) SYSTEM

The VIDL (VOR/ILS Data Link) system is a remote-controlled integrated radio system.
The VIDL system is an airborne navigation receiver that operates in VOR (Very High Frequency Omnidirectional Radio Range) and ILS (Instrument Landing System).
The VIDL system is part of the modular radio system installed in each MRC.
The VOR function supplies bearing in degrees to and from the ground station. The pilot and copilot use this function for navigation of the aircraft. The VOR function contains the VOR airborne receiver, VOR/LOC antenna and the VOR ground station transmitter. The VOR airborne receiver is part of the VIDL receiver module.
The ILS supplies approach and landing navigation guidance data. The ILS is divided into the LOC (Localizer), GS (Glideslope), and MKR (Marker) beacon functions. These functions give azimuth, elevation angular deviation, and discrete position fixes related to the runway threshold.
The LOC function contains the airborne LOC receiver, VOR/LOC antenna, and ground station transmitter.
The GS function has the GS airborne receiver, GS antenna, and ILS GS ground station transmitter.
The MKR beacon function contains the airborne MKR beacon (MB) receiver, a MKR beacon (MB) antenna, and a ground station transmitter. A beacon transmitter can be a component of an ILS system or a component of an airway marker.
The LOC, GS, and MKR beacon receivers are part of the VIDL receiver module.
The VIDL system has the components that follow:

- VIDL Receiver Modules
- VOR/LOC Antenna
- GS Antenna
- MB Antenna.
- MB coupler (CP3)
- VOR coupler (CP1)
- GS coupler (CP2)

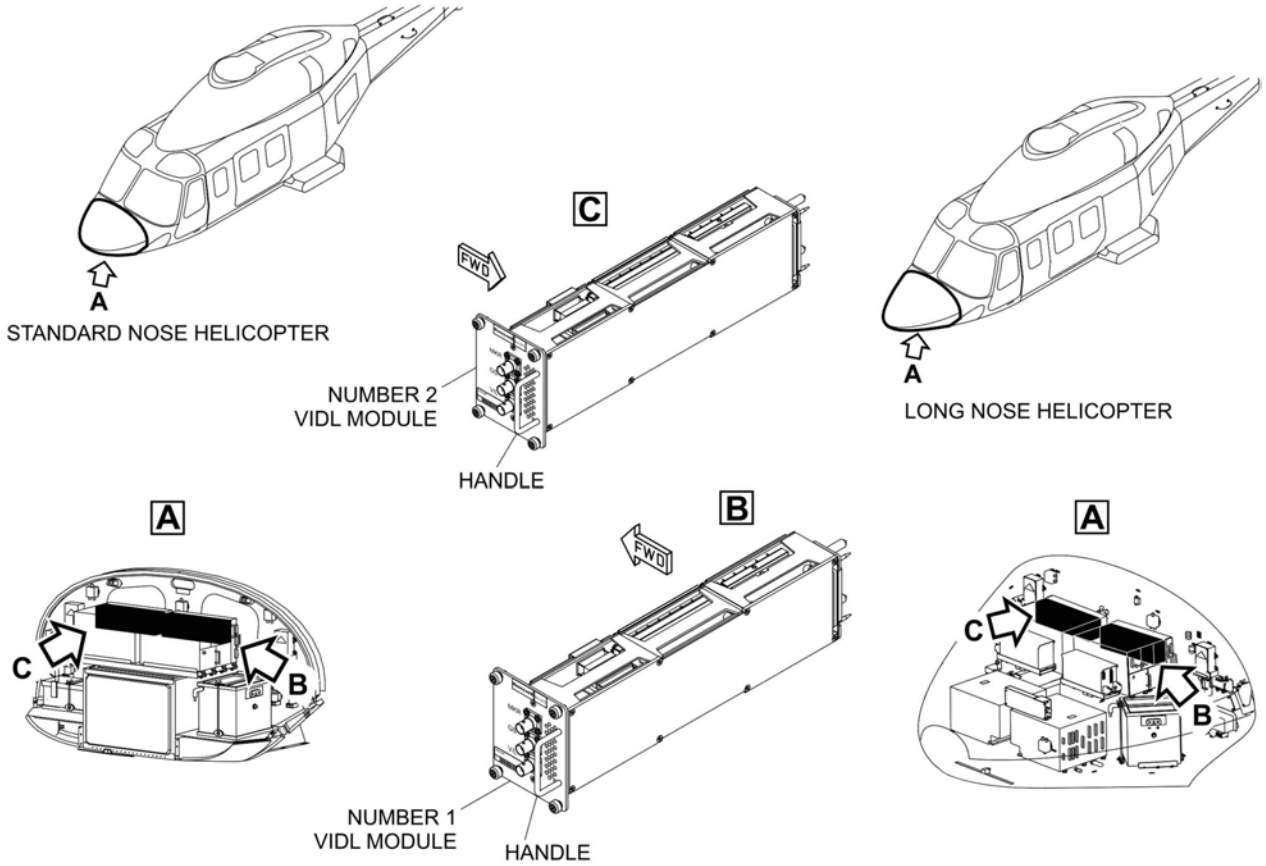


ICN-39-A-344000-G-A0126-01284-A-01-1

MRC 1 – 2 LOCATION

VIDL RECEIVER MODULES

The VIDL1 and the VIDL2 modules are installed respectively in the MRC1 and in the MRC2, in the left and in the right side of the nose compartment bay.
A handle is installed on each module front to make its installation and removal easier. Four captive screws hold each VIDL module to the MRC. Three connectors connect each VIDL module to the MB coupler, the VOR coupler and the GS coupler.
Electrical power is supplied to VIDL1 module through the circuit breaker CB23 and to VDL2 module through the CB18.
The VIDL receiver module is part of the modular radio system. The VIDL module includes a VOR receiver, LOC receiver, GS receiver and MB receiver. The Course Deviation Indicator (CDI) and Horizontal Situation Indicator (HSI) on the Primary Flight Displays (PFDs) show the guidance data from these receivers to the pilot and copilot. The VOR receiver uses signals transmitted from the VOR ground stations to find the bearing of the aircraft in relation to the ground transmitter.The VOR receiver communicates with frequency controls and displays, such as modular avionics units (MAU). It receives and demodulates amplitude modulated (AM) voice audio and sends it to the audio system. The VOR receiver also supplies fault history and status data to the maintenance system. The VOR/LOC ground stations transmit a 1020 Hz Morse code identification tone for the VOR and LOC receivers. The LOC receiver is the part of the ILS that supplies the lateral guidance in relation to the runway centerline of an approach. The LOC receiver receives localizer signals transmitted from ground stations. The LOC receiver uses these localizer signals to identify when the aircraft is at the middle of the runway approach centerline. The GS receiver is the part of the ILS that supplies the vertical guidance to the pilots and/or autopilots in relation to the glidepath of an approach. The GS receiver uses data transmitted from a ground station transmitter to identify when the aircraft is on the glidepath. The MKR beacon receiver is a component of the ILS that receives three MKR beacon tones at 75 MHz from a ground station transmitter. The receiver uses these MKR beacon tones to supply runway-threshold-distance data. The MKR beacon receiver sends this data to the PFDs to show the stable distances from the runway on the approach centerline for the pilots.

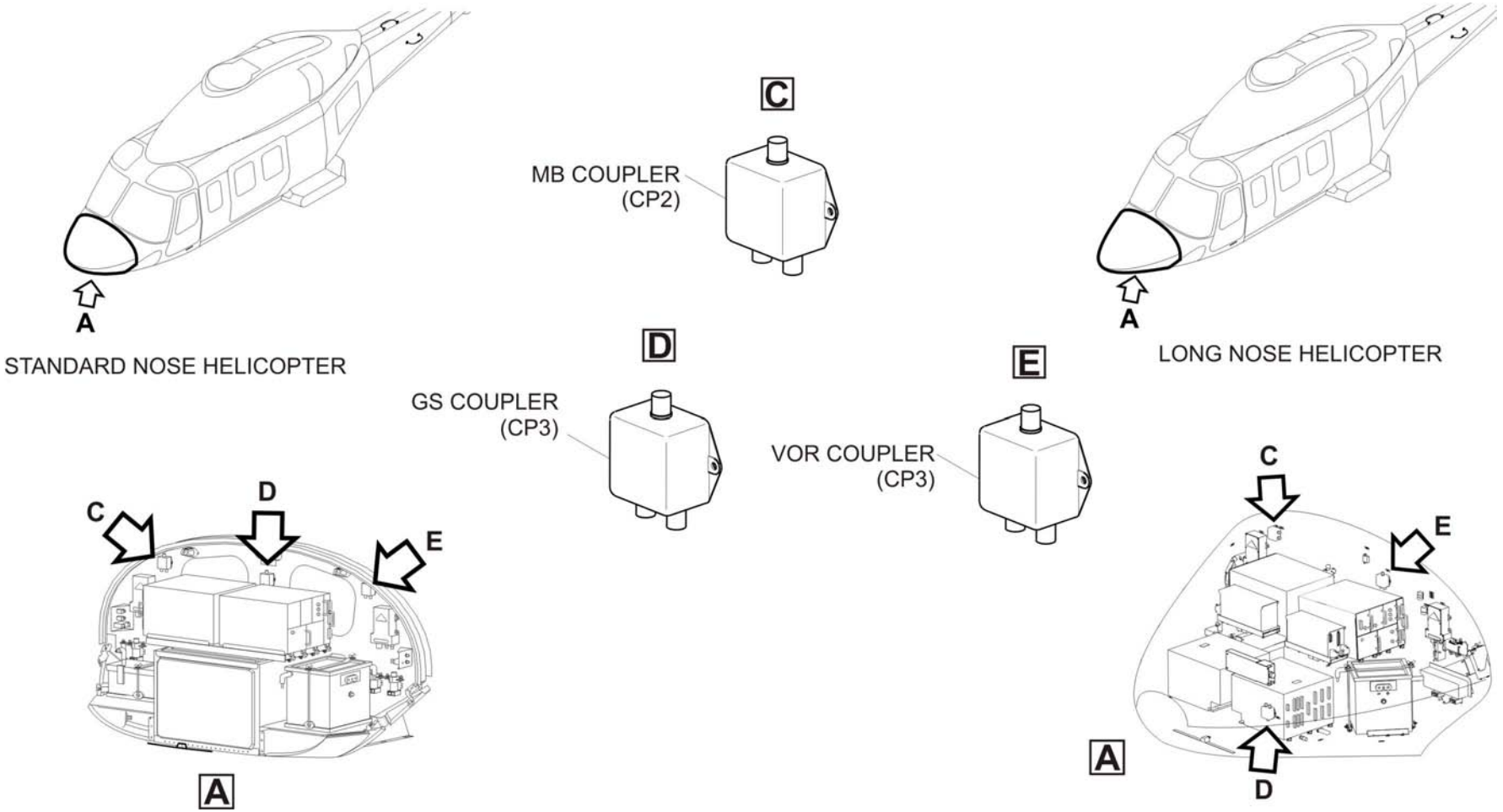


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VIDL RECEIVERS MODULES LOCATION

MB, VOR AND GS COUPLERS

The MB coupler is installed in the right side of the nose compartment. Two nuts and two washers attach the MB coupler to the nose structure. The MB coupler CP3 has three connectors that connect the MKR antenna (E6) to the VIDL modules.
The VOR coupler is installed in the left side of the nose structure. Two nuts and two washers attach the VOR coupler to the nose structure. The VOR coupler CP1 has three connectors. The coupler connects the VOR/LOC antenna (E5 E14) to the VIDL modules.
The GS coupler is installed at center in the nose radome of the helicopters [1J] (on top of the MRCs) or on the left side of the bottom of the nose radome in the helicopters [1L]. Two nuts and two washers attach the GS coupler to the nose structure. The coupler CP2 connects the GS antenna (E7) to the VIDL modules.

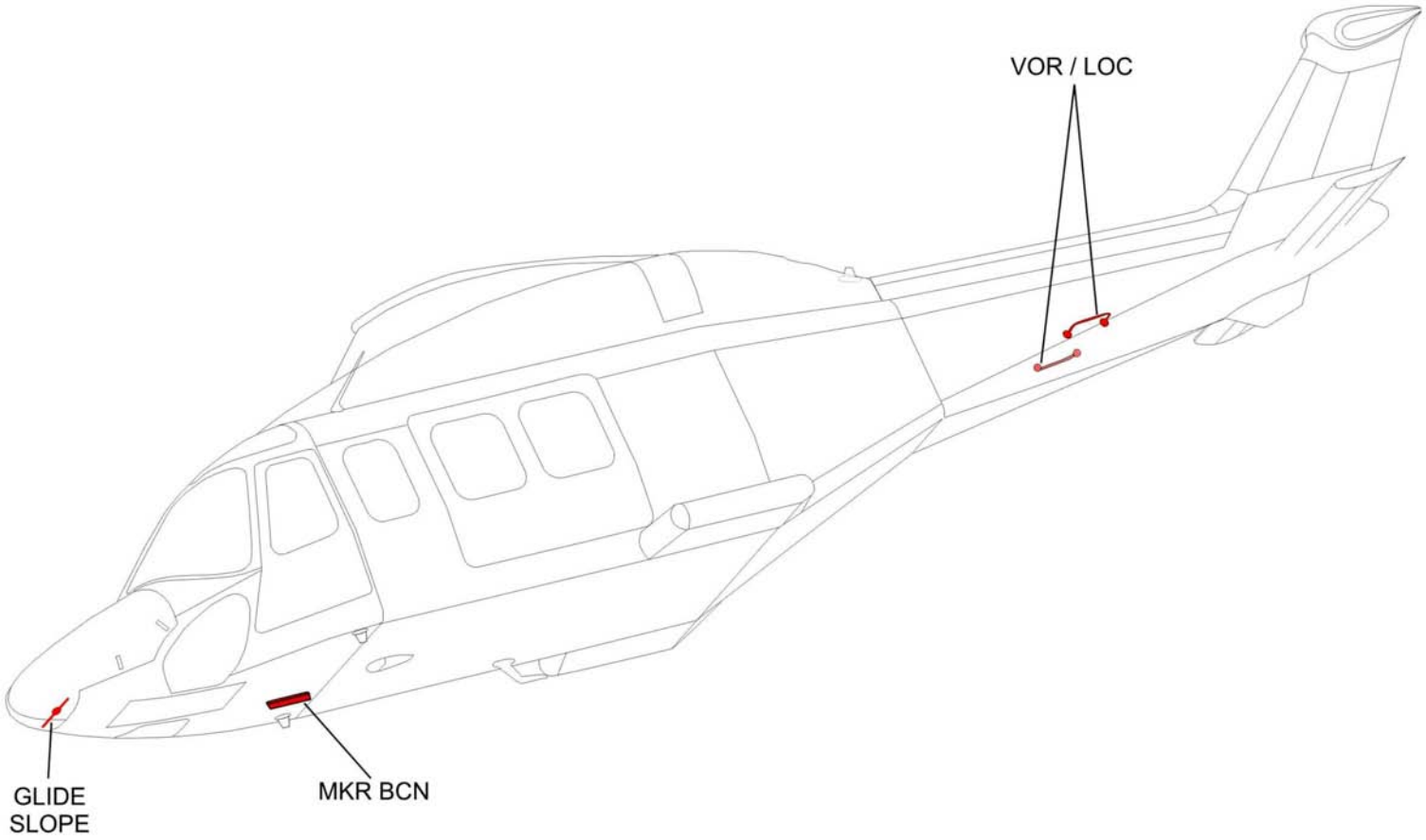


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MB, VOR AND GS COUPLERS LOCATION

VOR/LOC, GS AND MKR ANTENNAS

The VOR/LOC antenna is installed in the left and right side of the tail section. Eight screws attach each part of the VOR/LOC antenna to the tail structure. A connector E5P1 connects the VOR/LOC antenna to the VOR coupler CP1.
The GS antenna is installed in the helicopter nose. Two bolts and two washers attach the GS antenna to the structure. A connector E7P1 connects the GS antenna to the GS coupler CP2.
The MKR antenna is installed on the left side of the bottom fuselage. Three nuts attach it to the structure. A connector E6P1 connects the MKR antenna to the MB coupler CP3.



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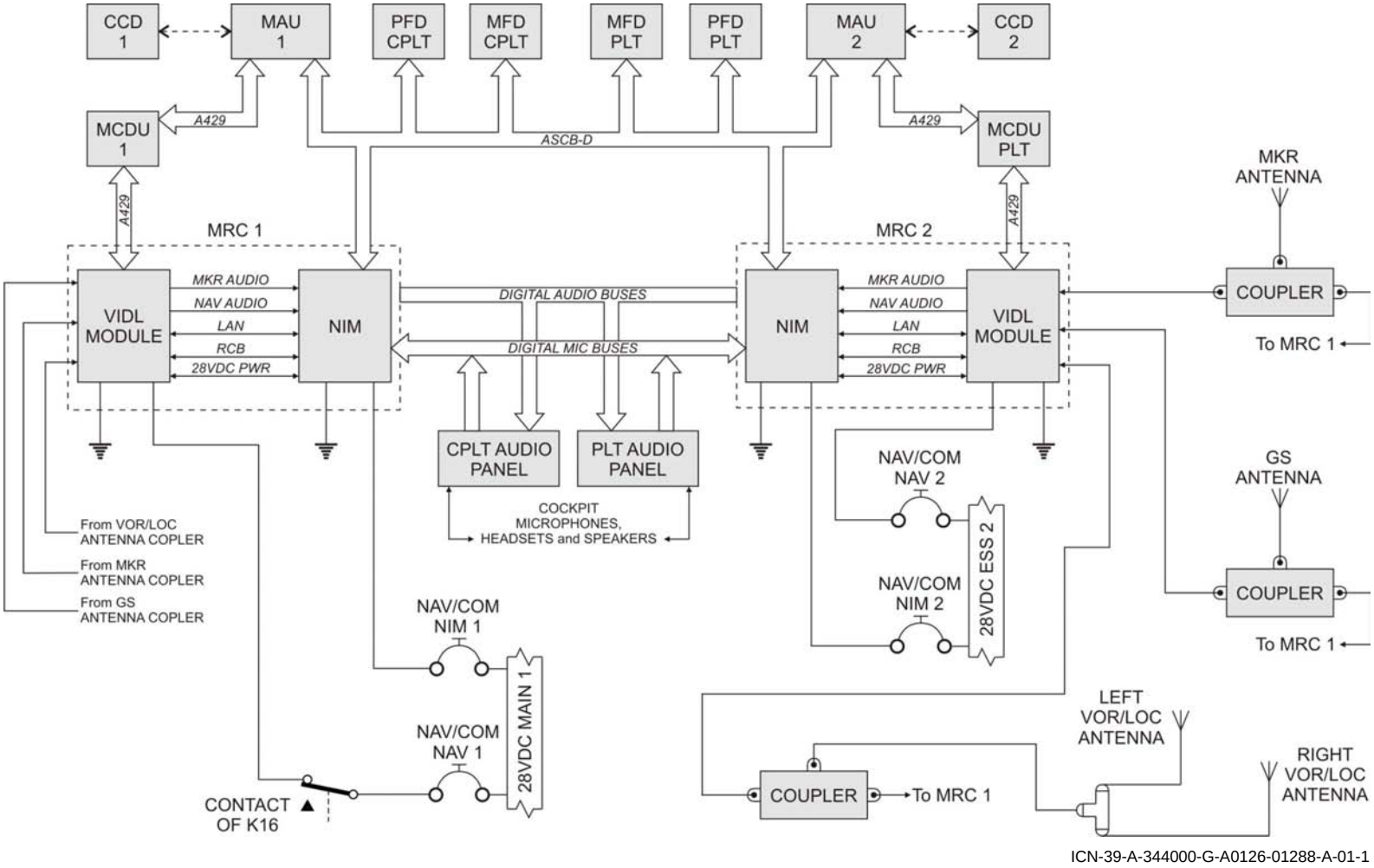
MB, VOR AND GS ANTENNAS LOCATION

OPERATION

The VIDL system transmits and receives data through the radio communication bus (RCB) to the network interface module (NIM) in the MRC. The NIM connects to the display system through the avionics standard communication bus, version-D (ASCB-D). Backup tuning is supplied by the MCDU through a dedicated ARINC 429 interface.
The VIDL system operates in one of the operational modes that follow:

- VOR
- ILS

The VIDL system operates in one operational mode at a time.
In the VOR mode, the system receives and processes VOR signals and MKR beacon signals. The VIDL system supplies the radio function of the VOR receiver that supplies bearing in degrees to and from the ground station and for navigation of the aircraft.
In the ILS mode, the system receives and processes LOC, GS, and MKR beacon signals. The VIDL system supplies the radio function of the ILS, which includes LOC, GS, and MKR beacon receivers, that supplies approach and landing navigation data.
The VOR receiver operates in the frequency range of 108.00 to 117.95 MHz with 50 kHz channel separation. It is tuned to a VOR ground station that radiates a signal that changes with azimuth related to the station. The ground station transmits a radio frequency (RF) signal with two 30 Hz modulated signals. One is AM and the other is frequency modulated (FM) on a 9960 Hz subcarrier. The relative phase of the two 30 Hz signals gives radial lines in space related to the ground station. The VOR receiver can then find the bearing of the aircraft to the ground station. The VOR receiver supplies bearing, deviation, and alarms (flags) to the pilots from the VOR ground station signals. Deviation is calculated as the difference between calculated bearing and an analog course selection input. The LOC receiver operates on 40 channels in the frequency range of 108.00 to 111.95 MHz (on odd tenths) with 50 kHz channel separation. It supplies the azimuth lateral deviation from the runway centerline as part of the ILS. The LOC receiver compares the amplitude difference between the mostly 90 Hz and mostly 150 Hz AM modulated intersected lobes that are each radiated along one side of the runway. The two AM detected audio signals are equal in magnitude along a line that divides the center of the runway into two equal parts. The LOC receiver supplies left and right command data and alarms (flags) to the pilot and copilot from the LOC ground station signals. The GS receiver operates in the frequency range of 329.00 to 335.00 MHz with 150 kHz channel separation. There are 40 channels (or frequencies) specified for the GS in the frequency range of 329.00 to 335.00 MHz. The GS channels are put together one-to-one with the LOC channels. When the LOC is tuned, the GS is tuned with it. The GS receiver supplies the elevation deviation from runway GS as part of the ILS. The GS receiver also compares the amplitude difference between the mostly 90 Hz and mostly 150 Hz AM modulated intersected lobes that are each radiated one over the other. The two audio signals are equal in magnitude along a line 2.5 to 3 degrees above the runway called the GS. The GS receiver supplies up and down command data and alarms (flags) to the pilot and copilot from the GS ground station signals.
The MKR beacon receiver operates at 75 MHz. It supplies inner, middle, and outer MKR beacon crossing data. The receiver operates lamps and supplies audio tones when the beacon signals are received. The ground station transmits an RF signal with keyed audio frequency AM modulation at one of three modulation frequencies. The outer MKR frequency is 400 Hz. The location of the outer MKR is at a stable distance of approximately 5 miles. The middle MKR frequency is 1300 Hz. The location of the middle MKR is at a stable distance of approximately 3500 feet. The inner MKR frequency is 3000 Hz. The location of the inner MKR is near the end of the runway on the approach centerline. The MKR beacon receiver supplies inner, middle and outer visual data and audio to the pilot and copilot from the MKR beacon ground station signals.



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VIDL ELECTRICAL SCHEMATIC

CONTROLS AND INDICATORS

Display Controller (Dc)

The display controller gives the flight crew independent selection of short-range navigation (SRN) and long-range navigation (LRN) sources, on-side/cross-side bearing sources and the HSI format for display on the respective PFD. The display controller is directly connected to the cursor control device and the remote instrument controller.
Pushing circle (Number 1) and diamond (Number 2) bearing (BRG) buttons permit the bearing pointers can be displayed on the HSI. Each pointer has a range of 360 degrees. When selected the bearing pointers function as a radio magnetic indicator (RMI). Loss of bearing information from the selected NAV source results in the removal of the relevant bearing pointer.

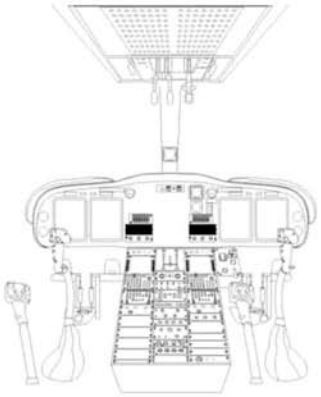
1 BRG pushbutton (○): ..... pushing the circle BRG push button sequence is: OFF, VOR1, ADF (ADF1 if is installed), FMS1 and OFF.

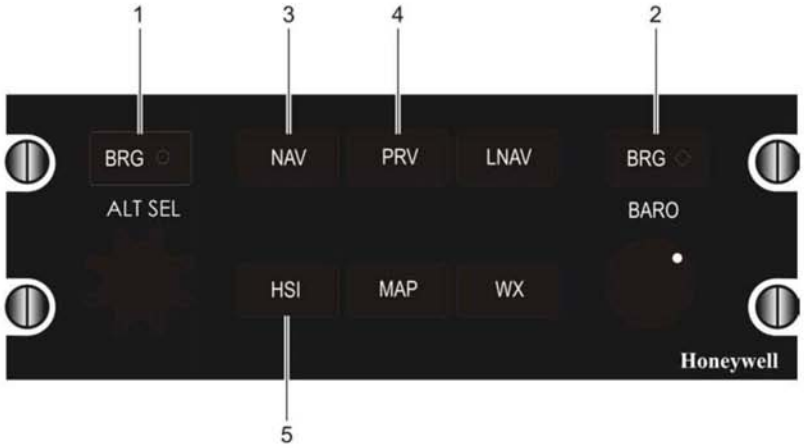
2 BRG pushbutton (◊): ..... pushing the diamond BRG push button sequence is: OFF, VOR2, ADF (ADF2 if ADF1 is installed), FMS2 and OFF.

3 NAV pushbutton: ..... pushing the NAV button provides data relevant to a Short-range Navigation (SRN) source, displayed on the on-side PFD. The button sequence is: VOR/LOC on side and VOR/LOC cross side. The power-up default is set to the on-side VOR/LOC.

4 PRV pushbutton: ..... pushing the PRV button shows the on-side or cross-side VOR/LOC on the display using a secondary course pointer when long-range NAV is the primary engaged navigation source. OFF is the default setting. The button sequence is: OFF, on-side VOR/LOC, cross-side VOR/LOC and OFF. The PRV button has no effect when the FMS (LRN) is not selected as the current NAV source.

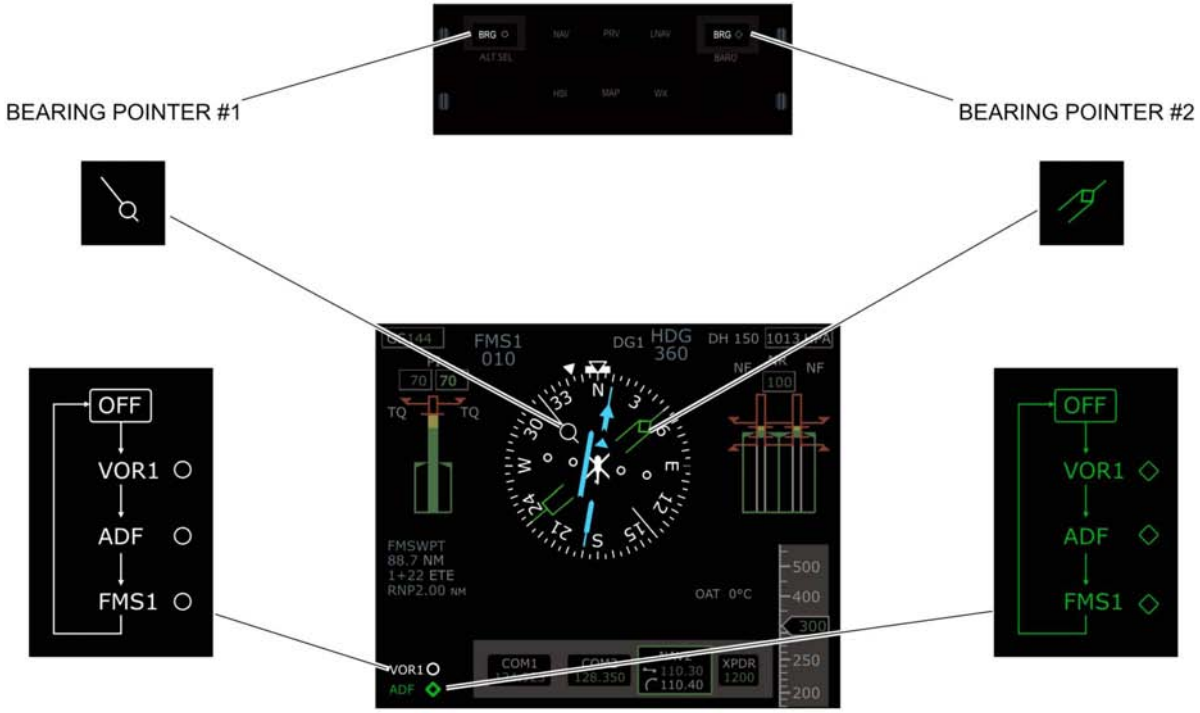
5 HSI pushbutton: ..... the HIS button permits the pilot to select between a full compass (360°) display or an arc (45° centered on the aircraft nose) compass display. Full compass is the default setting.





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DISPLAY CONTROLLER



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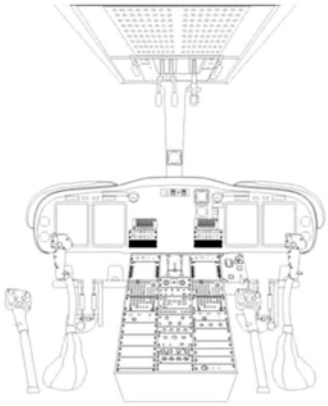
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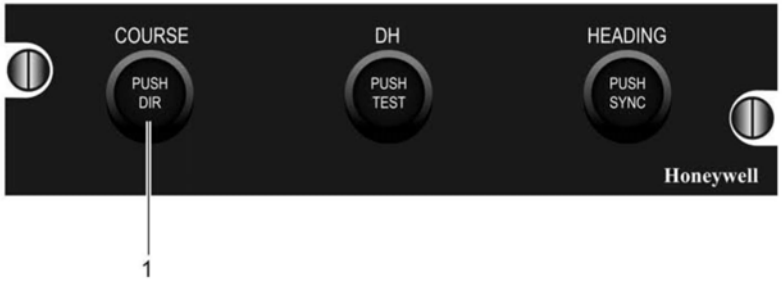
BEARING POINTER SELECTION

Remote Instrument Controller (Ric)

The RIC is used to set the course, decision height and heading. The control panel has three control knobs with associated buttons.

1 COURSE ..... the course select knob selects the desired course during short-range NAV or NAV preview on the on-side HIS. Pushing the PUSH DIR button synchronizes the selected course to the current VOR bearing.





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REMOTE INSTRUMENT CONTROLLER

PFD – HIS INDICATIONS

Indications relevant to the VIDL system are provided on PFDs by Course (CRS) and Bearing (BRG) pointers, annunciators and digital readout, when SRN system (VOR/LOC) are set through the on-side Display Controller.
Indications relevant to the VIDL system are as follows:

1 BRG Pointers ..... Two bearing pointers can be displayed on the compass. When selected, the bearing pointer functions as a radio magnetic indicator (RMI). They are presented on the display by pushing the BRG select buttons on the display controller. The circle bearing pointer is always white. The diamond bearing pointer is always green. Loss of bearing information from the selected NAV source results in the removal of the relevant bearing pointer.

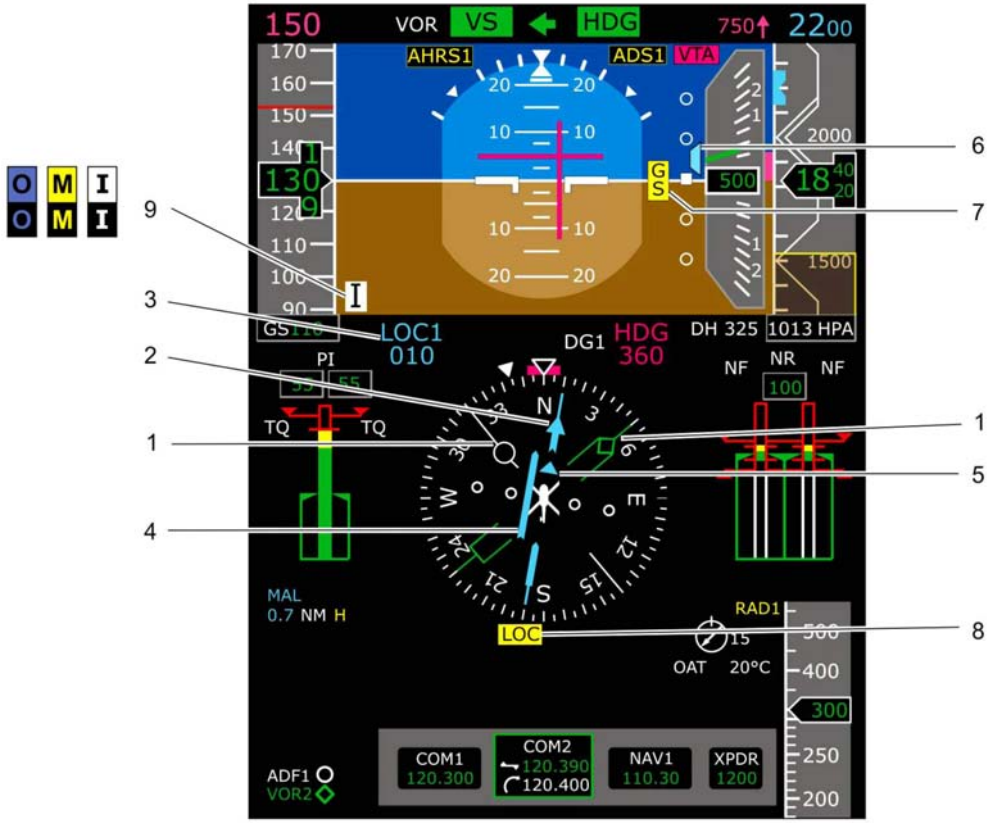
2 CRS Pointer ..... The course is a segmented needle that is superimposed over the HIS compass display. It represented a selected SRN course (CRS) of FMS desired track (DTK) and the helicopter position relative to it. The pointer is cyan when the flight director is not coupled and magenta when the flight director is coupled. The course pointer can be rotated 360° clockwise and couterclockwise using the course select knob on the on-side remote instrument controller. When, in arc mode, the course set is off the scale, a cyan or magenta colored arrow shows the shortest turn direction to the pointer. The arrow is removed when the Course Deviation Indicator returns in view. A digital readout legend of the selected course or track is displayed above and to the left of the HIS immediately below the associated NAVAID.

3 NAV Source ..... The selected navigation source legend is positioned at the upper left of the compass card immediately above the digital course readout. The NAVAIDS available for display are: VOR1/LOC1, VOR2/LOC2 for the SRN and FMS1, FMS2 for the LRN. The VOR1 or VOR2 legend is displayed when VOR/LOC mode is selected and the NAV source is not tuned to a localizer frequency. When the NAV source is tuned to a localizer frequency, the LOC1 or LOC2 annunciator is displayed.

4 CDI ..... The CDI (CRS Deviation Indicator) indicates lateral deviation from the centreline of a selected course. As shown at left, the CDI is composed of the course pointer segmented needle and a scale centered on a helicopter symbol. The deviation bar (middle part of the segmented needle) indicates course, deviation by moving laterally on a scale consisting of four

		dots; two on the left and two on the right of the helicopter symbol. The pointer parked position equals 2 ½ dots.
5	TO/FROM Indicator	The TO/FROM indicator is a white solid triangle superimposed on the center line of the course pointer. It is positioned at the nose or tail of the helicopter symbol and moves with the course pointer. When deviation from the VOR course is ≤88°, the TO/FROM indicator is positioned at the nose of the aircraft symbol indicating TO the Station. When deviation from the VOR course is ≥ 92° the TO/FROM indicator is positioned at the tail of the helicopter symbol indicating FROM the station.
6	G/S Vertical deviation	When tuned to an identified ILS, frequency, the pilot can use the display controller to select glideslope information for display on the vertical deviation scale from either the on-side or cross-side ILS. The scale consists of a rectangle with two dots above and below it. The vertical deviation scale is displayed to the right of the attitude sphere as a pointer moving on a scale when LOC is selected. Each dot represents a graduation of 1.5° above or below glideslope on an ILS approach. As shown in the example, the glideslope deviation pointer or bug is a truncated cyan triangle displayed on the right side of the vertical deviation scale. The bug represents the position of the aircraft on the ILS approach in relation to the glideslope. The rectangular box on the horizon line represents the position of the aircraft. The glideslope bug shows the pilot where the glideslopeis in relation to the helicopter. When the bug is above the box on the display, the helicopter is below glideslope. When the bug is below the box on the display, the helicopter is below glideslope. When the bug is below the box on the display, the helicopter is above glideslope,. Using the position of the b ug in relation to the dots, the pilot can estimate the number of degrees the aircraft is above or below the glideslope. When the bug is immediately adjacent to the box, the aircraft is on glideslope. In the example above, the aircraft is about 1° below the glideslope. The bug is in magenta when the flight director is coupled.
7	GS Miscompare	The monitor warning system continuously monitors the glideslope. When a glideslope

		miscompare is detected, the system alerts the pilot by displaying the GS miscompare annunciator on the left side of the vertical deviation scale. The GS miscompare annunciator has display priority over all scale markings. It flashes during the first six seconds. After six seconds GS lights steadily. The annunciator does not display when a glideslope miscompare is no longer detected. The loss of valid information from either NAV receiver does not display the GS annunciator.
8	LOC Miscompare	When a localizer miscompare is detected, the LOC annunciator is displayed on the HIS just below the helicopter symbol on the compass card. This alerts the pilot to a localizer miscompare. The LOC miscompare annunciator has display priority over all scale markings. It flashes during the first six seconds. After six seconds, the LOC shows steadily. The annunciator is removed when a localizer miscompare is no longer detected. A loss of valid deviation information from the NAV receiver or monitor warning function does not display the LOC annunciator.
9	MB Annunciator	Outer (O) Middle (M) and Inner(I) marker beacon annunciators are displayed in the lower left corner of the ADI and are controlled by the selected NAV receiver (VOR/ LOC). The pilot is alerted to marker beacon passage by the beacon annunciator. As long as the aircraft is in the cone of sensitivity, the annunciator switches between reverse video for one second and in normal video for a half second.



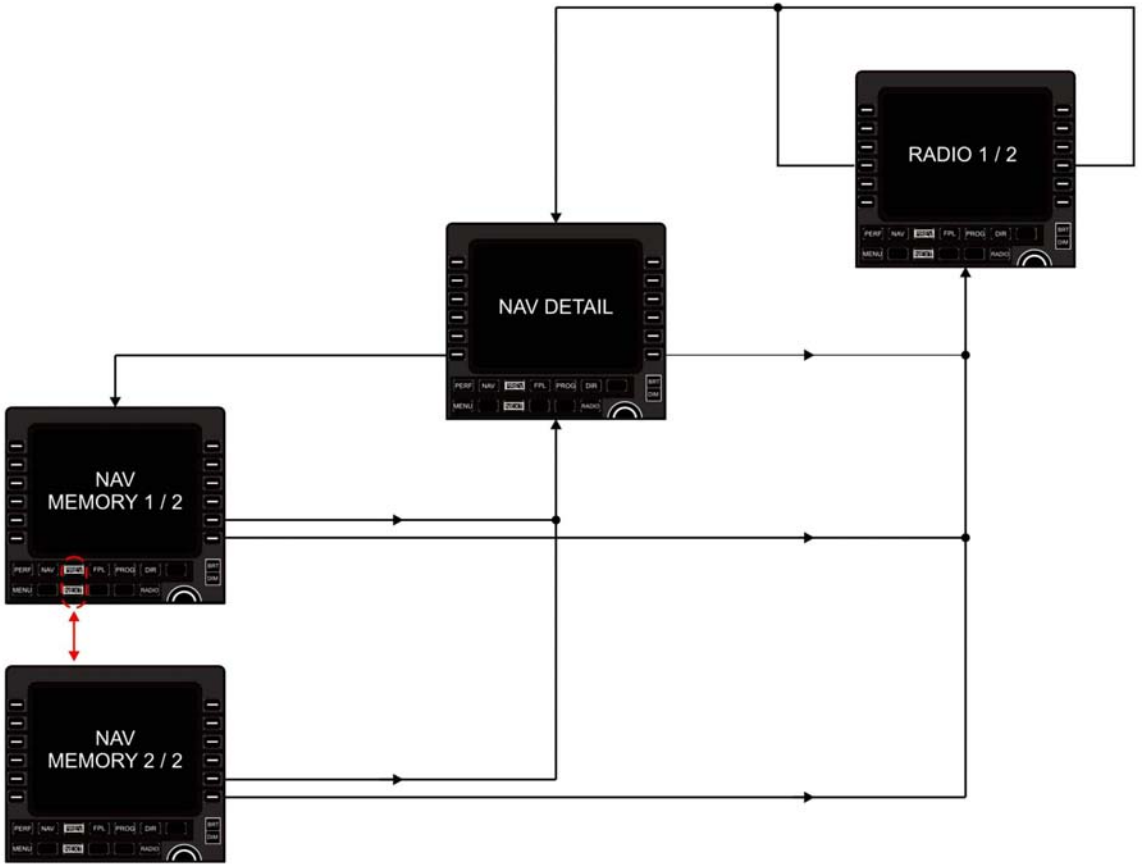
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PFD HSI INDICATIONS

VIDL SYSTEM OPERATION WITH THE MCDU

The MCDU is the primary control unit for the radio system. The MCDU controls all communication and navigation radios in the MRC, also it controls the modes, frequencies, and codes of all the radio modules in the MRC. A set of VHF NAV radio tuning pages is provided by MCDU, in order to tune the VIDL system.

- RADIO 1/2 page. The RADIO 1/2 page is displayed by pushing the RADIO function key on the MCDU. It contains the following radio data:
  - VHF COM1 and COM2 radios
  - VHF NAV1 and NAV2 radios
  - The currently selected TCAS/transponder mode
  - The aircraft ID (identification) (if available)
  - The current transponder code and status.
- NAV1 page. It is used to access and control VHF navigation radios, FMS automatic tuning, and DME hold mode. The format cursor defaults to the memory tuning field (4L), providing quick access to stored frequencies. The page can also access to the NAV memory pages.
- NAV MEMORY 1/2 and 2/2 pages. The radio tuning function supports 12 navigation radio memories displayed on two pages. In addition to entering or dialing-in frequencies for each memory, a text label of up to eight characters can be entered for each stored frequency. The default label for each memory is MEMORY, a dash, and the memory number, with the memory number always on the outboard edge of the display.



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VHF NAV OPERATION WITH MCDU (SHEET 1 OF 5)



ICN-39-A-344000-G-A0126-01296-A-01-1

VHF NAV OPERATION WITH MCDU (SHEET 2 OF 5)



ICN-39-A-344000-G-A0126-01297-A-01-1

VHF NAV OPERATION WITH MCDU (SHEET 3 OF 5)



ICN-39-A-344000-G-A0126-01298-A-01-1

VHF NAV OPERATION WITH MCDU (SHEET 4 OF 5)



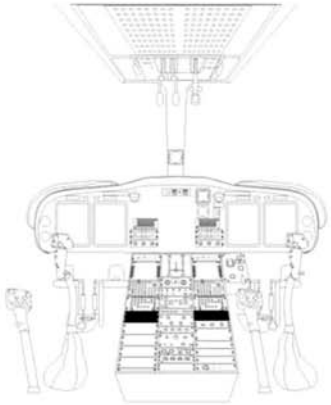
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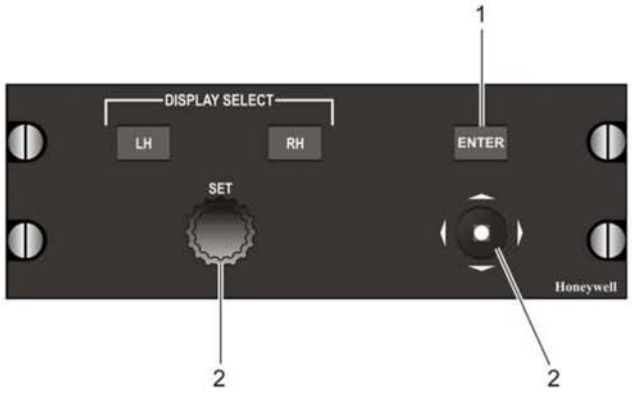
VHF NAV OPERATION WITH MCDU (SHEET 5 OF 5)

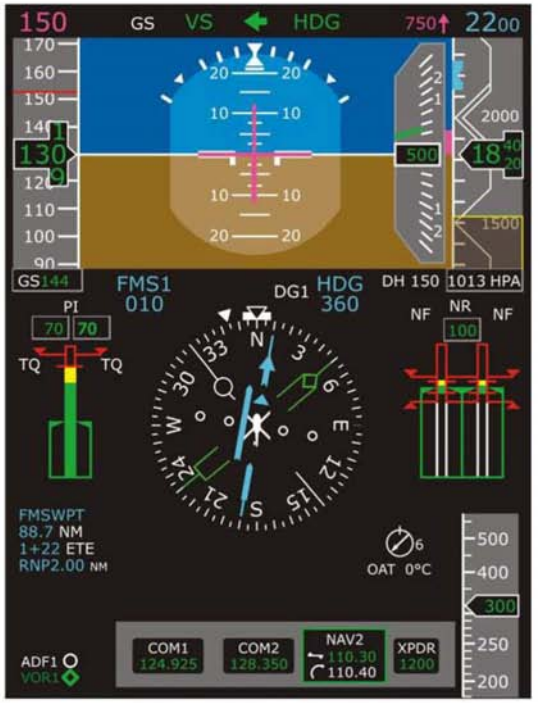
VIDL SYSTEM OPERATION WITH THE PFD

The PFDs and MFDs supply integrated displays of navigation. VIDL radio tuning on the PFD is set using the on-side CCD.
The CCD is used to change NAV/COM frequencies and the XPDR code on the PFD. A frequency box becomes active when the pilot positions the cursor over the desired frequency box. To indicate that a frequency box has been selected for tuning, all the items in the box are displayed using large letters and numbers or fonts.
The first item contained in a frequency box is the identifier displayed using white fonts. The example previously shows the NAV1 frequency box.
Immediately below the identifier is the active frequency position. The active frequency is always displayed in green. In this instance, the active frequency in NAV1 is 110.30.

1	ENTER:	Pushing the ENTER button on the CCD swaps the current and preset frequencies for the radio.
2	SET KNOB:	Turning the larger knurled knob clockwise increases and counterclockwise decreases the whole number component of radio frequencies. The smaller knob segment controls the decimal number component of frequency ranges when selecting a radio frequency.
3	JOYSTICK:	A green cursor box is positioned to highlight the function using the CCD. Moving the joystick to shift the cursor box between the COM and NAV radios.





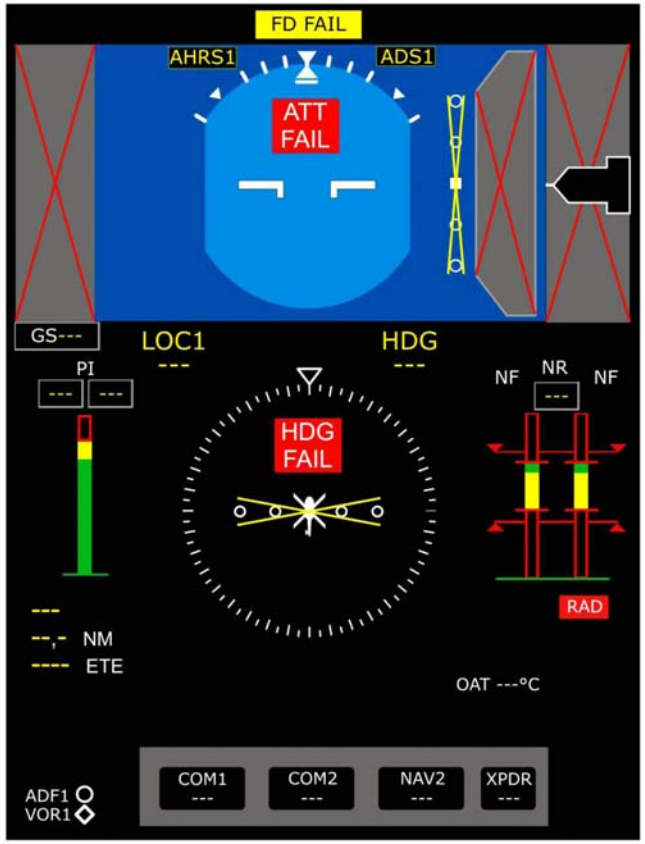


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CURSOR CONTROL DEVICE

CAPTIONS

No-one cautions message is available for the CAS window in the event of VIDL system failure. If a VIDL system is not supplying valid data to the MCDU or PFD, or a VIDL system failure occurs, the frequencies or the operating commands associated with that function are removed from the associated display replaced with dashes and cross amber.



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FAILURE INDICATIONS

LEADING PARTICULARS

The leading particulars for the VOR/ILS data link (VIDL) receiver module are as follows:

- Length..... 15.00 in (381.0 mm)
- Width..... 3.29 in (83.57 mm)
- Height..... 4.09 in (103.89 mm)
- Weight (maximum)..... 2.90 lb (1.136)
- Power Requirement..... 28 VDC
- Circuit Breaker..... 28 VDC 3 A

Vor Specifications

- Frequency Range..... 108.00 to 117.95 MHz
- Channel Spacing..... 50 KHz
- Frequency Stability..... 0.0025 %
- VOR Bearing Accuracy..... ± 0.75°
- Selectivity..... 6 dB at ±15 KHz minimum (60 dB at ± 41 KHz maximum)
- Spurious Response..... -80 dB
- Audio Output..... 100 mW into 600 ohm

Loc Specifications

- Frequency range..... 108.10 to 111.95 MHz
- Channel Spacing..... 50 KHz
- Deviation Centering..... ±0.0026 ddm
- Polarity..... Positive ddm for 90 Hz > 150 Hz (aircraft left of LOC course)
- Selectivity..... 6 dB at ± 15 KHz minimum (60 dB at ± 41 KHz maximum)
- Spurious Response..... -80 dB
- Audio Output..... 100 MW into 600 minimum

Gs Specifications

- Frequency Range..... 329.15 to 335.00 MHz
- Channel Spacing..... 150 KHz
- Deviation Centering..... ± 0.0058 ddm
- Polarity..... Negative ddm for 90 Hz > 150 Hz (aircraft above GS course)
- Selectivity..... 6 dB at ± 15 KHz minimum (60 dB at ± 41 KHz maximum)
- Spurious Response..... -60 dB
- User Replaceable Parts..... None
- Mounting..... MRC cabinet Honeywell P/N 7517959-902

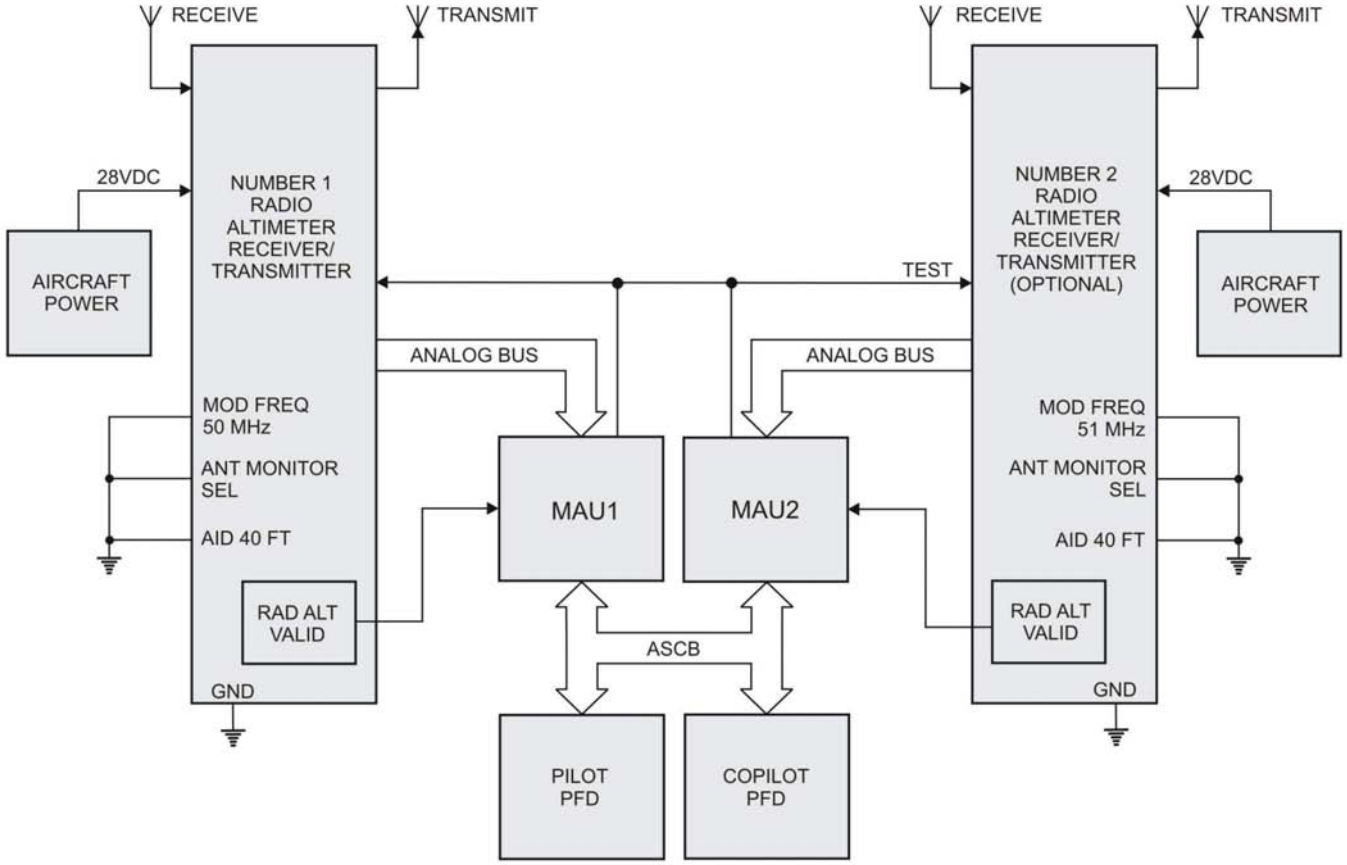
SECTION 40 - INDEPENDENT POSITION DETERMINING

DESCRIPTION

The independent position determining has the systems that follow:

- Radio Altimeter System

The radio altimeter (RAD ALT) system uses frequency-modulated continuous wave (FMCW) signals to calculate the aircraft altitude Above Ground Level (AGL) during the phases of flight, approach and landing. The system has a maximum range of 2500 ft of altitude.
Radio altitude data are shown on the PFDs (Compass mode, Arc mode and Reversion mode).
A single RAD ALT system is the standard installation and sends digital radio altitude data to the MAUs.
In a dual RAD ALT system, the receiver-transmitter no.1 supplies the radio altitude data for the co-pilot's PFD and the receiver-transmitter no.2 supplies the radio altitude data for the pilot's PFD.
The RA data is displayed in two locations on the PFD. First as the Low Altitude Alert Display (LAAD), this is part of the altitude tape when the aircraft is approaching the ground. This is designated by an amber line and brown shading. Secondly, a separate RA tape and/or digital readout are displayed in the bottom right hand corner of the PFD.
The flight crew can set the DH altitude with the DH knob on the remote instrument controller. The RAD ALT system supplies an automatic indication when the aircraft gets to that altitude.
The units are installed on the E-E avionic bays.
The primary components of the radio altimeter system are the:
Number 1 and 2 radio-altimeter receiver-transmitter (RAD ALTs)
Number 1 and 2 radio-altimeter receive-antennas.
Number 1 and 2 radio-altimeter transmit-antennas.



ICN-39-A-344000-G-A0126-01302-A-01-1

RADIO ALTIMETER SYSTEM SCHEMATIC DIAGRAM

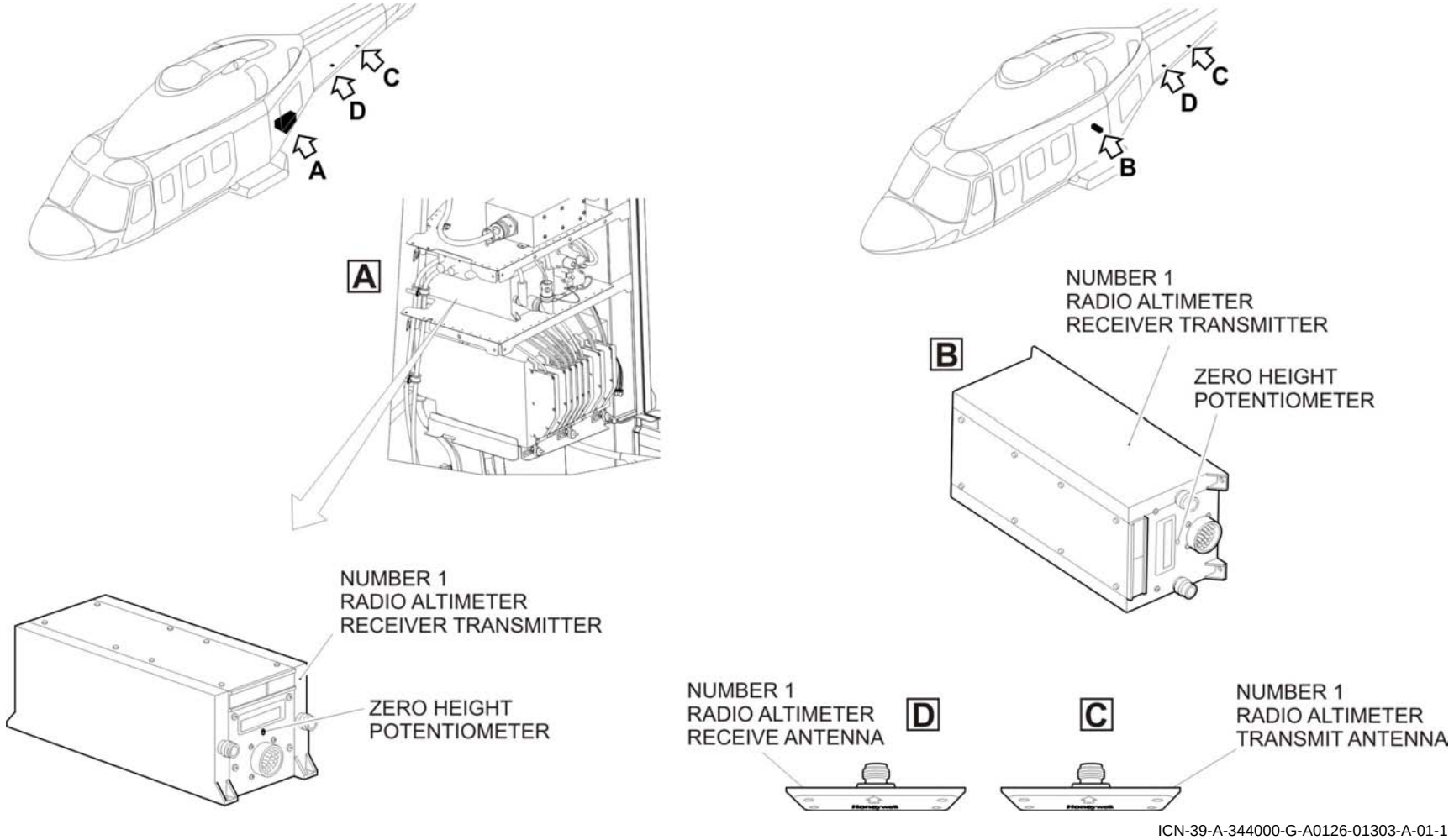
RADIO ALTIMETER (RAD ALT)

The RAD ALT is pulsed radar which utilizes separate external transmit and receive antennas.
The RAD ALT is a self-contained microwave transmitter receiver and range computation system. It operates on short pulse, leading edge track principles to measure and to show aircraft absolute altitude on the PFDs. It also has the zero altitude adjustment potentiometer.
The RAD ALTs (A17 and A18) are installed in the left and right avionic bay (standard nose helicopter) or on the left and right bottom side of the center fuselage (long nose helicopter).
Four screws and the four washers attach each RAD ALT to the support base (standard nose helicopter) or to the fuselage structure (long nose helicopter).
Three electrical connectors connect the RAD ALT1 to these components:

- The circuit breaker CB86
- The MAU1 (and MAU2 if only RAD ALT1 is installed)
- The antennas

Three electrical connectors connect the RAD ALT2 to these components:

- The circuit breaker CB143
- The MAU2
- The antennas.

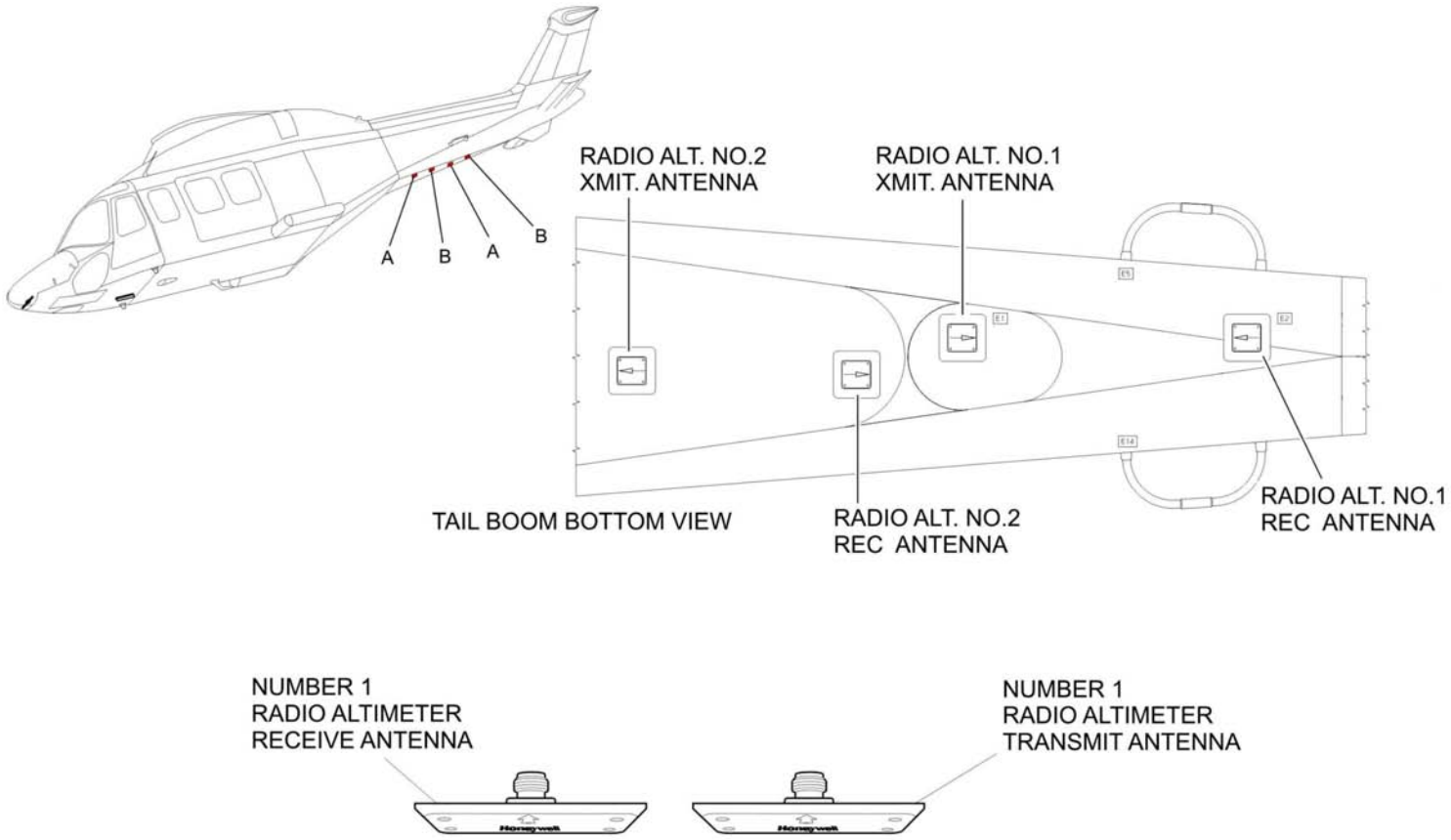


ICN-39-A-344000-G-A0126-01303-A-01-1

RADIO ALTIMETERS-COMPONENTS LOCATION

RADIO ALTIMETER ANTENNAS

The four radio-altimeter antennas are installed on the bottom of the tail section. These antennas are low-contour, rectangular, micro-strip. The antennas operate in the 4200 to 4400 MHz bandwidth.
Four screws and the four washers attach each antenna to the tail structure. A coaxial connector connects the antenna to the RAD ALT.
The RAD ALT1 antennas are installed on the bottom of the after tail section. The receive-antenna (E2) is located after the transmit-antenna (E1).
The RAD ALT2 antennas are installed on the bottom of the forward tail section. The receive-antenna (E18) is located after the transmit-antenna (E19).
The location of the antennas lets each antenna operate at the normal limits of pitch and roll.

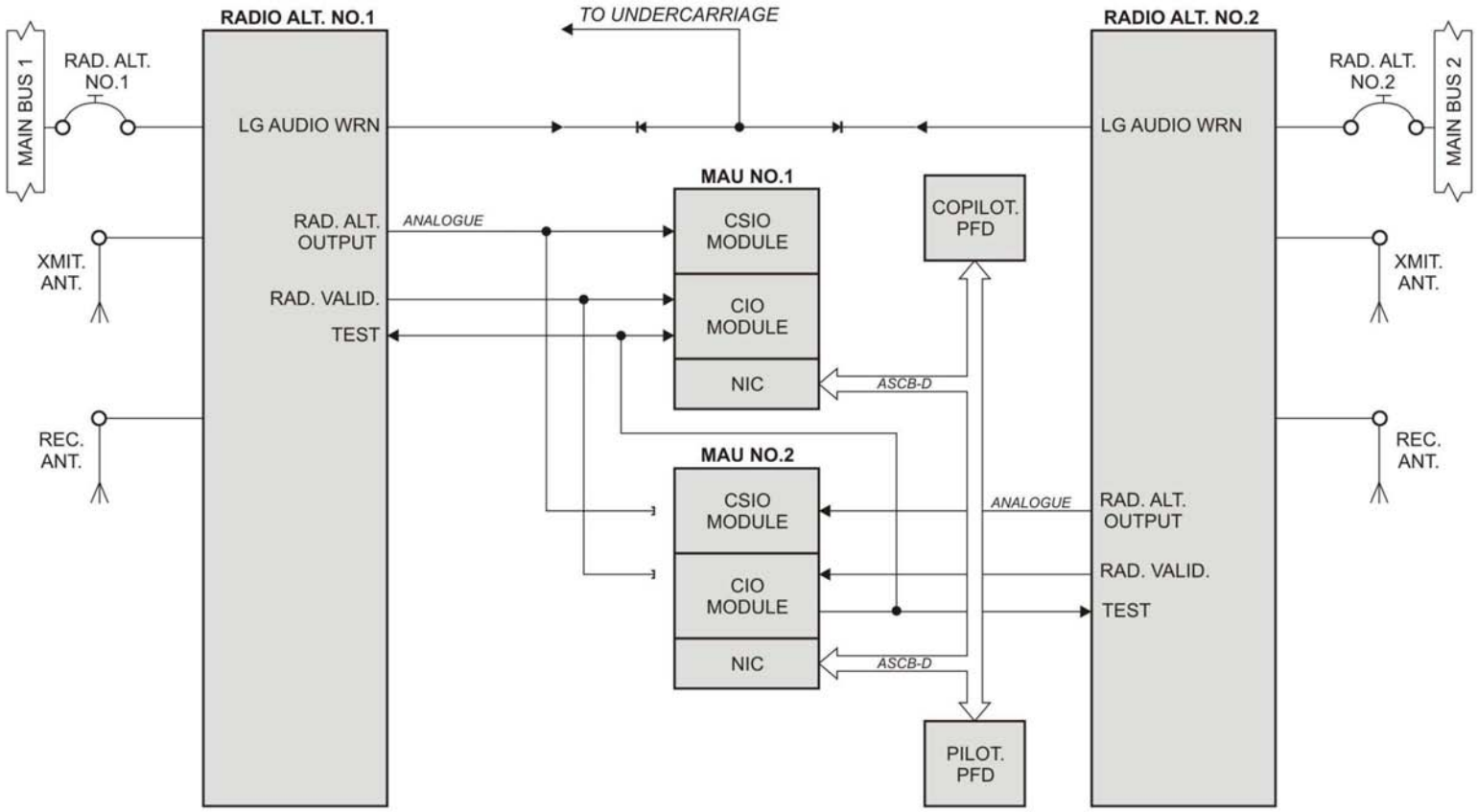


ICN-39-A-344000-G-A0126-01304-A-01-1

RADIO ALTIMETERS-ANTENNAS LOCATION

OPERATION

The RAD ALT system is energized before takeoff and can be used during the flight to monitor the aircraft altitude AGL up to the maximum range of the system.
The radio altimeter system operates over large differences in terrain, weather conditions and aircraft attitude, to calculate the aircraft's altitude.
The RAD ALT contains the power supplies, the transmitter and receiver circuits, with the timing circuits to calculate the aircraft's AGL altitude. The RAD ALT receives power from the aircraft's 28 V DC bus. The RAD ALT transmits 4300 MHz FMCW signals to the transmit antenna, which transmits the signal to the ground. The receive antenna receives the signal reflection from the ground and transmits it to the receiver part of the RAD ALT. It measures the time interval between the transmitted and received FMCW signals. The received signal changes to a DC analog voltage. The DC analog voltage transmits accurate analog outputs to the MAUs. The DC analogue output voltage is proportional to the aircraft' AGL altitude.
When one radio altitude is installed, it is wired to both MAUs. When two radio altimeters are installed, each is wired to the on-side MAU.
The discrete inputs for the modulation frequency selection set the modulation for each receiver-transmitter in a dual system. Receiver-transmitter 1 is set for a 50 Hz modulation frequency and receiver-transmitter 2 is set for a 51 Hz modulation frequency. The radio altitude information is removed from the view above 2500 ft AGL (no track).
The receiver-transmitter has a continuous self-test, and when a fault occurs, a RAD failure indication it is shown on each PFD. The test signal comes from the MAUs.
The RAD ALT system is connected with other systems through the input/output (I/O) modules of the MAU.
The RAD ALT system supplies the decision height (DH) output. The DH output sets the gain for the localizer (LOC) and glideslope (GS) signals for the flight director (FD). The DH range is thru 20 to 2500 ft. It is shown on the PFD.
The flight crew can set the DH altitude with the DH knob on the remote instrument controller (RIC). The RAD ALT system supplies an automatic indication when the aircraft gets to that altitude.
The RAD ALT system supplies the lower portion of the altitude tape functions as a Low Altitude Alert Display (LAAD). This display alerts the pilot at any time the aircraft is being operated within 550 ft AGL.



ICN-39-A-344000-G-A0126-01305-A-01-1

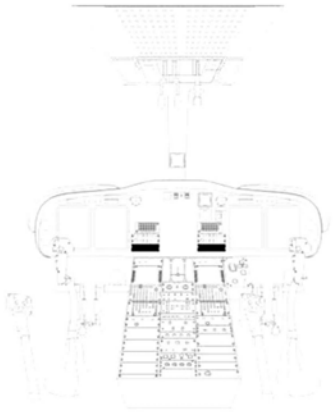
RADAR ALTIMETER ELECTRICAL SCHEMATIC

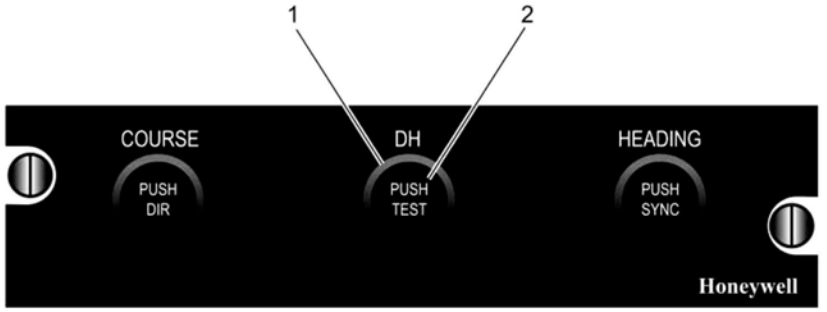
RADIO ALTIMETER CONTROLS

Remote Instrument Controller

The Radio Altimeter system control is housed on the Remote Instrument Controller (RIC) located on the instrument panel and on the Miscellaneous Control Panel (MCP) located on the inter-seat.
The RIC provides the following functions:

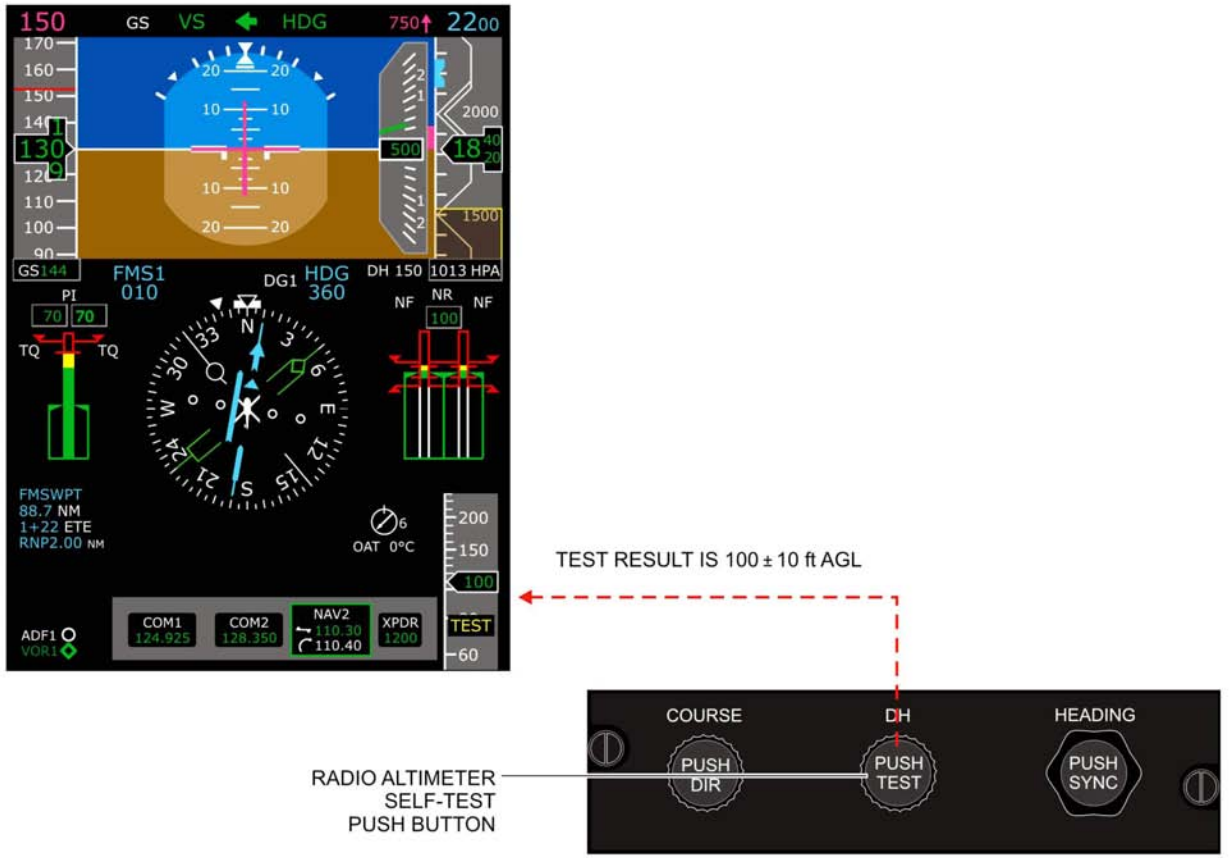
1	DH selector knob:	when rotate, it allows changing the decision height display on the on-side PFD from 20 to 2500 ft.
2	PUSH TEST push button ……	when pressed, the built-in test is associated for the radio altimeters. The display pointer indicates 100 ± 10 ft and TEST flag in inverse video (displayed on the radio altimeter tape) is represented until the button is released. Also the amber messages RAD1 and RAD2 are visible during the test. The test can be done both on ground and in flight.





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RADIO ALTIMETER-REMOTE INSTRUMENT CONTROLLER



ICN-39-A-344000-G-A0126-01307-A-01-1

RADIO ALTIMETER-TEST MODE

RADIO ALTIMETERS INDICATIONS

The RAD ALT indications that follow show on the PFDs (Compass/Arc/ Reversion mode):

- Radio Altitude Display
- Low Altitude Awareness Display (LAAD)
- DH Display
- DH Minimums

1 Radio Altitude Display: ..... When the radio altimeter tape is displayed, the digital readout is incorporated into the tape's pointer. The digital readout is divided into 1 foot increments between -20 to +250 ft AGL. Above that, increments are at 10 ft intervals from +250 to +2500 ft AGL. The radio altitude information is removed from view above 2500 ft AGL. Brown shading of the RA tape is present below zero (0) ft. gray shading is present above zero (0) ft. When the helicopter ascends through 1100 ft AGL, the tape is removed and the digital readout only is displayed below the triple tach display.

The PFD automatically selects the source for the radio altitude data. Each PFD displays radio altitude data from the on-side MAU, if that data is valid. If the on-side radio altitude data is invalid, the radio altitude data from the off-side MAU is displayed.

The radio altitude reference bug is displayed when the flight director RHT mode has been is selected. The bug is a magenta rectangle with a triangular notch, positioned along the right edge of the radio altitude tape corresponding to the radio altitude reference digital readout. When the radio altitude target is off scale, the direction that it is off is indicated. The reference digital readout and the bug are not displayed in either of the following circumstances:

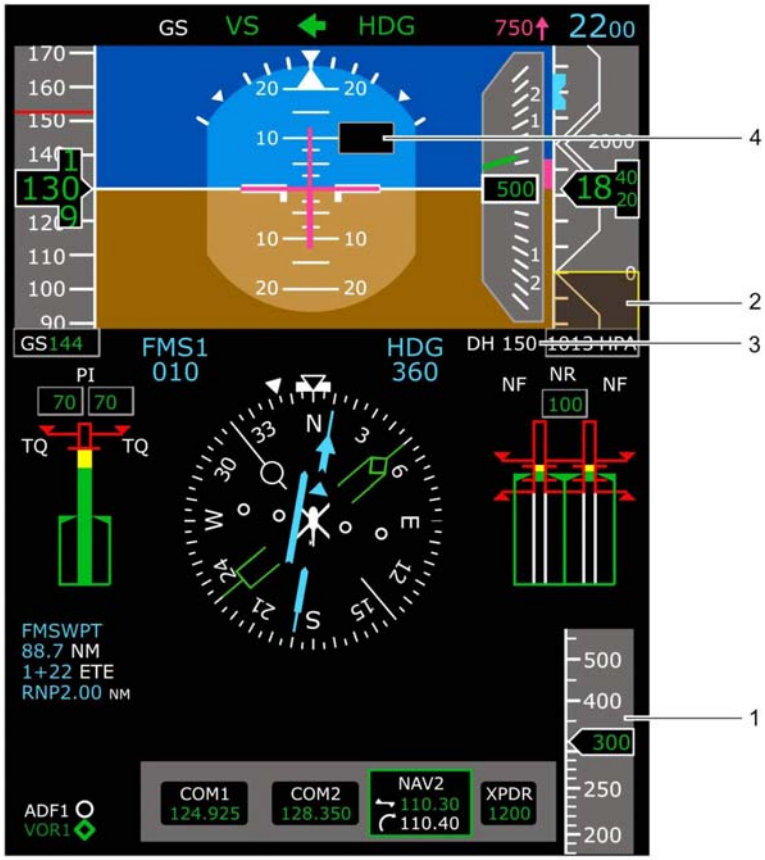
- Loss of valid radio altitude information from each MAU.
- Loss of valid radio altitude reference information from the priority flight director.

2 Low Altitude Awareness Display (LAAD): ..... When the aircraft descends through 550 ft AGl, the LAAD is presented on the barometric altitude tape to show ground with relation to the barometric altitude. The LAAD is an area of brown shading bordered by an amber line. The LAAD advances or recedes vertically along the altitude tape with changes in absolute altitude below 550ft AGL. The display repeats the data

from the LAAD display from the RA scale. All data fro the LAAd is derived from the RA.

3 DH Display: ..... The DH is displayed as a digital readout located below the attitude sphere and to the right. The DH is set in the cockpit by the on-side RIC. The data range corresponds with the radio altitude range (□2500 ft). Upon power-up, the DH display defaults to blank values. The first control knob turns starts the display reading at 200 ft. Each knob click there after is at intervals of 10 ft. The DH display is removed for settings below 20 ft. a loss of valid DH setting from the DC function results in the DH display is amber dashed.

4 DH Minimums: ..... During descent, when radio altitude is equal to DH + 100ft, an empty black box with an amber outline is displayed in the upper right portion of the attitude sphere. When RA is equal to or less than the DH setting, a MIN annunciator appears in the empty box on the attitude sphere. The MIN annunciator shares the location with the Vertical Track Alert (VTA) annunciator. The MIN annunciator is inhibited on the ground and through climb out until RA is greater than DH + 100 feet. A loss of valid RA information on ASCB or valid DH setting from the RIC inhibits the MIN annunciator.



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RADIO ALTIMETER - INDICATOR



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DH-DECISION HEIGHT

CAPTIONS

No caution is displayed in the event of the radio altimeter failure. When the 150 FEET condition is detected, the AWG shall provide the following:

- one tone (494 Hz) + 150 FEET (aural message).

This sequence is transmitted one time only. His message has priority number 9.

On the PFDs the following are proposed

1 Radio Altimeter source Annunciation (close to the radio altimeter tape).

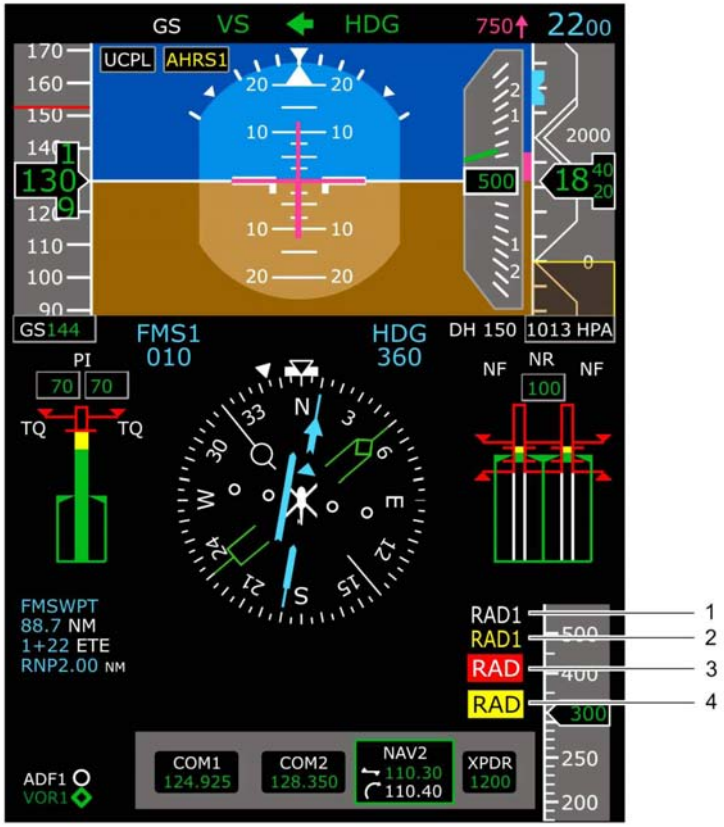
2 RAD1 or RAD2: this amber message is automatically in view when the PFD is displaying data from the off-side radio altimeter sensor.

Failure annunciations

3 RAD: when loss of valid radio altimeter information from either MAU, the red flag (inverse video) is displayed.

Miscompare annunciation

4 RAD: when radio altitude miscompare is detected by the EDS monitoring software, a RAD2 to RAD to RAD2 annunciator flashes for the first six seconds and then shows RAD steadily above the radio altitude tape. Once the miscompare is no longer detected, the annunciator is removed.
The miscompare function is performed wheather there are one or two radio altimeters installed. When only one radio altitude is installed, the miscompare function compares the same data from the two MAUs.



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RADIO ALTIMETER – CAPTIONS

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SECTION 41 - WEATHER RADAR SYSTEM

DESCRIPTION

The Weather Radar System (WRS) identifies the location and intensity of rain, hail, and turbulence in the aircraft flight path. Also, the system identifies the contour of the ground (terrain) with a ground mapping (GMAP) function.

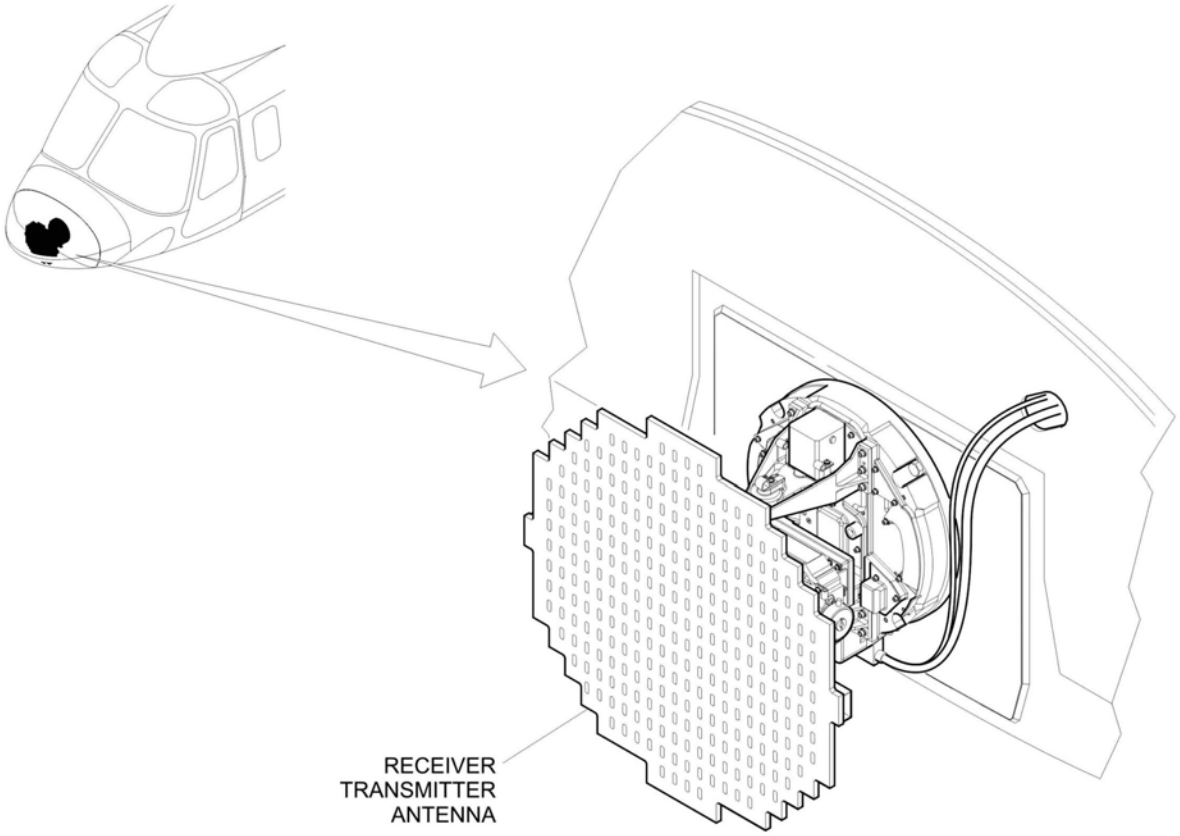
PRIMARY COMPONENTS

The WRS system includes these primary components:

- The receiver transmitter antenna (RTA)
- The weather radar controller.

RECEIVER TRANSMITTER ANTENNA (RTA)

The position of the installation of the transmitter and receiver components is on the rear of the antenna. The installation position of the RTA is on the front bulkhead in the nose of the aircraft below the radome. The RTA connects to the weather radar controller and other avionics systems. The RTA operates from +28 V dc power. An internal fan in the RTA keeps temperatures in a safe range.
The receiver transmitter sends the radar signals received back from the flat-plate antenna. The RTA does examine of the radar signals and controls the interfaces to other system components, thru the MFD and the PFD. The RTA has the mechanism that controls and set the flat-plane antenna.
The receiver uses a low-noise, microprocessor controlled circuit. The receiver has an adjustable video band width that changes with the pulse width and the range selections.
The receiver has manual and automatic controls.The transmitter operated at 9375 MHz (±25 MHz). The transmitter uses a magnetron circuit that supplies 10 kW of power to send radar signals to the antenna. The transmitter has an adjustable pulse width of 1.0 and 2.0 µs that changes with the range and mode selections.
The antenna is an electromechanically-controlled, movable flat-plate. The antenna does a full symmetrical scan every 10 seconds in sectors of 60 degrees/sector scan sweep mode or 120 degrees (normal sweep mode) in azimuth. Sector scan sweep has a range of 30 degrees for each side. Normal sweep has a range of 60 degrees for each side.

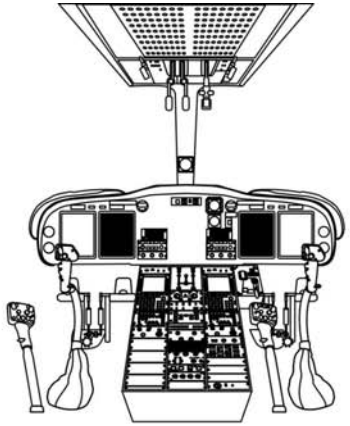


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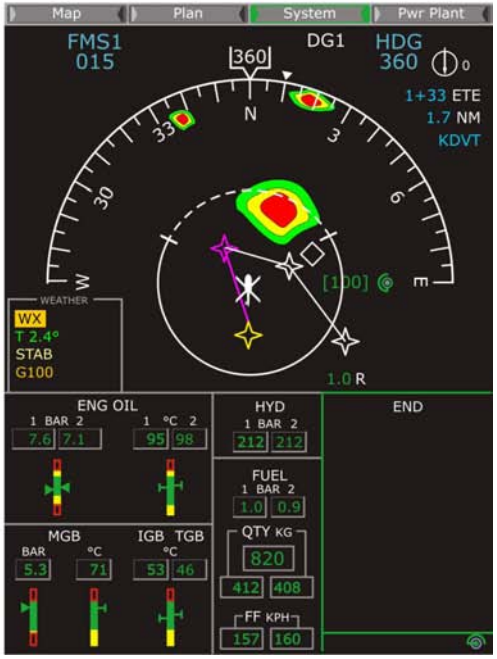
RECEIVER TRANSMITTER ANTENNA LOCATION

WEATHER RADAR CONTROLLER

The install position of the weather radar controller is in the interseat console. Four fasteners hold the controller in its position. Two connectors attach the weather radar controller to the RTA and to other avionics systems. The controller operated from +28 V dc aircraft power.The weather radar controller operates the RTA remotely. The buttons and control knobs on the weather radar controller set the modes of operation for the RTA. They also set the tilt angle, and gain for the antenna. This lets the pilots show different weather conditions on the MFDs and PFDs. The RCT, STAB, TGT, and SECT functions use momentarily buttons. The GAIN, RADAR, and TILT functions use control knobs.







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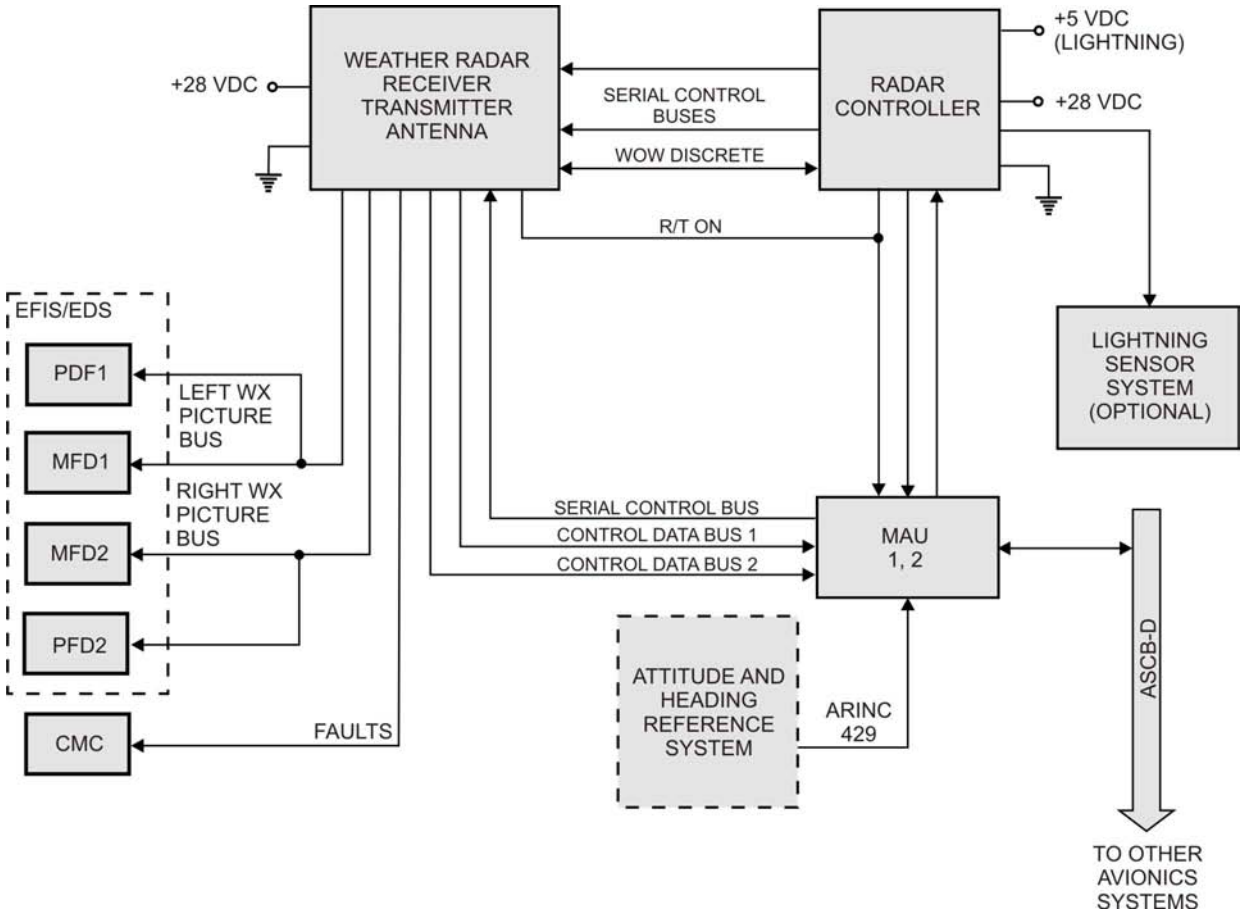
WEATHER RADAR CONTROLLERS

DESCRIPTION OF FUNCTION

The Weather Radar System (WRS) is a remote controlled system that has the Receiver/Transmitter/Antenna (RTA) in one unit. The RTA system operates in the X-band at 9375 MHz (25 MHz).
The system finds the storms along the flightpath of the helicopter and gives the pilots a visual indication of storm intensity and turbulence. The weather and terrain data shows on the Multifunction Displays (MFD) and Primary Flight Displays (PFD). The WRS can receive data from the lightning sensor system (LSS). The data shows the location and the intensity of lightning on the MFDs.

WEATHER RADAR RECEIVER TRANSMITTER ANTENNA (RTA)

The air data system sends true airspeed for the turbulence processors in the RTA on the avionics standard communication bus (ASCB). The rain echo attenuation compensation technique (REACT) override is grounded so selection of the RCT mode on the controller overrides the GAIN control settings.
The RTA receives the mode commands and tilt, gain, and range signals from the weather radar controller on a serial control interface (SCI) bus. The RTA outputs mode, range, and tilt signals to the MAU on two control buses. The RTA outputs scan converted data to the MFDs and PFDs on two weather radar picture data (WXPD) buses. The attitude and heading reference system (AHRS) sends antenna stabilization data to the RTA in ARINC 429 form.
The RTA operates at ranges of 5, 10, 25, 50, 100, 200, and 300 NM full scale in the WX, RCT, or GMAP modes. The range is controlled remotely from the MFD through an ARINC 429 bus. The weather radar data is shown on the PFD or MFD when the WX button is depressed on the display controller. Selection of the WX button will automatically set the ARC mode for the HSI display. The range is set by rotation of the SET knob on the cursor control device (CCD). The range selection increases to 1000 NM full scale in the flight plan (FP) mode. With two MFDs in operation, the pilot or copilot can look at different antenna ranges and tilt angles at the same time. The RTA shows the radar data independently on each MFD.



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WEATHER RADAR SCHEMATIC DIAGRAM

WEATHER RADAR CONTROLLER

The weather radar controller is connects to the RTA SCI bus. The weather radar controlled the two directions of the antenna sweep. The pilot or copilot can adjust the variable gain of the RTA with the GAIN control (push/pull).
Selection of the RCT or TGT mode overrides the variable gain selection and sets the receiver gain to a fixed and calibrated gain value. The variable gain value shows on the MFDs and PFDs with a G indication adjacent to the value.
The WAIT mode starts automatically each time the RTA is energized. In this mode, the system is on but not prepared for full operation. The WAIT mode continues until the magnetron completes its warm-up cycle, and the WAIT indication flashes on the MFDs and PFDs. This mode stops after 5 seconds for a warm start, or 90 seconds for a cold start.
A forced standby (FSBY) safety mode engages during a weight-on-wheels (WOW) condition. In this mode, the RTA is on and prepared for full operation but does not transmit. A FSBY indication shows on the MFDs and PFDs until the WOW condition stops. The pilot or copilot can manually override the FSBY mode. When on the ground, the pilot or copilot can push the STAB button four times in 3 seconds to override FSBY. The FSBY mode is a safety feature that keeps the transmitter from transmitting on the ground to eliminate the X-band microwave radiation hazard.

CONTROL AND INDICATORS

The Weather Radar control panel has the following controls:

1 RCT button: ..... this momentary button sets the react mode ON or OFF.

2 STAB button: ..... this momentary button turns the pitch and roll stability ON or OFF. This button also sets the STAB trim mode. FSBY override occurs when this button is pushed four times in 3 seconds.

3 TGT button: ..... this momentary button enables and disables the radar TGT alert mode.

4 SECT button: ..... this momentary button sets the SECT scan mode to 120° scan or 60° scan.

5 TILT (ACT pull): ..... this single turn rotary control adjusts the antenna tilt angle ± 15° up or down relative to the horizon. When the TILT control is pulled out, the RTA enters the ACT mode.

6 SLV ..... The SLV indication is only used in dual-controller installations. The SLV mode lets one controller be slaved to the other controller.

7 RADAR: ..... Seven-position rotary switch sets the primary modes of operation for the RTA.

RADAR OFF: ..... The OFF position turns off power to the RTA.

RADAR SBY: ..... The SBY position puts the RTA in a non-operational standby mode.

RADAR WX: ..... The WX position sets the weather mode.

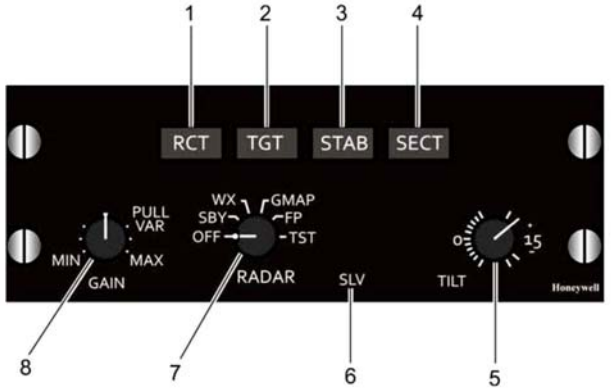
RADAR GMAP: ..... The GMAP position sets the ground map mode.

RADAR FP: ..... The FP position sets the flight plan mode.

RADAR TST: ..... The TST position sets the self-test mode.

8 GAIN: ..... This control is active when pulled. When pushed, the receiver gain is set to a calibrated preset value.





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CONTROLS AND INDICATORS

SECTION 51 - GLOBAL POSITION SYSTEM

GLOBAL POSITIONING SYSTEM – GENERAL

The GPS supplies data to the flight management system (FMS) for flight guidance and other information during flight. The GPS is a 12-channel GPS receiver that receives from the NAVSTAR GPS satellite constellation. In addition the GPS interface the Air Data System (ADS) and the Flight Management System (FMS).
The GPS modules have the primary function of determining the aircraft position from the signal codes. The output data includes:

- three dimensional aircraft position and velocities
- satellite position
- pseudo range
- delta range data

MAIN COMPONENTS

The GPS main components are

- one receiver (GPS module inside MAU 2)
- one antenna

The GPS receiver utilizes the commercial access (C/A) code of the NAVSTARGPS satellite constellation and is capable of operation when selective availability (SA) is activated and deactivated.

PRINCIPLE OF OPERATION

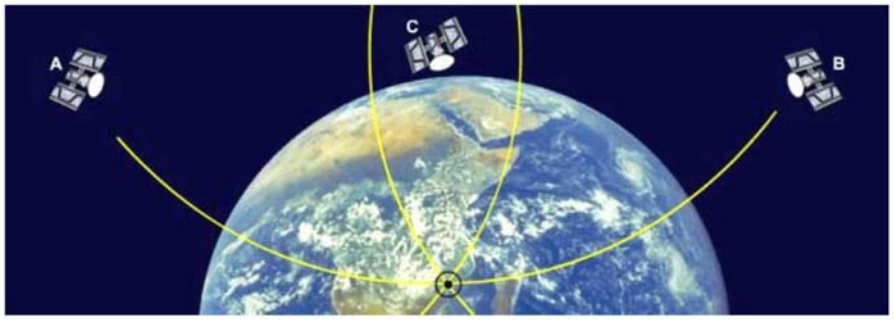
The GPS receiver operations are transparent to the crew. Each GPS has RAIM (Receiver Autonomous Integrity Monitor) outputs for the current position and time in the form of horizontal and vertical integrity limits (HIL and VIL). In order to compute RAIM, the GPS must have a minimum of five satellite signals. The FMS does not accept GPS data unless a valid RAIM figure is available.
Therefore the GPS RAIM function assures the integrity of the data because the GPS RAIM function can detect satellite failures. It isolates and removes failed satellites when it is tracking a sufficient number of satellites for measurement redundancy.
The FMS uses predictive RAIM to determine the integrity levels at specific locations/times to support a non precision approach and the flight planning activities of the pilot.
The GPS has the following types of RAIM predictions:

- destination
- alternate waypoint

The destination and alternate waypoint predictions are made at specific locations or they are the estimated time of arrival (ETA) when the FMS makes the request for flight planning purposes. Satellites can be manually deselected or enabled for destination and alternate waypoint predictions.

CONTROLS AND INDICATORS

The performance of the GPS can be monitored by the GPS 1 STATUS pages (1/2 and 2/2) and PREDICTIVE RAIM page (1/1).
Predictive RAIM (PRAIM) calculates the estimated value of the HIL at some future place and time. The FMS can interrogate the PRAIM function of the GPS through the ARINC 429 interface. However, RAIM integrity performance requirements cannot be selected with the GPS.


A
C
B





ICN-39-A-345100-G-A0126-01315-A-01-1

GPS CONTROLS AND INDICATORS

GLOBAL POSITIONING SYSTEM – CAS CAUTION MESSAGES			
CAS CAPTION	**FAILURE DESCRIPTION**	**PROCEDURE NAME**	**AW139-RFM-4D**
GPS FAIL	GPS system failure	GPS FAIL	Section 3 EMERGENCY AND MALFUNCTION PROCEDURES AVIONICS
1(2) FMS/GPS MSCP	Miscompare between FMS and GPS position data	FMS/GPS MISCOMPARE	
FMS/GPS MSCP UNAVL	FMS/GPS miscompare function not available due to FMS or GPS DATA INVALID	FMS/GPS MISCOMPARE UNAVAIL-ABLE	

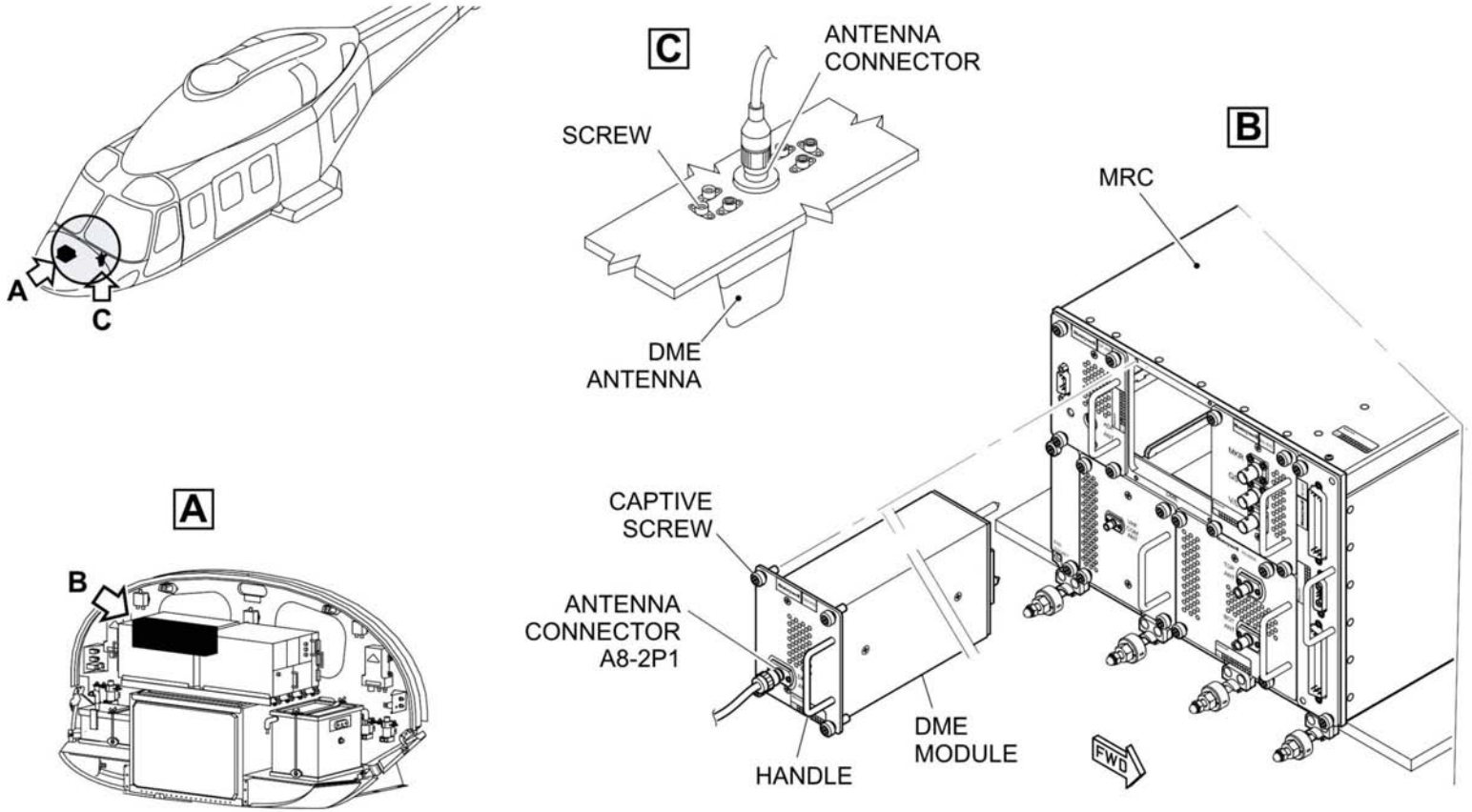
SECTION 52 - DME SYSTEM

DESCRIPTION

The Distance Measuring Equipment (DME) system supplies the data for in-flight navigation, terminal navigation, and area guidance.
The Distance Measuring Equipment (DME) system has the components that follow:

- The DME module
- The DME antenna.

The DME module is installed in the MRC2, which is a part of the modular radio system. Four captive screws attach the DME module in its position in the MRC2. The electrical connector connects the DME module to the DME antenna.
The installation position of the DME antenna is on the bottom side of the fuselage. Six screws attach the DME antenna to the fuselage structure. The coaxial connector connects the DME antenna to the DME module.



ICN-39-A-345200-G-A0126-01316-A-01-1

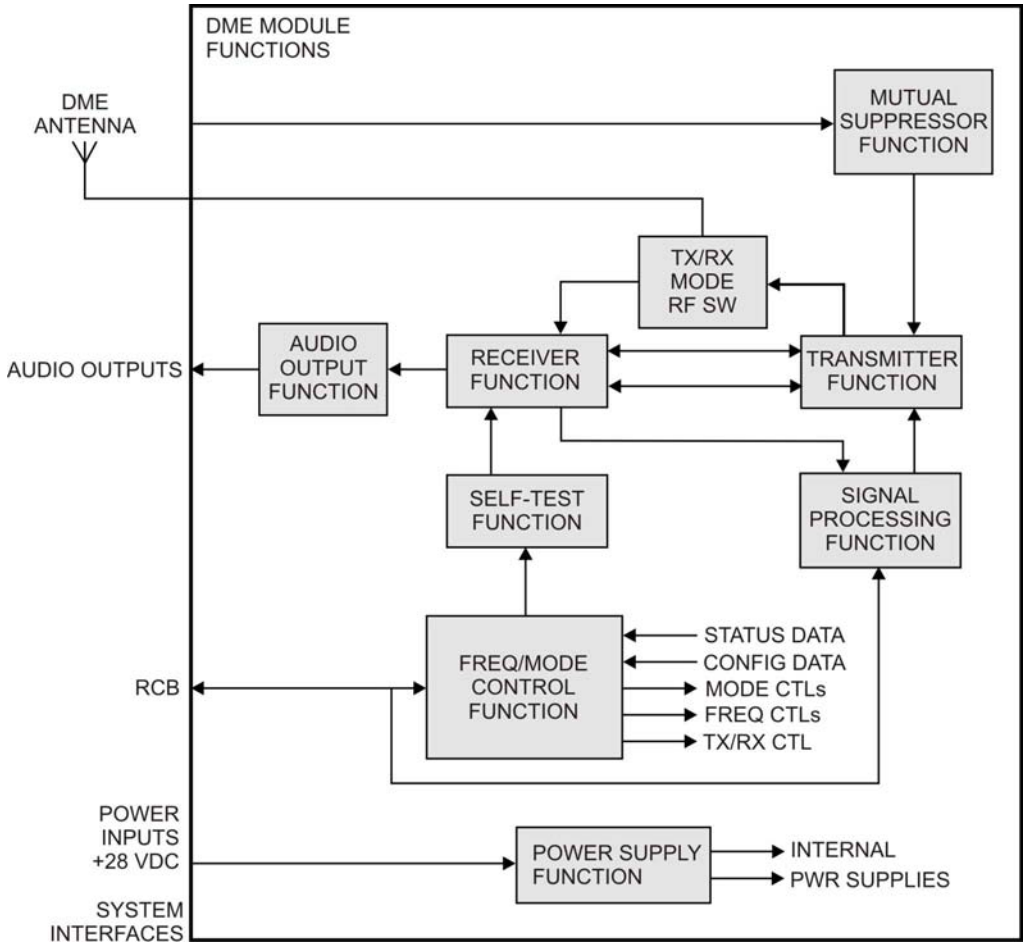
DME MODULE AND DME ANTENNA LOCATION

OPERATION

The DME system operates with radio pulses in the frequency range of 960 to 1215 MHz. The airborne system transmits from 1025 to 1150 MHz to a ground station. The ground station (beacon) transmits from 960 to 1213 MHz.
The DME system does the functions that follows:

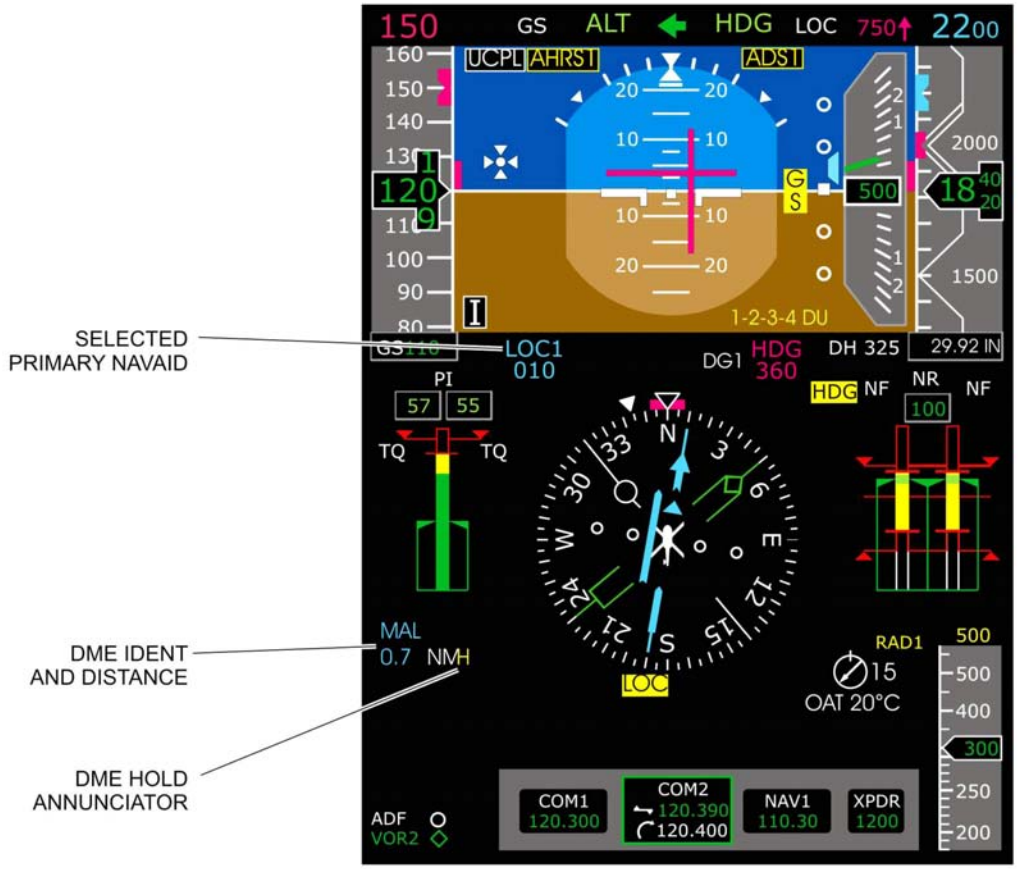
- Mutual suppressor
- Transmitter
- Receiver
- Audio output
- Signal processing
- Frequency/Mode Control
- Power supply
- Self/Mode

The PFD and MFD supply integrated displays of primary flight, navigation, engine performance, and aircraft systems data, and caution/warning indications.
The MFD receives the distance data from the DME module and from the FMS, when is installed. The MFD shows the distance indication adjacent to the Horizontal Situation Indicator (HSI).
When the source of NAV data shown is the DME-hold, a DME HOLD indication shows adjacent to the distance indication on the MFD.
The MCDU is the primary control unit for the radio system. The MCDU controls the modes, frequencies, and codes of all the modules in the MRC, included the DME system.



ICN-39-A-345200-G-A0126-01317-A-01-1

DME SYSTEM SCHEMATIC DIAGRAM



ICN-39-A-345200-G-A0126-01318-A-01-1

VOR/ILS/MB AND DME – INDICATIONS

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SECTION 53 - AUTOMATIC DIRECTION FINDER (ADF) SYSTEM

DESCRIPTION

The Automatic Direction Finder (ADF) system supplies the relative bearing data from a ground station for in-flight navigation, terminal navigation, and area guidance. This data gives the aircraft direction to a ground station with relation to the heading.

The ADF receiver module, installed in the MRC2, receives RF signal from the ADF antenna, compute the relevant information and sends its data to the NIM. The NIM sends this NAV data on the ASCB-D bus to the NICs in the MAUs and to the EDS.

The ADF data, received by NICs, is provided to the FMS processor modules through the MAU backplanes. The FMS uses NAV data from the ADF system to do its NAV functions.

The electronic display system (EDS) shows ADF data that gives the radio bearing relative to the aircraft heading.

The MRC2, also, transmits the ADF audio signals through the digital audio bus to each audio panel in the audio system.

The ADF system uses two bandwidths:

- A narrow band mode to reduce noise during navigation.
- A wideband mode to improve the quality of voice signals.

The ADF system is composed by the components that follow:

- DF-855 Automatic Direction Finder (ADF) Module.
- AT-860 Automatic Direction Finder (ADF) Antenna.

Df-855 Adf Module

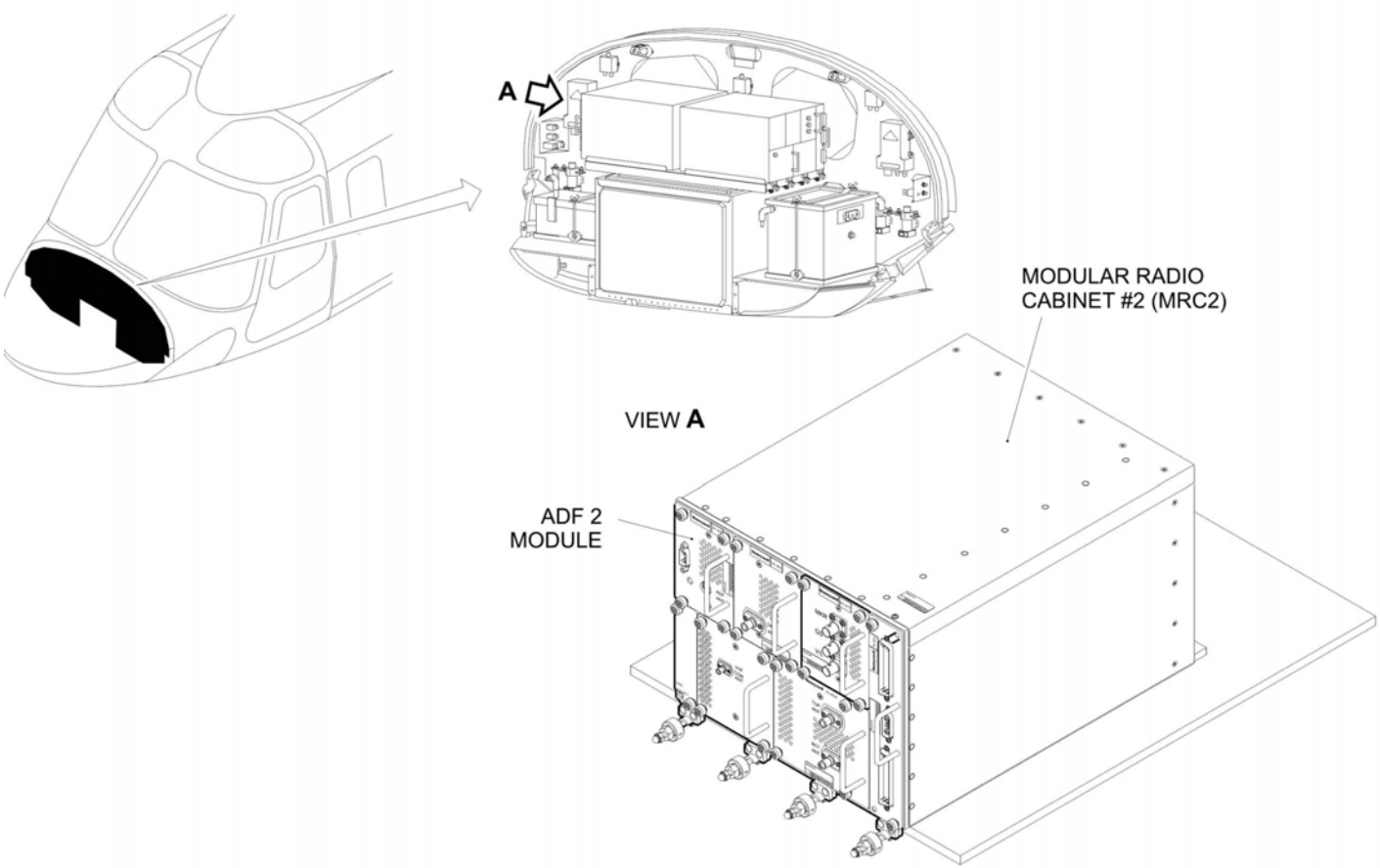
The ADF module is installed in the MRC2, is powered by +28 V dc, and supplies +15 V dc to the amplifier in the ADF antenna.
The ADF module supplies steering data to low-frequency transmitters (beacons) with set frequencies from 190 to 400 kHz, and 1600 to 1799 kHz.
It also supplies steering data to the following stations:

- Broadcast Stations that Operate with Amplitude Modulation (AM).
- Marine Stations that Operate in the Frequency Range of 100 to 190 kHz.
- Stations on the High Frequency (HF) Distress Frequency of 2182 kHz.

The ADF module provides the relative bearing data to the NIM of the MRC2 through the internal bus, but it is also able to supply data to external users on the following way:

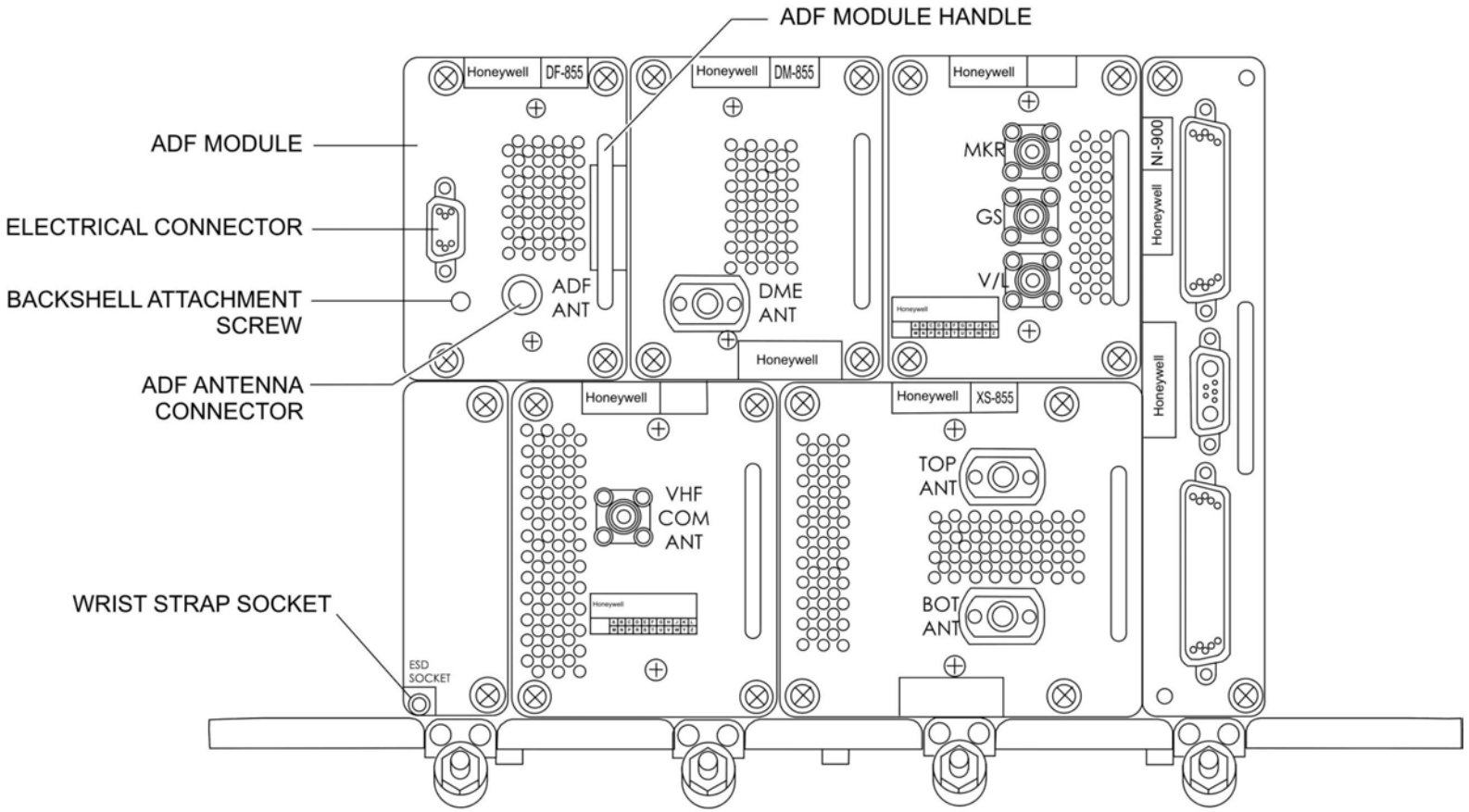
- as ARINC-407 synchro data,
- as direct current (dc) sine/cosine data, and
- as RS-422 data.

The ADF module has a mute (cancel) function that cancels the radio interference caused by HF signals transmitted from the aircraft. (This interference can cause some avionics to show incorrect ADF bearing indications).



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MRC2 AND ADF2 MODULE COMPONENT LOCATION (SHEET 1 OF 2)

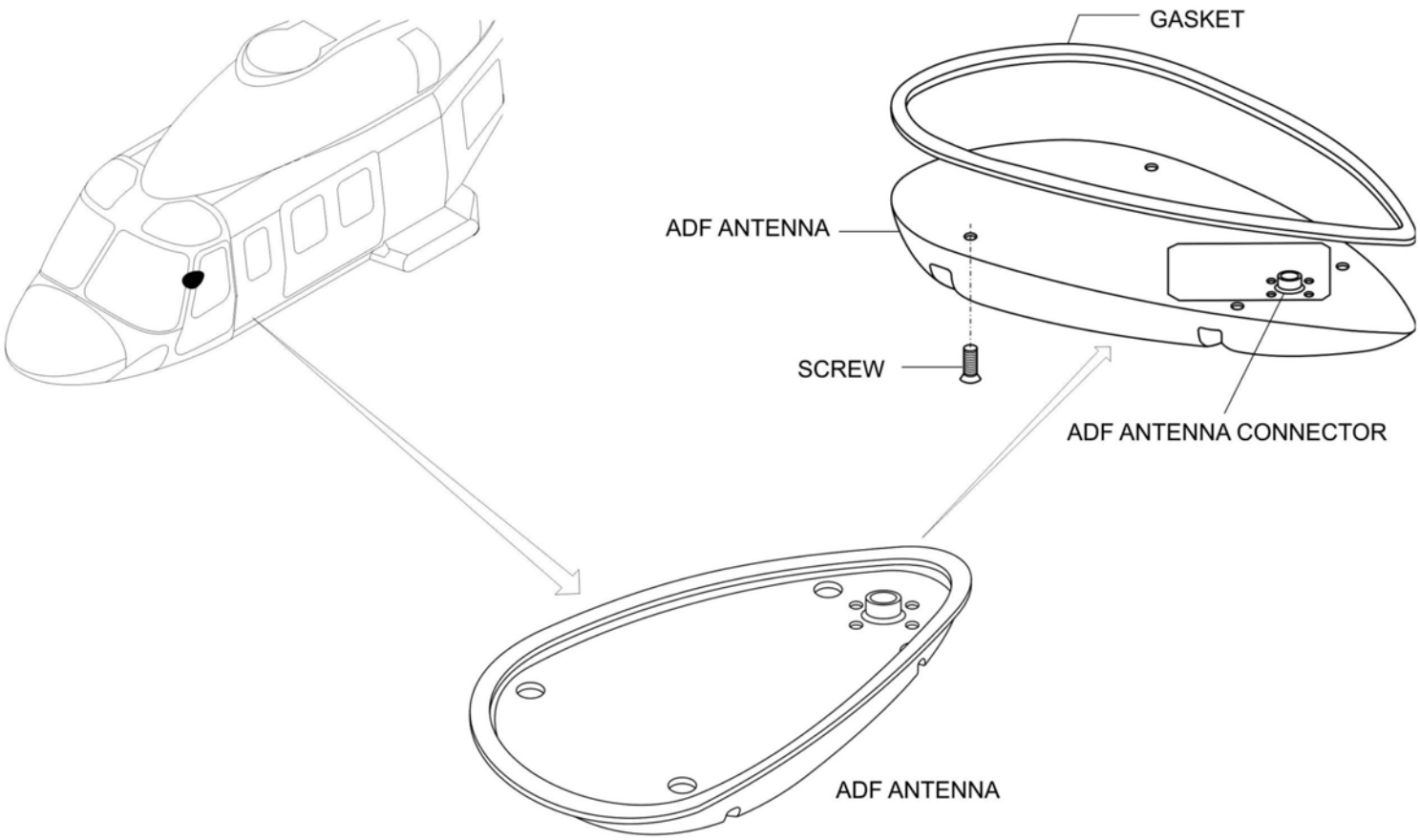


ICN-39-A-345300-G-A0126-00864-A-01-1

MRC2 AND ADF2 MODULE COMPONENT LOCATION (SHEET 2 OF 2)

At-860 Adf Antenna

The ADF antenna system contains an internal sense antenna, sine/cosine loop antennas, an amplifier,
and a self-test function. The sense antenna is a vertically-polarized antenna that receives the AM audio signals. The sine/cosine loop antennas are horizontally-polarized, perpendicular antennas that receive the relative bearing signals. The amplifier increases the input signals from the internal antennas. The self-test function does a test of the ADF system components.
The ADF antenna has a coaxial connector that connects to the ADF module. The antenna is installed on the bottom center outer fuselage.



ICN-39-A-345300-G-A0126-00865-A-01-1

ADF2 ANTENNA LOCATION

OPERATION

The ADF system operates in the frequency range of 190.0 to 1799.5 kHz in increments of 500 Hz (for usual operation). It also tunes frequencies between 100 and 190 kHz, and monitors the maritime emergency frequencies between 2181 and 2183 kHz.
In case of an airborne HF radio starts to transmit, the ADF module holds the ADF bearing pointer.
If the HF radio transmits continuously for more than 5 seconds, the electronic ADF bearing pointer, shown on the PFD, goes out of view.
These conditions continues until the HF radio stops in order to prevent possible errors caused by bearing indications that are not accurate.
The following table shows the four modes of operation for the ADF system

MODE	DESCRIPTION
ANT (antenna)	The system receives the ADF signals but does not calculated the relative bearing. Audio sensitivity and range are set to maximum.
ADF	The system receives the ADF signals and calculates the relative bearing. There is some decrease of received audio range and sensitivity.
BFO (Beat frequency oscillator)	The system receives the continuous wave (CW) signals. The system uses a beat frequency oscillator (BFO) to sense these signals.
VOICE	The system increases the intermediate frequency (IF) bandwidth of the ADF module to make the quality of the ADF audio signals better. This mode does not supply bearing.

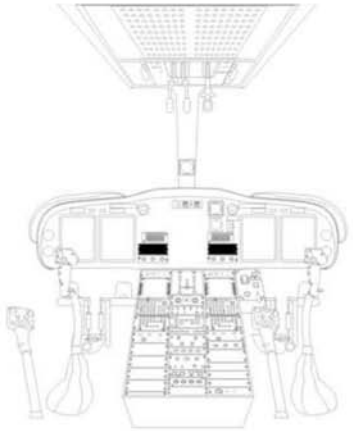
The VOICE mode supplies high quality audio signals for the headsets and flight compartment speakers. The high quality audio is a result of the wider bandwidth available in the VOICE mode. To find the direction to the ADF station, narrow band reception is necessary to increase the bearing accuracy. Because of the wider bandwidth, it is possible to hear BFO interference when the system monitors different ADF stations that are less than 3 kHz apart. To prevent this interference, use the ANT mode.

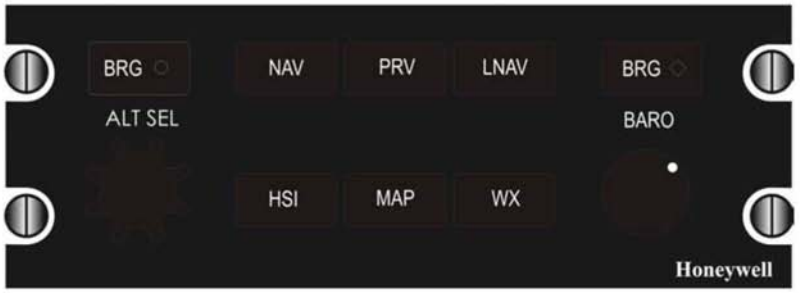
CONTROLS AND INDICATORS

The bearing data from the helicopter, provided by the Automatic Direction Finder (ADF2) over the ASCB-D, is shown on the PFD. This data is shown on the PFD by the diamond pointer when selected by the Diamond Bearing 2 Button of the Display Controller.

Adf Tuning Function

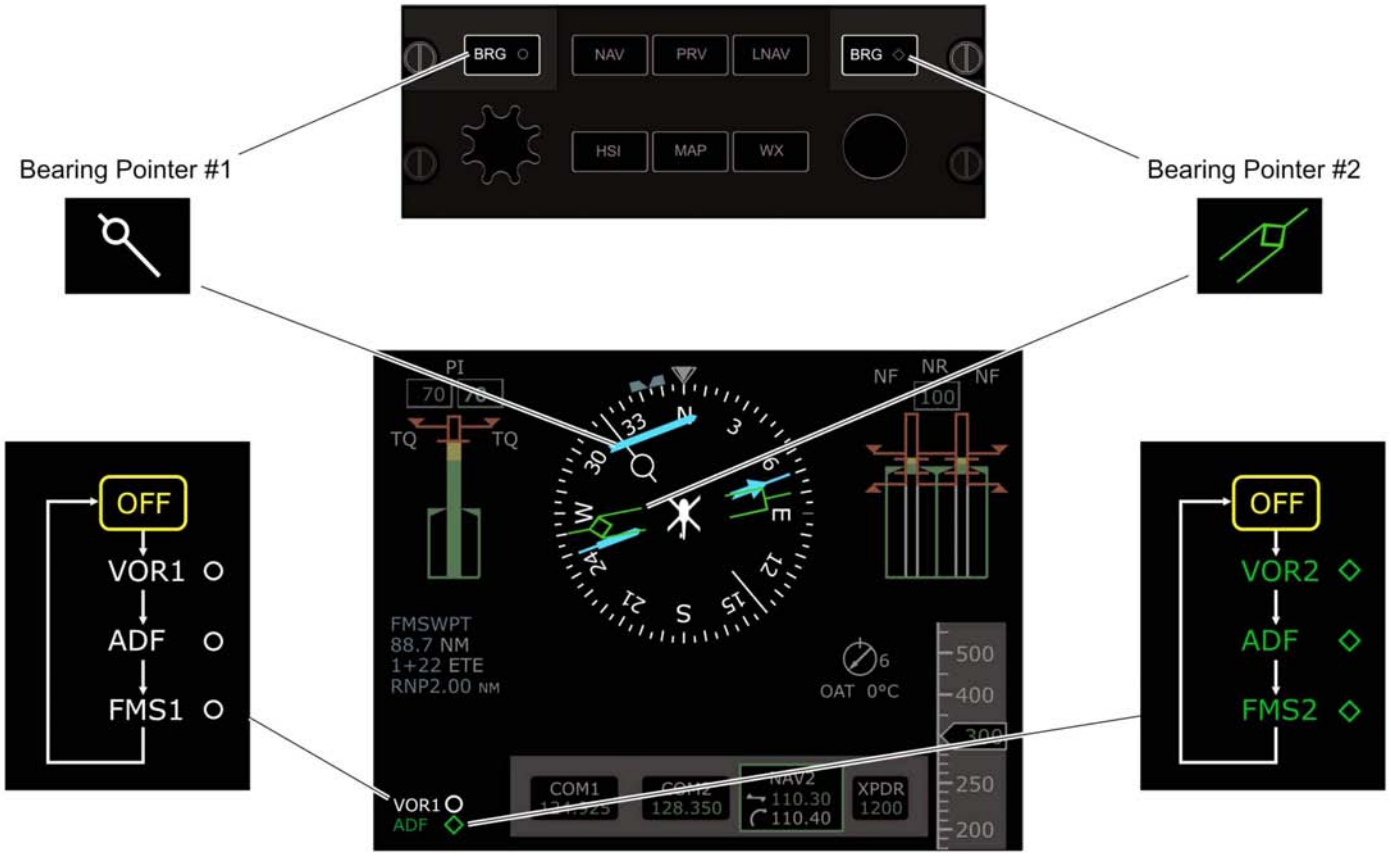
The ADF tuning function is done with the MCDU through the ASCB-D, in case of ASCB-D failure a backup MCDU tuning mode lets the pilot or copilot tune the on-side COM and NAV through a direct ARINC 429 link from the MCDU to the radios. Because of this, and having the only ADF2 system installed onboard, during tuning backup mode the ADF system in not available.





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CONTROLS AND INDICATORS



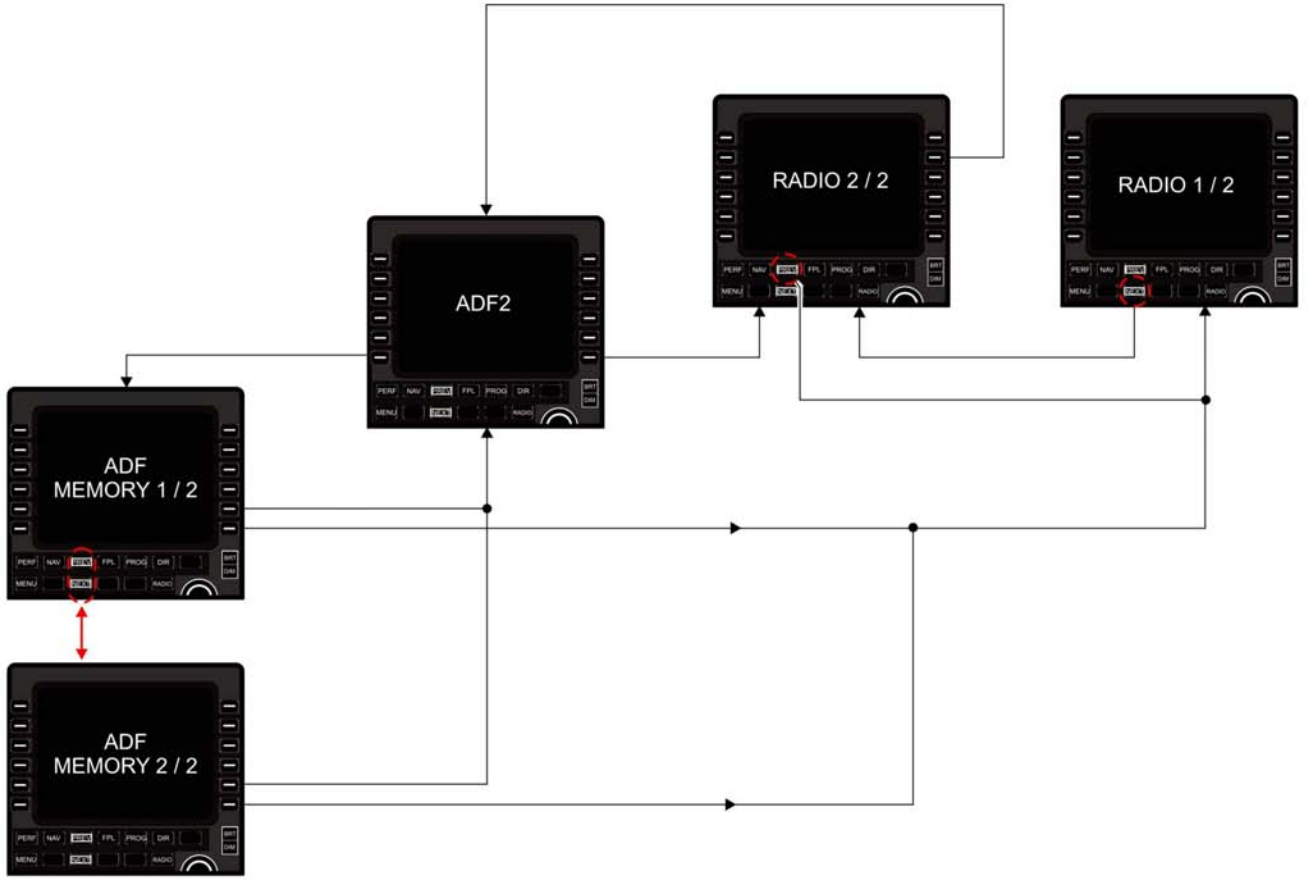
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BEARING POINTER SELECTION

Adf System Operation With The Multifunction Control Display Unit (Mcdu)

The MCDU is the primary control unit for the PRIMUS EPIC radio system. The MCDU controls the operative mode and frequency tune for the ADF2 system through the pages as follows:

- RADIO 2/2: RADIO 2/2 page displays the radio data for the systems ADF2, COM radios, NAV radios and NAV/ COM radios.
- ADF 2: The ADF2 page displays the active, preset, and selected memory frequencies for the automatic direction finder, the controls for the active mode (antenna or ADF) and the beat frequency oscillators (BFO) setting (ON or OFF). It is also used to access the ADF memory pages.
- ADF MEMORY: The ADF MEMORY 1/2 and ADF MEMORY 2/2 pages are utilized to store up to 12 ADF memories channels each one having an own associated frequency. In addition to entering or dialing-in frequencies for each memory location, a text label of up to 8 characters can be entered for each stored frequency. The default label for each memory is MEMORY, a dash, and the memory number, with the memory number always on the outboard edge of the display.



ICN-39-A-345300-G-A0126-00868-A-01-1

ADF OPERATIONS WITH MCDU (SHEET 1 OF 4)



ICN-39-A-345300-G-A0126-00869-A-01-1

ADF OPERATIONS WITH MCDU (SHEET 2 OF 4)



ICN-39-A-345300-G-A0126-00870-A-01-1

ADF OPERATIONS WITH MCDU (SHEET 3 OF 4)



NOTE: ADF MEMORY 2/2 page is the same as ADF MEMORY 1/2 page. The only difference is the locations numbered from 7 to 12.

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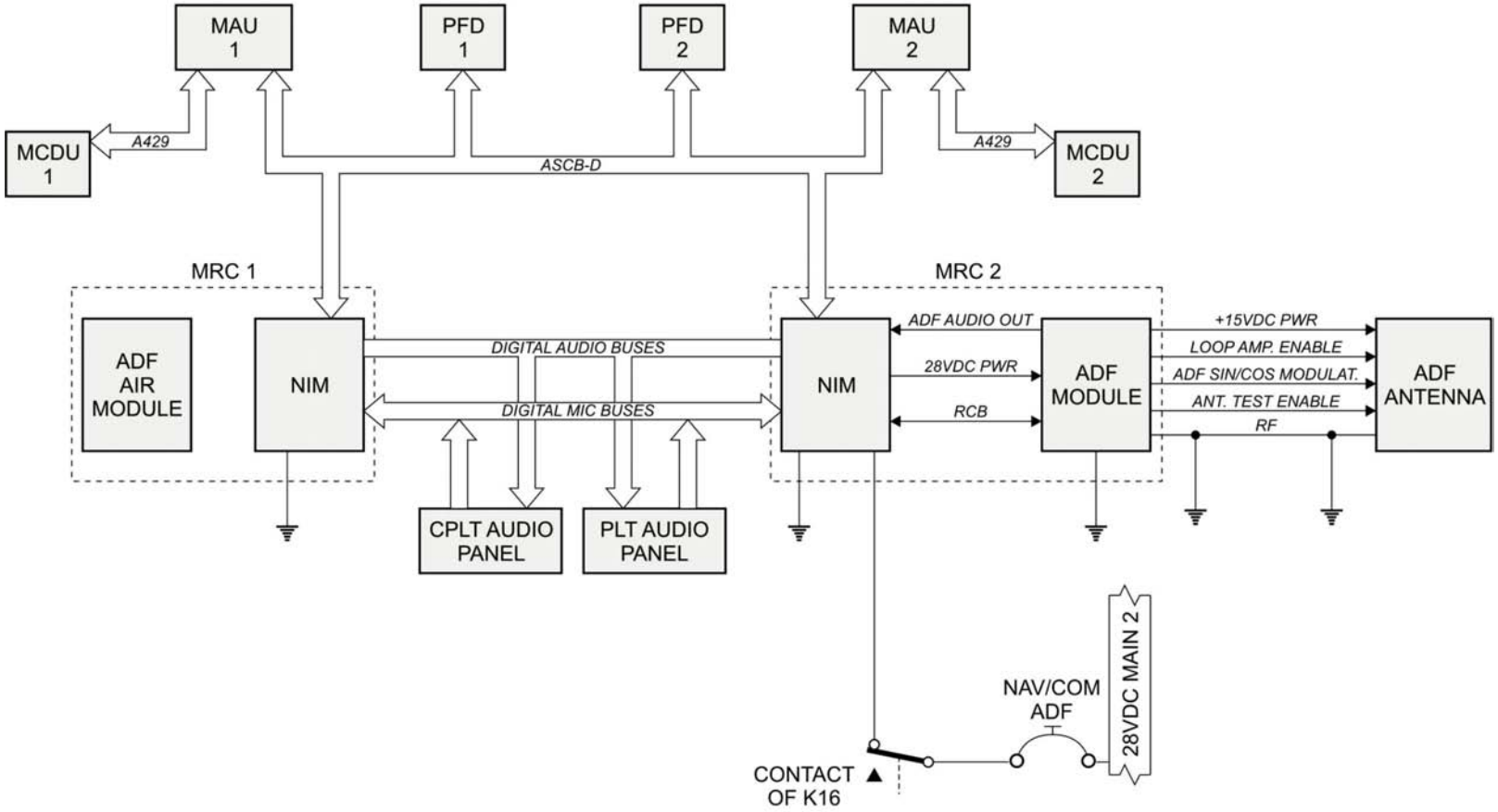
ADF OPERATIONS WITH MCDU (SHEET 4 OF 4)

CAPTIONS

No-one cautions message is available for the CAS window in the event of ADF system failure.
Acknowledgement of a not availability of the ADF bearing data or ADF system failure occurs as follows:
ADF bearing pointer frozen: If the onboard HF radio starts to transmit.
ADF bearing pointer out of view: in case of ADF failure or if the onboard HF radio transmits continuously for more than 5 seconds.

LEADING PARTICULARS DF-855 AUTOMATIC DIRECTION FINDER (ADF) MODULE

- Power Requirements..... 28VDC nominal
  30.3VDC maximum
  22VDC minimum
- Circuit Breaker..... 28VDC 3.0 amps
- Types of Reception..... CW, AM
- Frequency Range..... 100.0 to 1799.5 KHz in 0.5 KHz steps
- Intermediate Frequencies (IF):
  - First IF..... 10.7 MHz
  - Second IF..... 4.3 MHz
- Bandwidth:
  - ANT, ADF and BFO Modes..... ≤ KHz at six db
    ≤10 KHz at 80 db
  - Voice Mode..... ≥ 8.0 KHz at three db
    ≤ 26 KHz at 40 db
- Sensitivity:.....
  - ANT Mode: better than 50 mV/ meter for six db S + N/N for 400 Hz, 30% modulation-
  - ADF Mode: better than 100 mV/ meter for six db signal to noise ratio for 400 Hz, 30% modulation
  - BFO Mode: better than 70 mV/ meter for six db signal to noise ratio for 400 Hz, 30& modulation.
- Spurious Response:..... at least 80 db below the desired response at all frequencies.
- Audio input: ..... 100 mW into a 600 load with < 10% distortion
- Audio Frequency Response:.....
  - ANT, ADF and BFO modes: not more than ± 3 db from 350 to 1100 Hz
  - Voice Mode: not more than ± 4 db from 350 to 3200 Hz



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ADF ELECTRICAL SCHEMATIC

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SECTION 54 - ATC TRANSPONDER SYSTEM

GENERAL

The (ATC XPDR) system has the components that follow:

- The ATC XPDR module
- The ATC XPDR top antenna
- The ATC XPDR bottom antenna.

ATC XPDR MODULE

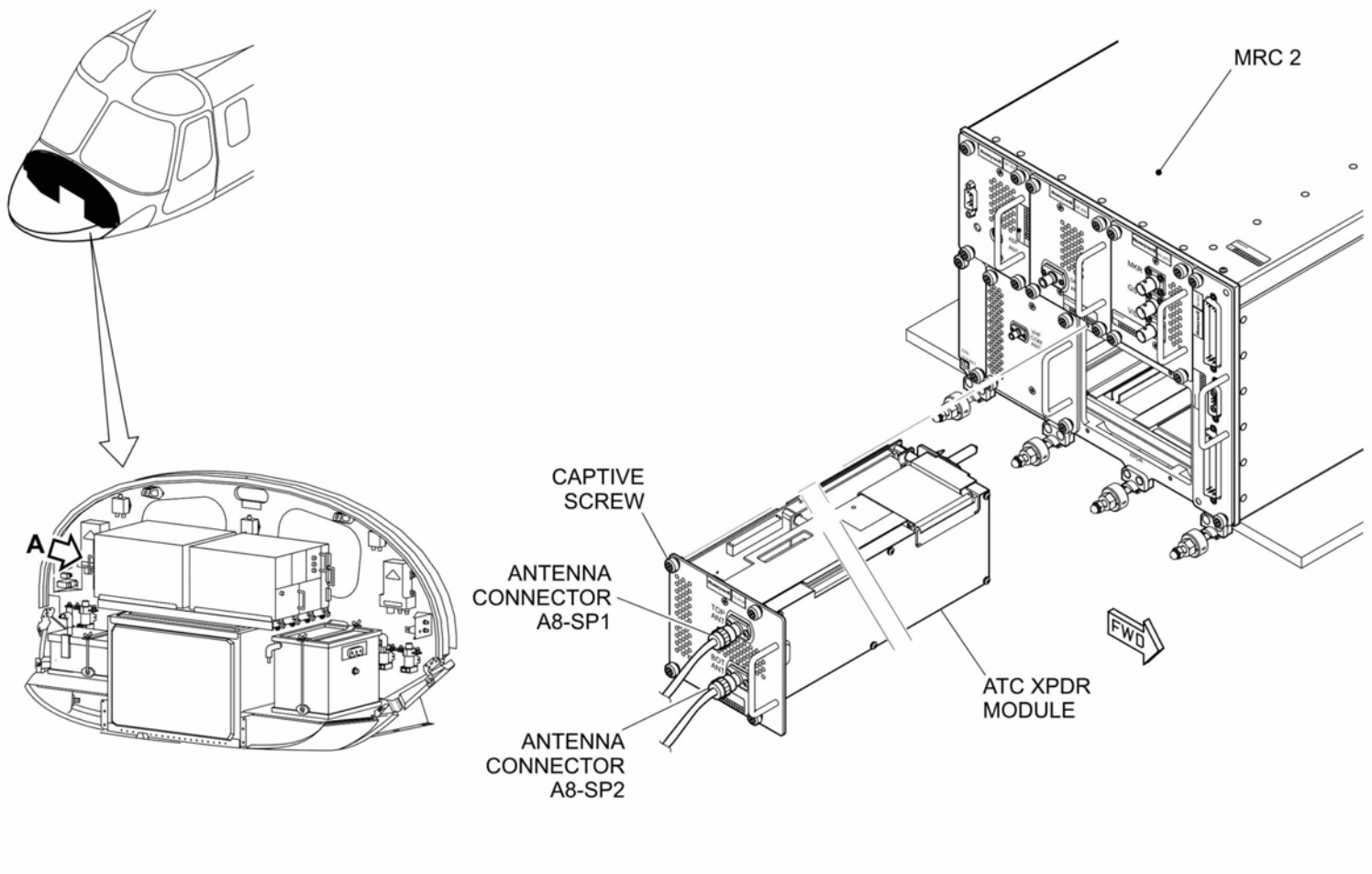
The ATC XPDR module is installed on the Number 2 Modular Radio Cabinet (MRC 2). The MRC 2 is installed on the right electronic equipment bay (EE-bay) in the nose compartment. Four captive screws attach the ATC XPDR module to the MRC2. The antenna connectors A8-P1 and A8-P2 connect the ATC XPDR module (A8) to the ATC XPDR top antenna (E13) and the ATC XPDR bottom antenna (E10).

ATC XPDR TOP ANTENNA

The ATC XPDR top antenna is installed in the top side of the rear fuselage. Six screws attach the ATC XPDR top antenna to the fuselage structure. The antenna connector E13P1 connects the ATC XPDR top antenna to the ATC XPDR module.

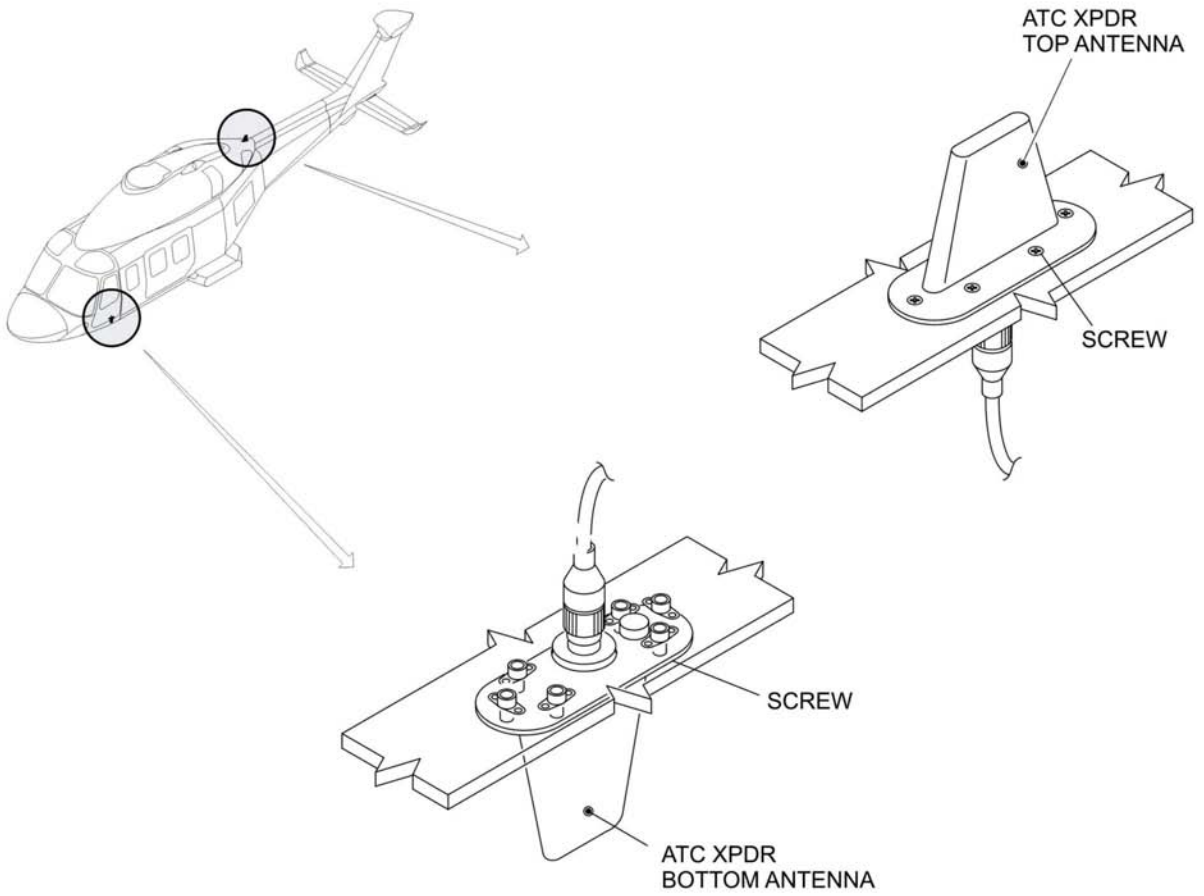
ATC XPDR BOTTOM ANTENNA

The ATC XPDR bottom antenna is installed in the left bottom side of the center fuselage. Six screws attach the ATC XPDR bottom antenna to the fuselage structure. The antenna connector E10P1 connects the ATC XPDR bottom antenna to the ATC XPDR module.



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ATC XPDR MODULE LOCATION



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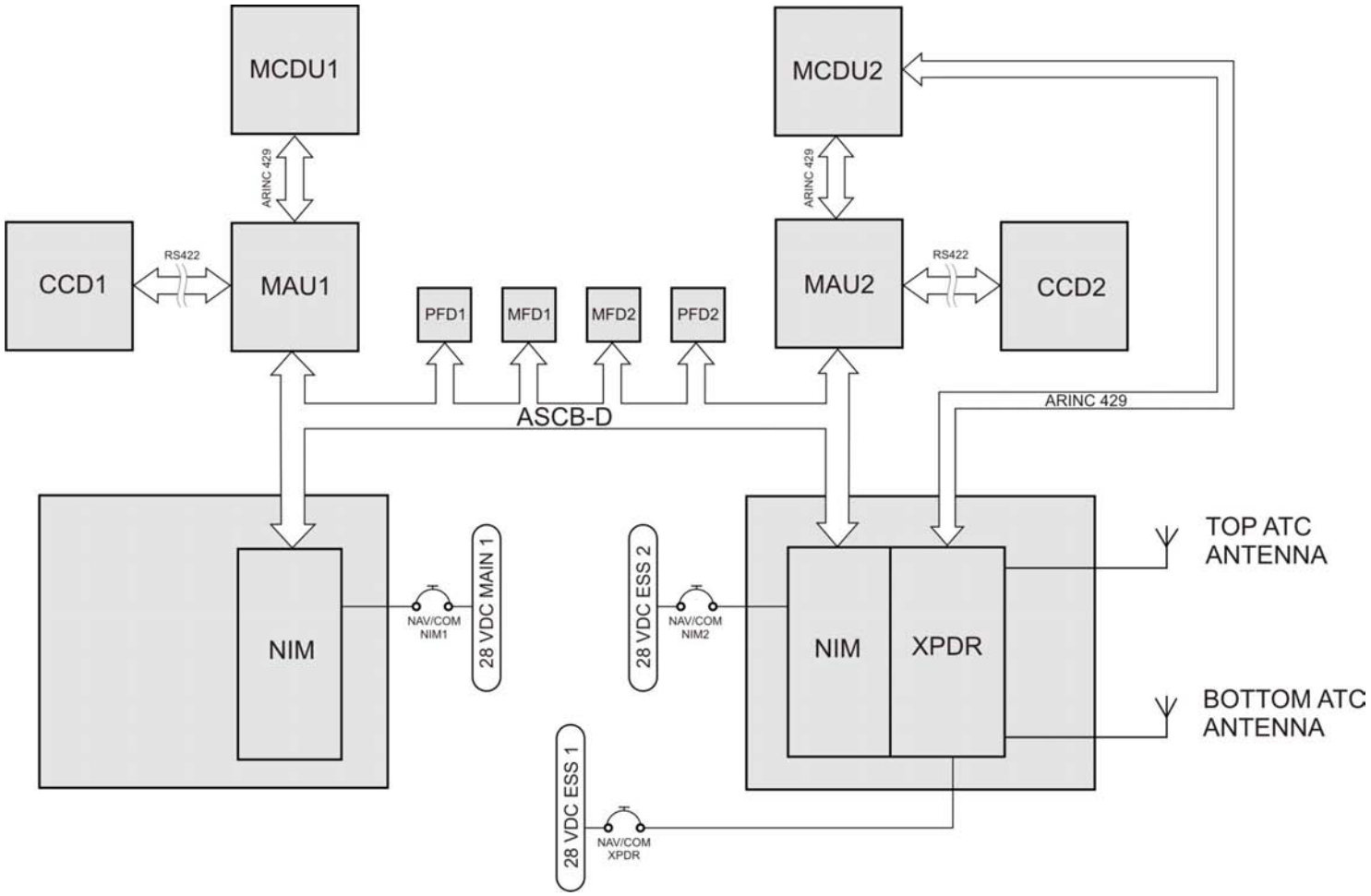
ATC XPDR ANTENNAS LOCATION

SYSTEM OPERATION

The Air traffic Control Transponder (ATC XPDR) system transmits and receives the identification code (ID) and the barometric altitude (ALT) data of the helicopter. The aircrafts and the ATC ground stations use this data to prevent midair collisions with other aircraft. The system operates with the Traffic Alert and Collision Avoidance System (TCAS) and the Airborne Data Link Processor (ADLP) system.

The ATC XPDR system is a part of the modular radio system. It is a remote-controlled transceiver system. It operates with the ATC-Radar Beacon System (ATCRBS) and the Modes A, Mode C, and Mode S transponder systems.

The ATC XPDR system receives interrogation pulses and transmits reply pulses. The reply pulses give the ground station access to computer data such as aircraft distance, bearing, altitude, and aircraft identity. The pilot and copilot can control the XPDR mode of operation and record the reply code by the multifunction control display-unit (MCDU). The ATC XPDR system is connected to the display system through the Avionics Standard Communication Bus, version-D (ASCB-D). The ASCB-D is connected to the back of the modular radio cabinet (MRC). The system transmits and receives data from the Network Interface Module (NIM) in the MRC, through the radio communication bus (RCB). The NIM supplies the interface to the ASCB-D.



ICN-39-A-345400-G-A0126-00875-A-01-1

ATC XPDR SCHEMATIC DIAGRAM

ATC XPDR SYSTEM FUNCTION

The ATC XPDR system has five modes of operation. The angle at which the antenna receives the reply gives the azimuth of the aircraft. The ATC XPDR system receives the interrogation pulses at 1030 MHz and transmits the reply pulses at 1090 MHz.
The ATC XPDR system has the components that follow:

- ATC XPDR module
- ATC XPDR antennas

One single ATC XPDR antenna is necessary for the XPDR module. The second antenna is installed for diversity to make the data acquisition better.

Mode	Description
STANDBY	The XPDR system is on, but it does not transmit the altitude data.
ALT OFF	The XPDR system transmits the reply in Mode A and S, but it does not transmit the altitude data.
ALT ON	ATC ALT the XPDR system transmits the reply in Modes A, C and S and it transmits the altitude data.
TA	The TCAS traffic advisory (TA) mode is on. (This mode operates when a TCAS system is installed in the aircraft).
TA/RA	The TA/resolution advisory (RA) mode is on. (This mode operates when a TCAS system is installed in the aircraft).

XPDR MODULE FUNCTION

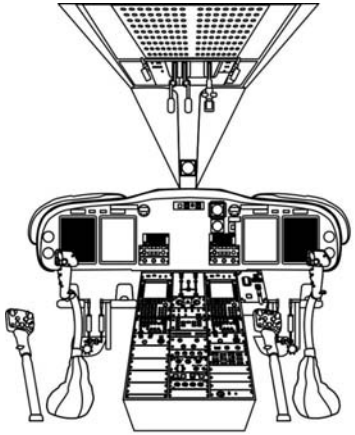
The XPDR module is a two-channel radar transceiver that operates with the ATCRBS ground stations and the Modes A, C and S transponders. The modes are as follows:

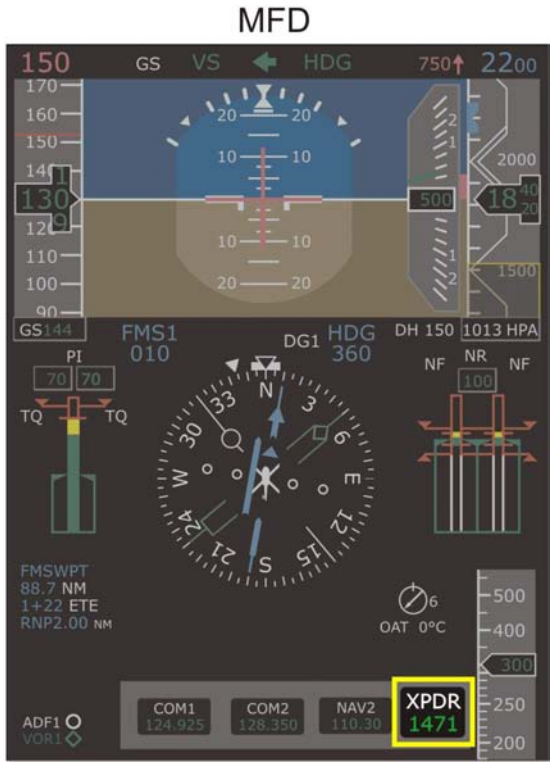
- The Mode A reply contains the ID code of the helicopter.
- The Mode C reply contains the barometric altitude (ALT) data of the helicopter (with a tolerance of ± 100 ft)
- The Mode S reply contains the ID code, the ALT data and the Mode S address of the helicopter.

The XPDR module is connected to two different antennas. When the system operates, the radio frequency (RF) receiver in the XPDR module automatically sets the antenna that sends the best signal and it sends the reply to that antenna.

ATC XPDR ANTENNAS FUNCTION

The ATC XPDR system has one antenna on the top and the other antenna on the bottom of the fuselage to make the interrogation/reply signals acquisition better. They send/receive these signals to/from an ATCRBS ground station or/and Modes A, C and S transponder.





ICN-39-A-345400-G-A0126-03008-A-01-1

PFD ATC XPDR INDICATION



ICN-39-A-345400-G-A0126-00877-A-01-1

MCDU ATC XPDR INDICATION

SECTION 60 - FLIGHT MANAGEMENT SYSTEM

GENERAL

The Flight Management System (FMS) is based on the Honeywell FMS. The function of the FMS is to give flight planning capability, navigation information and flight performance data. The FMS manages flight details from takeoff to touchdown. These details include Standard Instrument Departures (SID), Standard Terminal Arrival Routes (STAR) and non precision approaches with missed approaches.
The FMS combines inputs of other aircraft systems (GPS, DME, VOR, AHRS, ADS) to output navigation, lateral and vertical commands and aircraft performance predictions. FMS uses GPS as long range sensor and DME/DME and VOR/DME as short range sensors.

- Primary functions of FMS are position computation and flight planning. These functions work with the associated guidance in both lateral and vertical axes. The FMS Database is essential to these functions in order to easily retrieve, navaids, airways, procedures, airports, other navigation data and store and retrieve waypoints and flight plans.
- Secondary function of FMS are performance including fuel management, time estimates for the flight, vertical navigation estimates, stored flight plans and other functions.

Provided the FMS is receiving usable signals, it has been demonstrated capable of and has been shown to meet the accuracy specifications of:

- VFR/IFR en route domestic, terminal, and instrument approach (GPS, VOR, VOR-DME, TACAN, NDB, NDB-DME, RNAV) operation using the WGS-84 (or NAD 83) coordinate reference datum in accordance with the criteria of AC 25-11, AC 00-314, AC 20-130A, FMS PS 7028826.

Satellite navigation data is based upon use of only the NAVSTAR Global Positioning System (GPS).

Refer to the Honeywell PRIMUS EPIC® PILOT'S GUIDE for a detailed description of the FMS.

FMS DATA BASE

The FMS Database consists of two parts: a navigation database and a custom (or pilot defined) database.
The navigation data base (NDB) cannot be changed by the pilot but, using the custom database, the pilot can customize the FMS by defining waypoints (WP) and storing flight plans (FP).

NAVIGATION DATA BASE

The NDB contains worldwide coverage of navaids, airways, Standard Instrument Departure / Standard Terminal Arrival Route (SID/STAR) procedures, approach procedures, airports, heliports and runways as following:

- navaids (VOR, DME, ILS, MLS, TACAN, NDB)
- Airports and Heliports
- Runways and Airways
- SID and STAR
- Approaches
- Named Intersections
- Outer Markers

CUSTOM DATA BASE

The custom database consists of pilot defined WP (up to 1000) and stored FP (up to 3000).

FLIGHT PLAN - GENERAL

A flight plan is presented as a series of legs that are sequenced according to their occurrence during the flight. The legs are bounded by waypoints.
During flight, the active flight plan automatically sequences so that the first leg of the active flight plan is the active leg that is referenced to the guidance parameters. Normally, the FMS sequences before the waypoint for an inside turn when the aircraft is on or close to on course: a flyover option is also available to the pilot. If the aircraft is not on course, the normal sequence occurs no later than a point abeam of the waypoint. Some waypoints have unique sequence criteria.
For example, a holding fix is a flyover waypoint. The holding fix should be over flown before entering or exiting holding. Some waypoints in SID and STAR procedures also have unique sequence criteria.
The FMS is programmed to automatically comply with these requirements. There are situations where the sequence criteria cannot be satisfied by the FMS. Under these conditions, the pilot must perform the sequence manually to aid the FMS.

NAVIGATION OPERATION DESCRIPTION

The FMS navigation function is responsible for tuning the NAV and DME radios. The FMS chooses the best NAVAID to tune on the radios for VOR/DME or DME/DME radio position updates.

LATERAL GUIDANCE (LNAV) – GENERAL

Lateral guidance (LNAV) is the function in the FMS that sends commands to the flight guidance computer to laterally steer the aircraft. LNAV guides the aircraft along a flight path set by the pilot including flight plans, departures, holding patterns and approaches.
LNAV maintains the aircraft within an airway or protected airspace. LNAV general rules are the following:

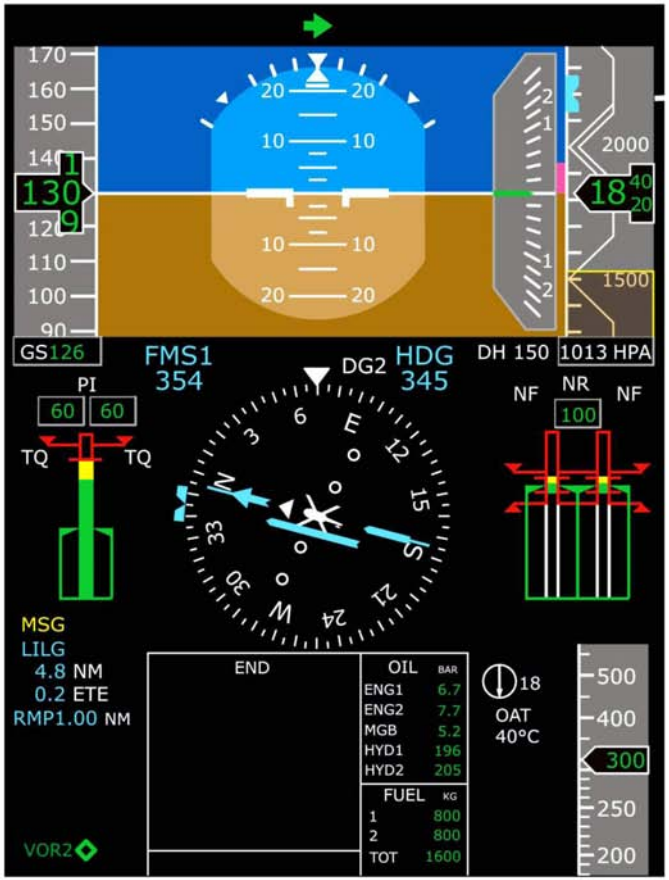
- The FMS must be selected as the navigation source.
- A minimum of one leg must be defined for LNAV calculations.
- LNAV is available for all phases of flight.
- Leg sequencing is performed with inside turns between two legs.
- LNAV bank angles are normally as for rate 1. Roll rate is 3° per second.
- The distance shown for each leg of the flight plan accounts for the distance travelled due to the change in course from one leg to the next.
- A lateral track alert (waypoint identifier flashing) is given for each waypoint sequence. The alert is given 30 seconds before starting a turn.

VERTICAL GUIDANCE (VNAV) – GENERAL

Vertical guidance (VNAV) gives a complete vertical profile for the entire flight and is integrated with the lateral flight plan. Using FMS VNAV, the operator can define vertical profile information that may be manually flown by pilot. VNAV cannot be coupled to the FD.
VNAV altitude profile must be flown manually. VNAV computes TOC (Top Of Climb) and TOD (Top Of Descent) points and gives a vertical deviation for path descents. VNAV general rules are the following:

- For VNAV operation, performance initialization and activation of altitude selector (ALT SEL) is required.
- It is recommended to set the altitude selector at the altitude cleared by ATC

- For climbs, only TOC information is provided on the MFD. The TOC is the location at which the aircraft should complete the climb.
- For descents TOD information is provided on the MFD and VPATH (vertical flight path) on the PFD. On MCDU a BOD (Bottom of descent) information is also presented. The TOD is the location at which the aircraft should commence a descent. The BOD is the location at which the aircraft should complete the descent.
- When the altimeter is adjusted to display height above the ground (QFE) rather than sea level, VNAV should not be used.



PDF Symbology - Full Compass HSI

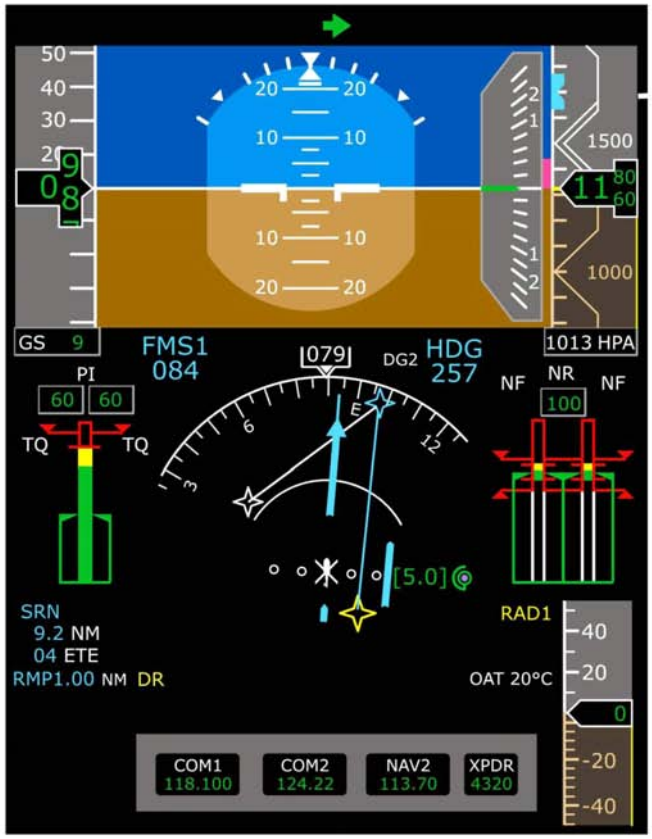
- DESIRED TRACK
- CROSS TRACK DISTANCE BAR
- TO / FROM
- BEARING
- GROUND SPEED
- ACTIVE WPT IDENT
- ETE (Estimate Time Enroute)
- RNP (Required Navigation Performance)
- WIND VECTOR
- ACTUAL TRACK
- Annunciators



DISPLAY CONTROLLER

ICN-39-A-346000-G-A0126-00878-A-01-1

FMS – FULL COMPASS HSI



PDF Symbology - Arc Mode HSI

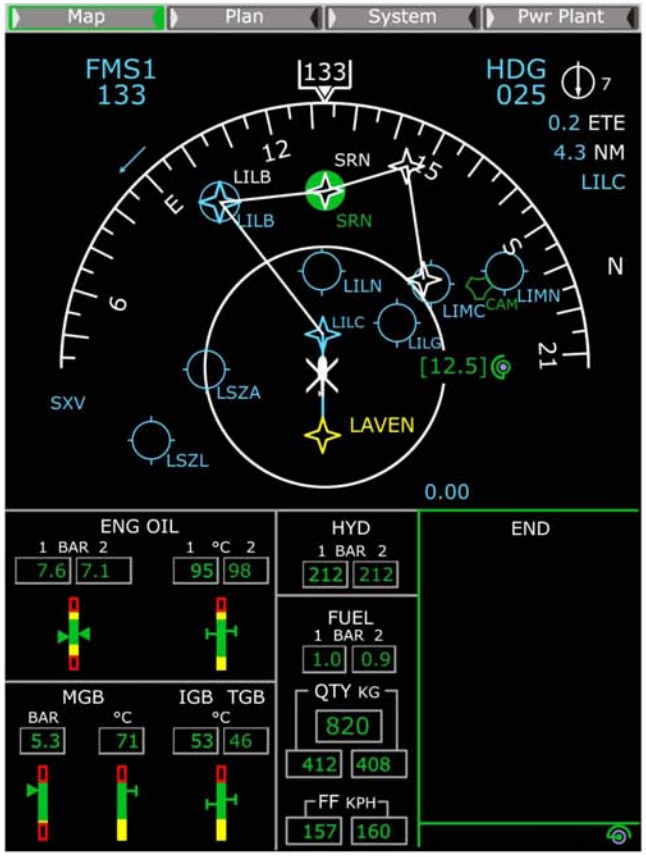
- Same as full compass HSI
- Flightplan



DISPLAY CONTROLLER

ICN-39-A-346000-G-A0126-00879-A-01-1

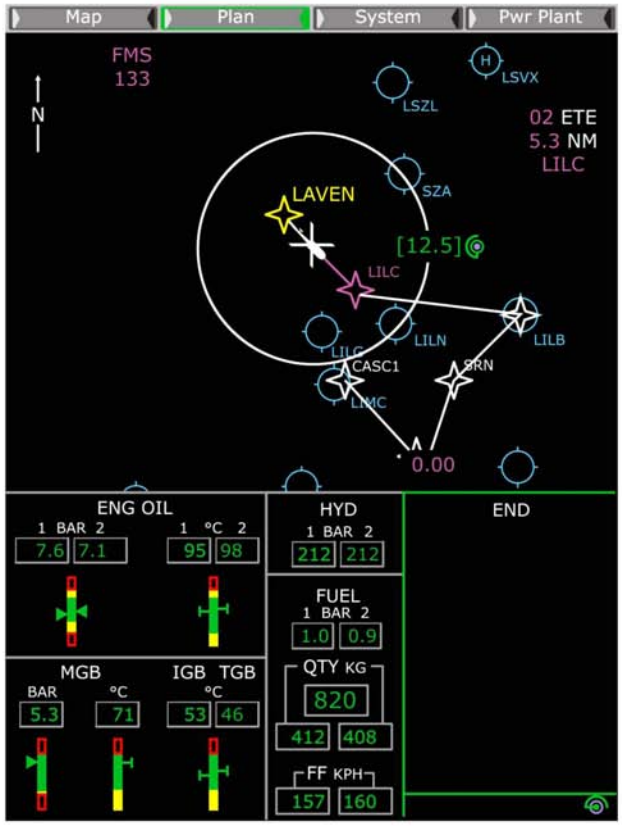
FMS – ARC MODE HSI



MFD Symbology: MAP Format
FPL

ICN-39-A-346000-G-A0126-00880-A-01-1

FMS – MAP FORMAT HSI



MFD Symbology: PLAN Format
FPL

ICN-39-A-346000-G-A0126-00881-A-01-1

FMS – PLAN FORMAT HSI

FMS DISPLAYS AND CONTROL – GENERAL

FMS displays and data are presented on the MFD (MAP page), PFD (HSI) and MCDU. Displayed data includes:

- a map presentation that shows:
  - radio navigation aids
  - airports
  - waypoints on the active flight plan
  - legs, patterns and approach profiles of the active flight plan
- FMS mode annunciations

Electronic maps integrate route map data with auxiliary navigation data to display the aircraft's situation at any time. Electronic displays integrate map data with weather radar displays, terrain maps and electronic charts. Control panels used in FMS operation are :

- MCDU (Multipurpose Control Display Unit)
- RIC (Remote Instrument Controller)
- DC (Display Controller)
- GC (Guidance Controller)

MCDU – GENERAL

The MCDU has a full color display and contains 14 lines each one of 24 characters. The first line is the title line and the fourteenth line is the scratchpad. Color on the MCDU display page is designed to highlight important information but color assignments are not fully coordinated with MFD and PFD displays.

Assigned Color	Parameter
CYAN	Vertical, Performance and Atmospheric Data
GREEN	Lateral, Modes
YELLOW	Warnongs, FROM Waypoint, Flight Plan Names
MAGENTA	TO Waypoint
WHITE	Names, Titles and scratchpad messages
RED	Failures

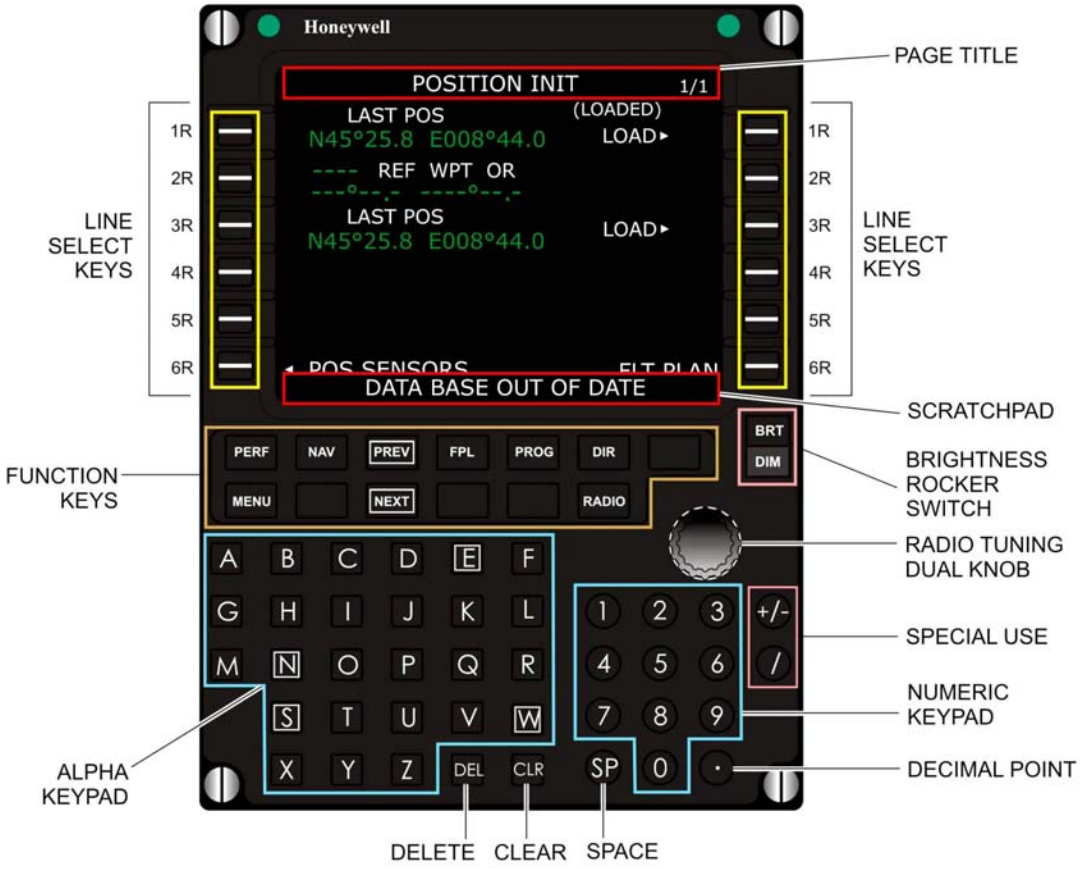
The bottom line on the MCDU display is the scratchpad that is a working area where the pilot can enter data and/or verify data before line selecting the data into its proper position. Alphanumeric entries are made to the scratchpad using the keyboard. As each key is pushed, the character is displayed in the scratchpad. Information in the scratchpad does not affect the FMS until it is moved to another line on the display. Data is retained in the scratchpad throughout all mode and page changes. Functions keys are presented directly below the screen. These keys access primary functions, indices (menus) and page selection.

- PERF Key. Displays page 1 of the performance index. The pilot can select any of the index functions by pushing the respective line select key.
- MENU Key. Displays the MCDU menu page which accesses the maintenance and status information.
- NAV Key. Displays page 1 of the navigation index. The pilot can select any of the index functions by pushing the respective line select key.
- PREV/NEXT Keys. The specific page and number of pages in a particular function or menu display are shown in the upper right corner of the display. The page number format is AA/BB where AA is the current page and BB is the total number of pages available. Page changes are made by pushing the PREV (previous) and NEXT keys. The keys can be held down for repeated page changing.
- FPL Key. Displays the first page of the active flight plan. If no flight plan is entered, the pilot can perform the following: Manually create a flight plan; Select a stored flight plan; Create and store flight plan
- PROG Key. Displays the first progress page. This mode shows the current status of the flight. The first progress page displays distance to, the Estimated Time Enroute (ETE) and fuel projection for the TO waypoint, the NEXT waypoint and destination. It also displays the current NAV mode (i.e. GPS, DME-DME, VOR-DME,AHRS), the required and estimated navigation performance, and the navaids that are presently tuned.
- DIR Key. Displays the active flight plan page with the DIRECT, PATTERN, and INTERCEPT prompts. If other than an active flight plan page is displayed when pushing the button, the first page of the flight plan is displayed. If the active flight plan is already displayed when pushing the button, the display remains on the same page with prompts displayed. DIRECT is the primary function. PATTERN and INTERCEPT must be selected at 6L or 6R, respectively.
- RADIO Key. Pushing the RADIO function key results in the display of the RADIO 1/2 page. From this location and also the RADIO 2/2 page, the pilot can tune various radios including COM1, COM2, NAV 1, NAV 2, ADF and set Transponder code.
- FUNCTION KEY PAGING. Function key paging is an option that, when activated (via FLIGHT CONFIG), can be used to advance function pages using additional pushes of the function key instead of the NEXT key. For example, when initially selecting a function key, the MCDU displays page 1 for that function key. Pushing the same function key again displays page 2 for that function.
- LINE SELECT KEYS (LSK). Line select keys are six on each side of the MCDU displays. Data is selected to a line from the scratchpad or viceversa using the line select keys. These keys are identified from top to bottom as 1L through 6L on the left side and 1R through 6R on the right side. The line select keys are the most often used keys on the MCDU. Typical LSK functions are:
  - Direct Access Prompts/Function Selects In the case of an index display, the line select keys are used to select functions from the index. In displays other than index, the bottom line select keys (6L, 6R) are primarily used for direct access to other functions in the FMS. The FMS prompts the pilot at 6L and 6R for the most likely functions to be selected. An example is the ARRIVAL prompt that is displayed on the active flight plan pages when within 200 NM of the destination. These types of prompts reduce the number of key strokes in order to minimize pilot workload. The pilot can also access functions through the main navigation and performance indices.

- Transfer Line Data to Scratchpad. If the scratchpad is empty, pushing a line select key transfers the respective line data to the scratchpad.
- Transfer Scratchpad Data to Line Fields. Once data has been entered into the scratchpad either through line selection or manual keyboard entry, it can be transferred to any of the allowable line select fields on a page. To transfer the data, push the key adjacent to the line where the scratchpad data is intended.

- ALPHANUMERIC KEYBOARD. Is used by the pilot for input to the FMS. The alphanumeric keys make entries to the scratchpad only. The following are each represented with a key on the MCDU:
  - letters of the alphabet
  - the numbers 0-9
  - the decimal
  - the plus/minus
  - the space
  - the slash

  The SP (space) key is used to insert a space following a character in the scratchpad. The +/- (Plus/Minus) key is used to enter a - or + in the scratchpad. The initial push of the +/- key results in a - being entered. A subsequent push changes the – to a +. Continued pushing of this key toggles the +/- display.
- CLR KEY. This key performs the following functions:
  - When a message is displayed in the scratchpad, pushing the CLR key deletes the message.
  - When a scratchpad entry begins with an asterisk (*) or pound sign (#), pushing the CLR key removes the entire entry.
  - When an alphanumeric entry is made in the scratchpad, one character is cleared from the scratchpad (from right to left) each time the CLR key is pushed. If the CLR key is held down after the first character is cleared, other characters are cleared, one at a time, until the key is released.
- DEL Key. The DEL key is used to delete items from the FMS. When the DEL key is pushed, *DELETE* is displayed in the scratchpad. The DEL key can be line selected to delete waypoints or other items displayed in the MCDU data fields. When there is a message displayed, the delete operation is inhibited. Delete is also used to return default values after entries have been made. The DEL key can also be used in the scratchpad edit mode. With a dash (-) at the end of the scratchpad entry, pushing the DEL key deletes the entire scratchpad entry.
- BRIGHTNESS CONTROL. Both manual and automatic (photo sensor) brightness controls are used to increase or decrease the MCDU display brightness. When manually selected, a bright/dim bar is displayed in the scratchpad. The bright/dim bar level is controlled by pushing BRT or DIM. Following manual adjustment, the photo sensors monitor the ambient light and maintain the brightness level of the MCDU display over various lighting conditions. Note that the brightness can be adjusted during evening hours such that, during daylight hours, the display cannot be seen.



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FMS CONTROLS –MCDU

FLIGHT MANAGEMENT SYSTEM - CAS CAUTION MESSAGES			
CAS CAPTION	**FAILURE DESCRIPTION**	**PROCEDURE NAME**	**AW139-RFM-4D**
1(2) FMS FAIL	Failure of flight management system. FMS navigation not available	FLIGHT MANAGEMENT SYSTEM FAILURE	Section 3 EMERGENCY AND MALFUNCTION PROCEDURES AVIONICS

CHAPTER 45

CENTRAL MAINTENANCE SYSTEM (CMS)

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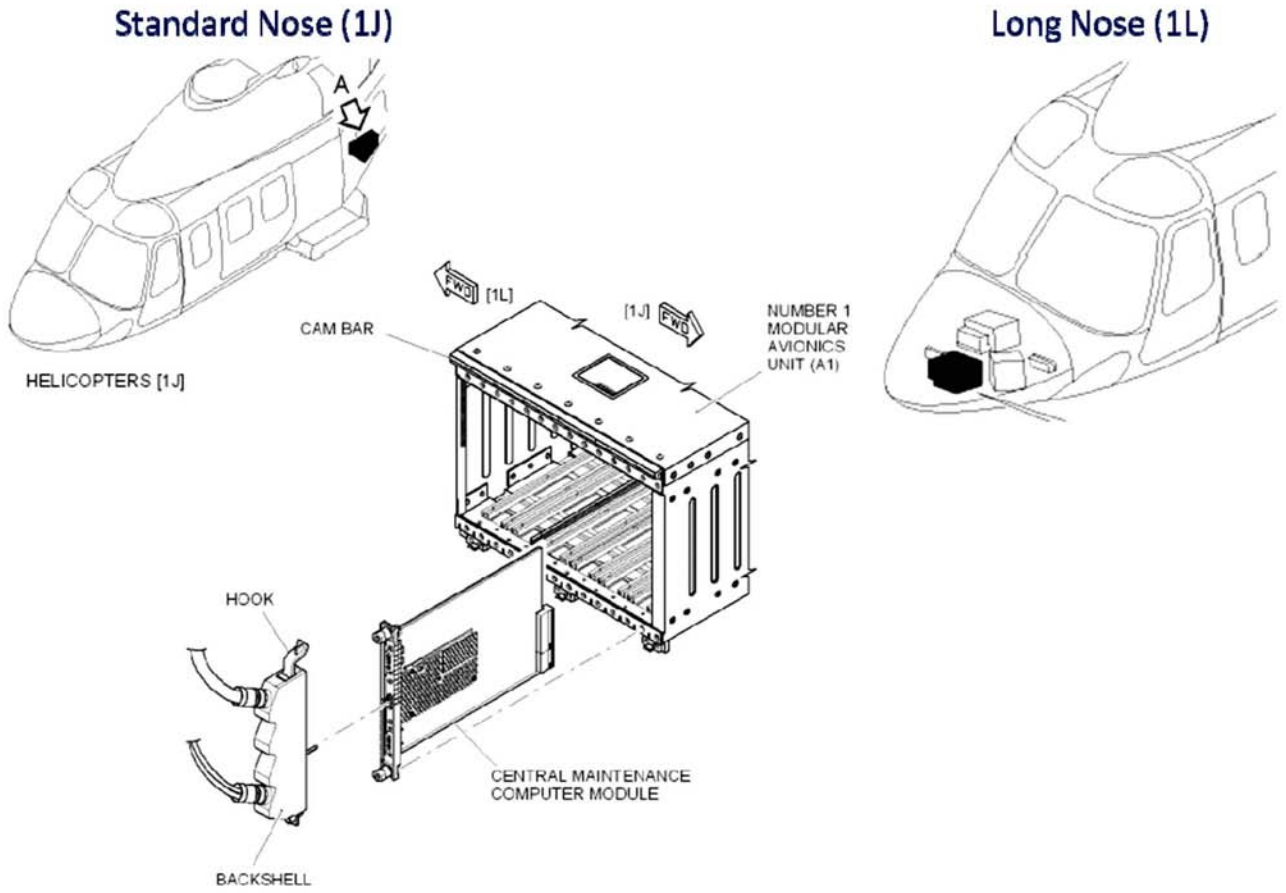
SECTION 45 - CENTRAL MAINTENANCE SYSTEM (CMS)

GENERAL DESCRIPTION

The central maintenance system (CMS) is a combination of fault recording and maintenance access.
The CMS is a modifiable, distributed maintenance system with central fault data collection. The distributive nature of the system lets each member system have its own set of maintenance functions. A centralized maintenance function is used to access each member system that uses standard protocols to combine all systems into a single centrally-focused maintenance system. Maintenance member systems are avionics subsystems that obey the central maintenance requirements and have a connection to the central maintenance computer (CMC).

CMC-RT

The CMC-RT is an off-the-shelf laptop computer configured with Ethernet 10–Base-2 hardware and Honeywell software. Maintenance and data loading is done on the ground with the CMC-RT to upload database files and/or software and download stored (memory) data or fault history files.



ICN-39-A-454500-G-A0126-00593-A-01-1

CMC LOCATION

INTERFACE

ACCESS

The operator may access the CMC software using one of two methods :

- From the cockpit using the Multi Function Display Unit (MFD) for the display, and the installed Cursor Control Device (CCD) for control.
- Access the system from a Remote Terminal (RT) connected to the aircraft LAN Bus.

The Central Maintenance System provides single point access to the maintenance functions of all Member Systems.

CONNECTIONS OVERVIEW

The CMC Module is connected to the MAU through a Backplane Interface Controller (BIC) and through an Ethernet LAN to the Remote Terminal and Network Interface Controller (NIC).
The BIC interface is used by the CMCF software to receive all of its data from the Member Systems.
It is also through the BIC that the CMCF software sends commands to the Member Systems.
The transmit command structure the aircraft is on the ground.
The LAN is used for file transfers within the MAU, and is also used by the Maintenance System to support the Remote Terminal.

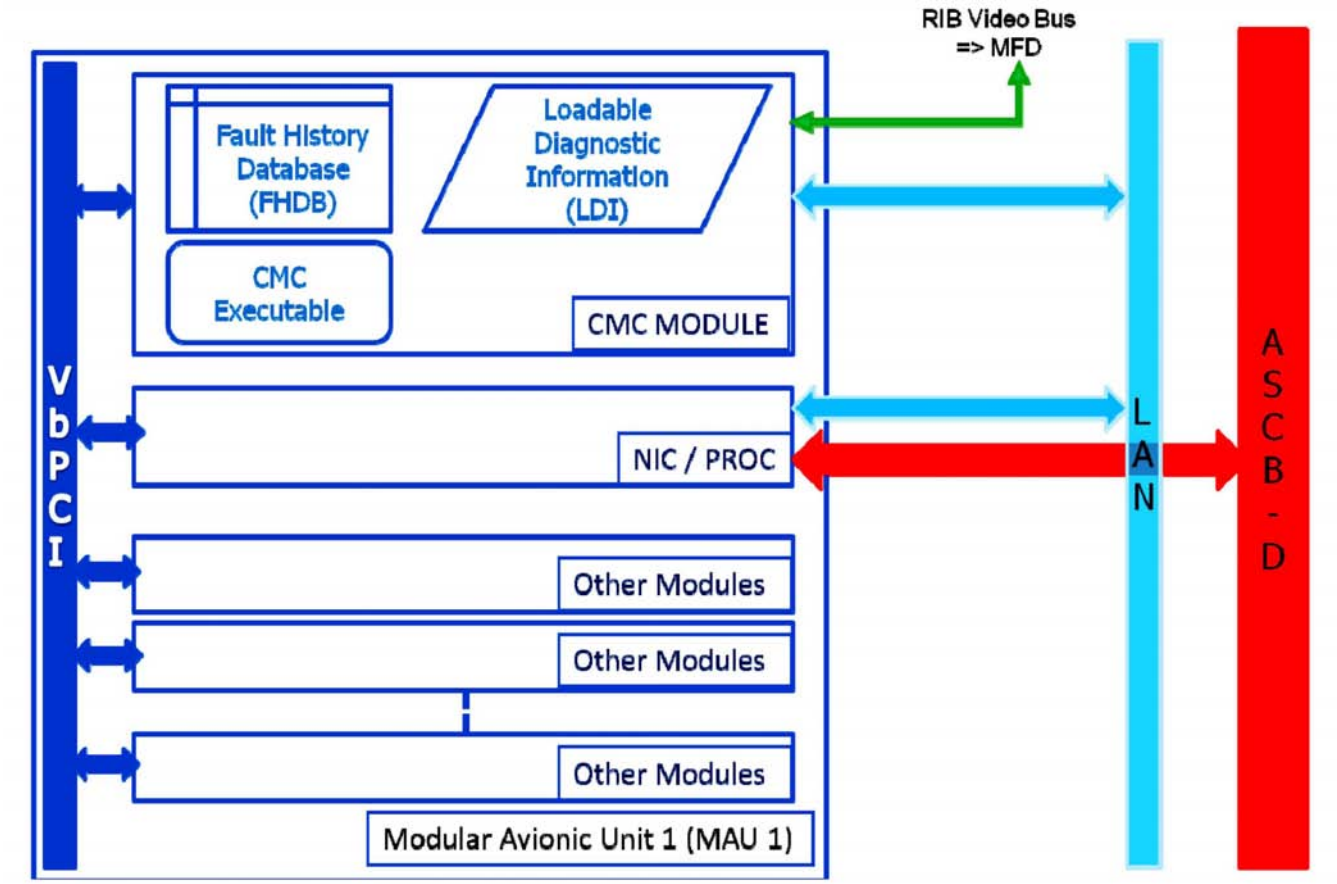
HARDWARE/SOFTWARE INTERFACES

CENTRAL MAINTENANCE COMPUTER MODULE

The Central Maintenance Computer is a single LRM that is in MAU1. The CMC Module derives its normal operating power from the backplane. The CMC Module uses personal computer architecture, including Basic Input/Output System (BIOS) and operating system. The CMC Module has front panel connectors for connecting to Ethernet LAN, and for external connection to battery.

MULTI FUNCTION DISPLAY (MFD)

A MFD serves as the primary display for the Central Maintenance Computer Function. The video outputs from the CMC Module are piped to the display using a Remote Image Bus (RIB) interface. When the maintenance function is selected, the normal graphics generation function for the display is superseded and the CMC Module’s video output is routed directly to the display.
The software that is used to select the desired function to be displayed resides within the Graphics Generator Function (GGF). The GGF software is required to have sufficient interlocks to ensure that the CMC Module video is displayed only when it is safe and appropriate to be displayed.



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CMS INTERFACE

CURSOR CONTROL DEVICE (CCD)

A CCD is used to control the CMS software from the cockpit.

MODULAR AVIONICS UNIT (MAU)

The CMC Module resides in MAU 1 cabinet. The MAU provides the physical housing, the conditioned regulated power, and the interface to the MAU backplane. The backplane provides the interface to all member system signals required by the CMS software.

REMOTE TERMINAL

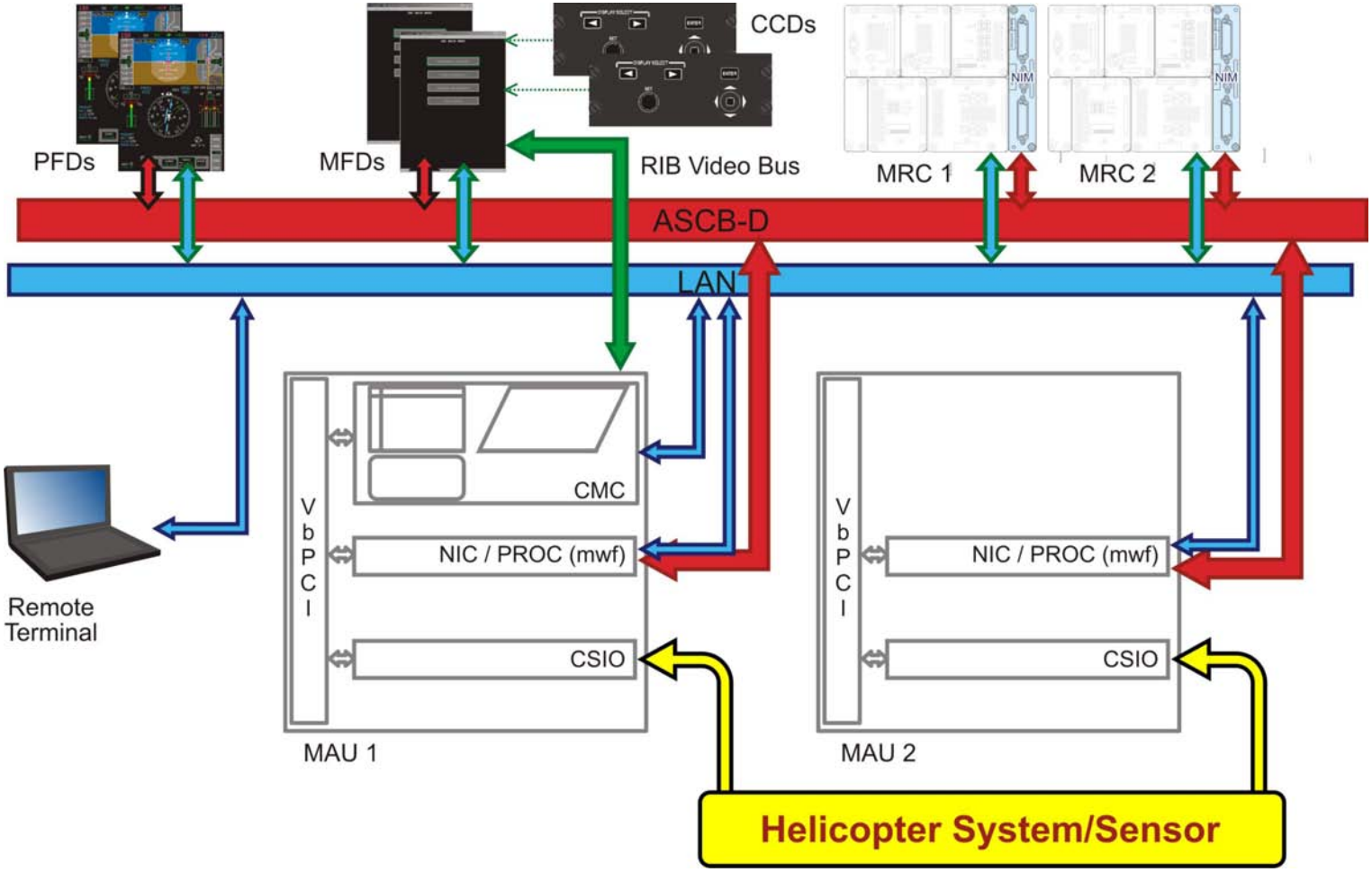
The Remote Terminal (RT) is used to provide access to the CMCF software outside of the cockpit. The RT is hosted on a personal computer that the maintenance personnel can connect to the Ethernet LAN to access the CMS software. One of the requirements for the RT host computer is that it has an Ethernet LAN interface to connect into the CMS software.

COMMUNICATION MANAGEMENT FUNCTION (CMF)

The CMF provides a method for the CMCF software to downlink files to a ground based system from the aircraft.
The CMF takes files from the CMCF software and formats these files into a form that allows the successful transmission of these files from tha aircraft to the ground. The data from the CMCF software can be tought of as a file that is transferred to the CMF. The CMF then determines which available communications link it can use to send the data to the ground. The CMF then determines which available communications link it can use to send the data to the ground. The CMF adds additional information to the file from the CMCF software to ensure that the ground based receiving station can successfully recreate the file and deliver the data to the end user.

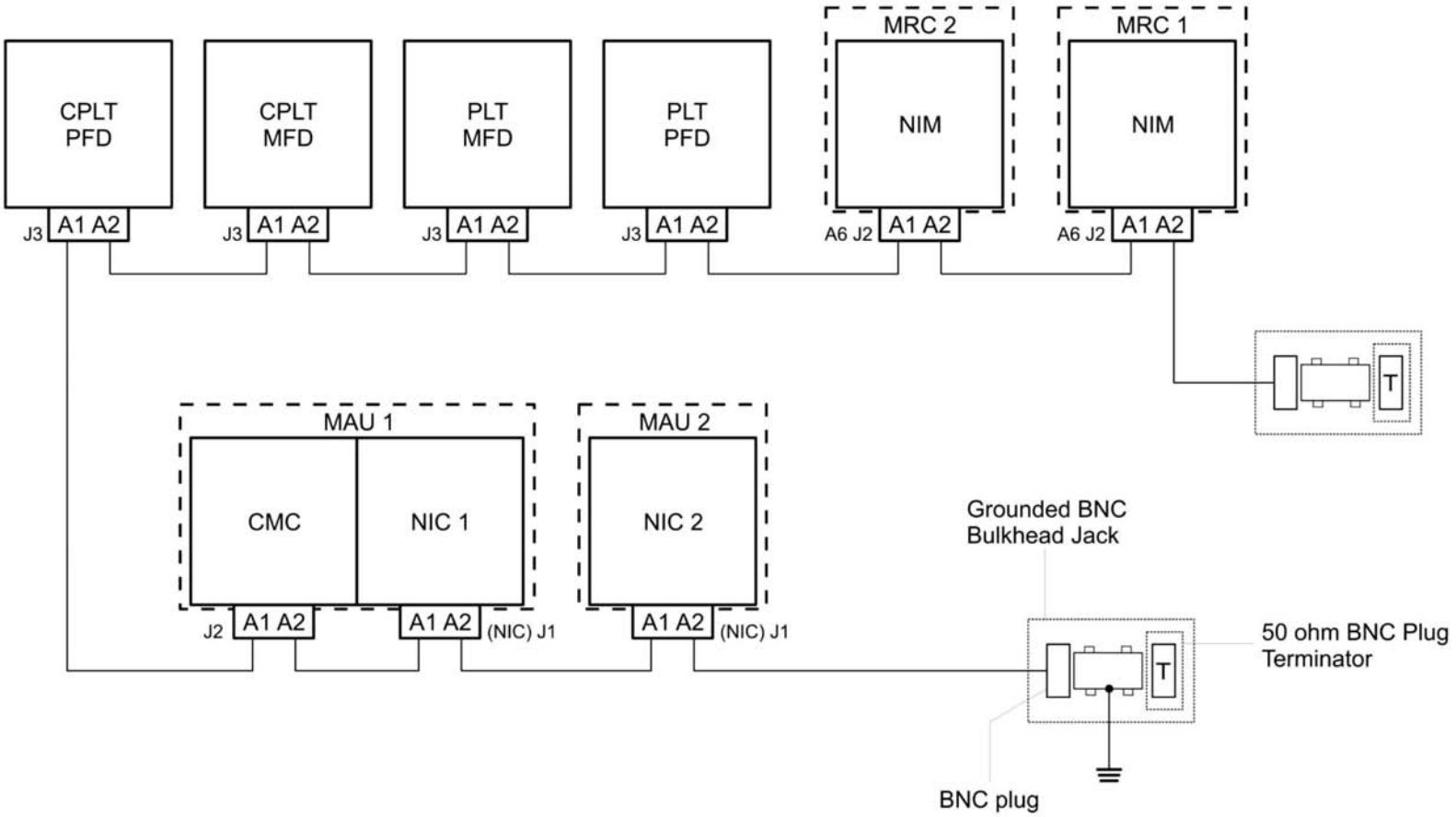
PROCESSOR (S)

The CMC Module uses a 333 MHz Intel Pentium IIMobile Module processor.



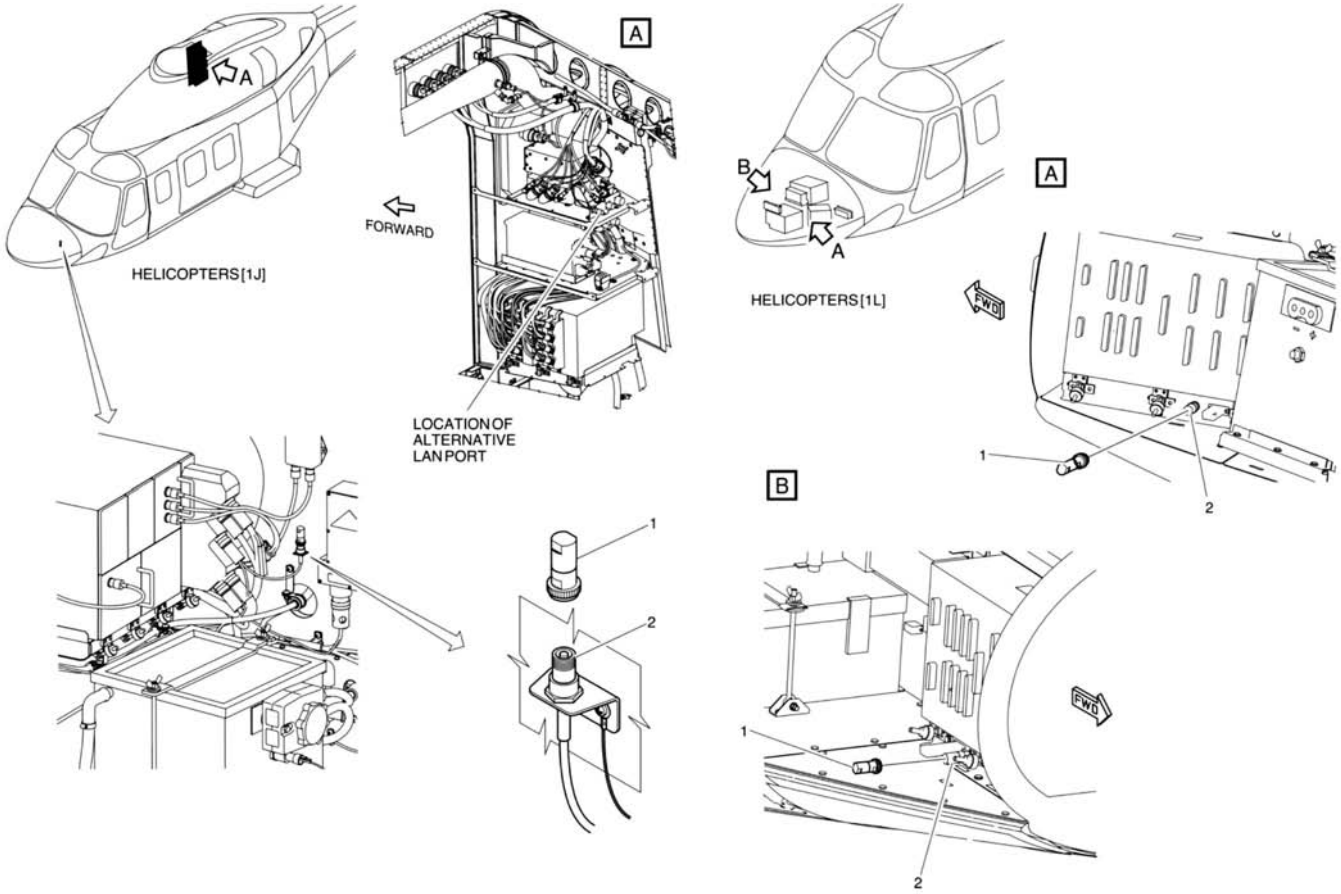
ICN-39-A-454500-G-A0126-00595-A-01-1

CMS GENERAL SCHEMATIC



ICN-39-A-454500-G-A0126-00596-A-01-1

LOCAL AREA NETWORK (LAN) MEMBERS



ICN-39-A-454500-G-A0126-03009-A-01-1

LAN CONECTION [1J] AND [1L] HELICOPTERS

FUNCTIONAL DESCRIPTION

The central maintenance system (CMS) records faults and provides maintenance access to member systems. Member systems are any function, line replaceable unit (LRU) or line replaceable module (LRM) that send results of built-in tests (BIT) to the central maintenance computer (CMC). The CMC performs the following tasks :

- View the aircraft fault history
- Download member system fault data
- Run member system tests
- Show member system configuration data
- Show real-time aircraft data.

The CMC supplies access to member systems from a single user interface. Two methods are available to access the maintenance system.

1 The multifunction display (MFD) to show the CMC menu and the cursor control device (CCD) for control of the MFD.

2 The the central maintenance computer remote terminal (CMC-RT) wich connects to the local area network (LAN) port on the aircraft.

The CMC uses a Microsoft® Access database to keep CMC configuration and member system data. The CMC database is called the loadable diagnostic information (LDI) database. The LDI database is separate from the CMC executable code. Member systems are added or removed from the LDI without changes to the CMS executable code. This also lets a member system supplier update maintenance data for their system without recertifying their flight software.
The CMS works with any member system that follows the member system protocol specifications. The CMC keeps member system fault data in the fault history database (FHDB), and optionally in non-volatile memory (NVM) data storage in the member system. For member systems, the
CMC has modes to:

- View individual fault histories of one member system from data in the FHDB.
- Mix and view fault histories of two or more member systems together from data in the FHDB.
- Download fault data from the member system NVM.
- Move files between removable storage devices and the CMC.
- Engage member systems in tests and show real-time data.

The CMC function does no critical functions on the aircraft and has no impact to safety. The CMC function supplies maintenance personnel with a centralized maintenance system. Maintenance personnel do troubleshooting and fault diagnosis of member systems from a single location.
The CMC does the several noncritical functions on the aircraft that follows:

- Single Point Access to Member Systems
- Maintenance Message Displays
- Ground Test
- Fault History Download
- System Configuration Display
- Member System Operational Status
- FDE to Maintenance Message Correlation
- FHDB
- File Transfer (Dataloading)
- CMC-RT
- LDI.

SINGLE POINT ACCESS TO MEMBER SYSTEMS

The CMC function supplies access to all member systems from a single point. The access can be from the CMC-RT or the MFD. In order for the CMC-RT to access the maintenance system, the CMC must have power and be on line. If the modular avionics unit (MAU) does not have power or the avionics standard communications bus, version D (ASCB-D) is off, the maintenance function is not available. The onground function and initiated test functions are not available while the aircraft is in flight. Member systems must supply the necessary and sufficient interlocks if the CMC function software fails. The interlocks keep the member system from receiving a command to go to an unsafe condition or to do an initiation of tests while the aircraft is in flight

MAINTENANCE MESSAGE DISPLAYS

The CMC gets fault reports from member systems and flight deck effects (FDE) from the monitor warning function (MWF), and keeps this data. FDEs are any unusual condition that the pilot or copilot see on the flight deck. Some examples of a FDE are EICAS messages and autopilot disconnects. The CMC shows “Active Maintenance Messages” and “Stored Maintenance Messages” in sequence by ATA chapter number. Many fault reports can be connected to a maintenance message. The maintenance messages show an internal fault, an external fault, or a probe/sensor fault.

GROUND TEST

The CMC can start ground tests on the member systems.
The CMC supplies commands only while the aircraft is on the ground. Passive status screens and commanded member system tests are onground tests. Access to these functions is locked out by the CMC while the aircraft is airborne.

FAULT HISTORY DOWNLOAD

The CMC supplies the interface to download a fault data from a member system that is kept in NVM.

SYSTEM CONFIGURATION DISPLAY

The CMC shows a list of hardware part numbers, software part numbers (PN), and serial numbers (SN) for the member systems. The system configuration display receives real-time data from the member systems. The system configuration display lets the user see the configuration of components installed on the aircraft. The data does not show that the equipment configuration is safe. It only shows the installed components.

MEMBER SYSTEM OPERATIONAL STATUS

The CMC supplies a member system operational status function that shows the activity status of each applicable member system. For ASCB-D member systems, the CMC monitors the transmissions of the maintenance-data parameter-group. For ARINC 429 member systems, the CMC monitors the transmission of label 377. This function lets the user find out if any given member system is on-line.

FDE TO MAINTENANCE MESSAGE CORRELATION

The FDE to maintenance message correlation supplies a display that connects FDEs to maintenance messages.
This lets the maintenance personnel find components that have a fault on the aircraft.

FHDB (FAULT HISTORY DATA BASE)

The FHDB function collects fault reports and EICAS messages at a one hertz rate. The CMC saves the EICAS messages and faults to the FHDB. The fault reports and EICAS messages show the current status (ACTIVE/INACTIVE), flight leg, flight phase, time, date, aircraft SN, and LRU SN. A fault processing algorithm connects the fault reports to the EICAS messages. This correlation is saved to the FHDB. EICAS messages are supplied by the MWF, and the CMC identifies them as ACTIVE or INACTIVE. The CMC sends changes of state of EICAS messages and fault reports to the FHDB. Fault reports are received from member systems as discrete bits transmitted on ASCB-D or ARINC 429. Each fault report transmitted by the member system is connected to a maintenance message. A maintenance message is a text string that shows a procedure that maintenance personnel can do.

LDI LOADABLE DIAGNOSTIC INFORMATION

The LDI is the database that keeps the maintenance data for all of the member systems. The maintenance data includes maintenance screens and messages. The screen builder data from each member system is put into a single database. The CMF software then gets access to the LDI database, which is a separate file. The verification of the data in the LDI is the responsibility of the member system suppliers. Once data in the LDI is validated, a different tool tracks changes to the data on new versions of the LDI. If any critical data in the LDI changes, the changed data must be revalidated. The LDI database can be changed through the CMC-RT.

FILE TRANSFER (DATALOADING)

The CMC sends files from the CMC-RT to aircraft member systems. An Ethernet port on the CMC-RT connects to the aircraft LAN.

SOFTWARE INSTALLATION PROCEDURE

CAUTION

DURING THIS PROCEDURE RECYCLE THE POWER WILL MEAN:

- TURN POWER TO OFF
- WAIT 4 MINUTES
- TURN POWER TO ON

IF YOU DO NOT OBEY THIS INSTRUCTION, MALFUNCTIONS MAY OCCUR.

TOOLING REQUIRED

The following equipment is required for the software installation:

1 DC External Power Bench (28VDC);

2 Computer with
- Windows NT//95/98/2000/Me/XP;
- 256 MB RAM;
- CD-ROM Driver;

3 LAN Network “Dongle Adapter” P/N LTEPIC-25 (Refer to IPD): PCMCIA slot to Coax LAN cable

4 WOW SIMULATOR kit P/N 3G3205G00731 (Refer to IPD).

5 PRIMUS EPIC® Software installation CD’s dedicated to the helicopter.

COMPUTER SETTING

1 Power on the computer.

2 From the START button select “Settings (Impostazioni)” → “Control Panel (Pannello di Controllo)” → “Network (Rete e connessioni remote)”.

3 Right click on the “Local Area Connection (Connessione alla rete locale)” being used to connect to the aircraft, then select “Properties (Proprietà)”.

4 Scroll down the window and highlight “Internet Protocol (TCP/IP) (Protocollo Internet (TCP/IP))”, select Properties button.

5 A window similar to Figure 3-2 will appear. Select the “Use the following IP address” button and enter the IP address 192.168.200.201 and the Subnet mask 255.255.0.0 in the window.
When a new PRIMUS EPIC® Software Release is issued, the relevant TOOL software must be installed on the Computer (Tools CD) before proceeding with the S/W installation on the helicopter. See the following instructions:

1 Install the Remote Terminal program supplied on the Tools CD (disk 1 of REQUIRED MATERIALS) of the current software version to be installed. Use the default settings and commands as you install the Remote Terminal software (Example: the installation exe file to be run to install the Remote Terminal will be a file similar to “Install_RT_V13.1_PS7031531_010.EXE”).

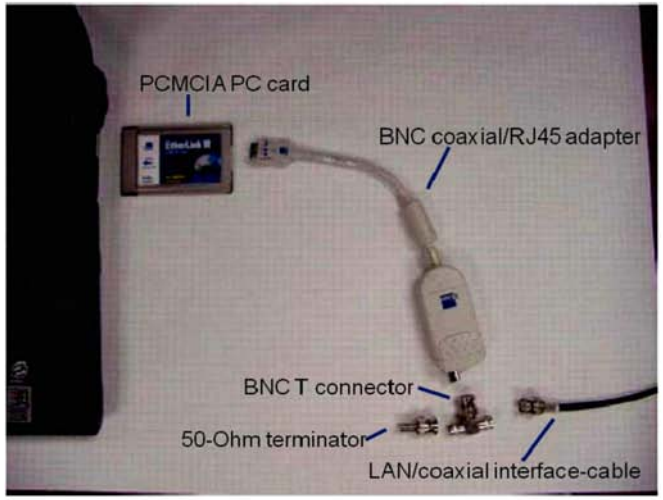
2 At the end of the installation procedure the icon of the application “CMC_RT.exe” will be generated on installation folder. Create a desktop link directed to C:\CMC\CMC_RT.exe.

WARNING

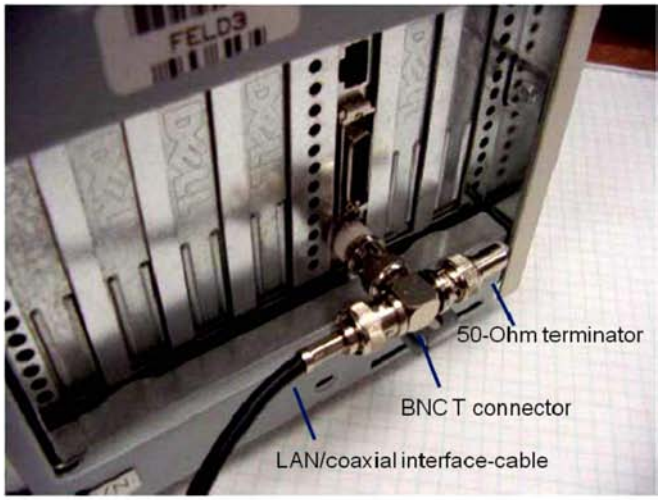
THE COMPUTER USED FOR THE SOFTWARE UPLOADING ON THE HELICOPTER SHOULD NOT BE USED FOR OTHER TASKS AND AN EFFICIENT ANTI-VIRUS SOFTWARE SHALL BE INSTALLED AND KEPT UPDATED.

The following equipment are required for the computer preparation:

(1) Computer with Windows NT//95/98/2000/Me/XP; 256 MB RAM; CD-ROM Driver;
(2) PRIMUS EPIC LAN interface kit
(3) EPIC tools for Agusta AW139



PRIMUS EPIC interface kit for Laptop computer

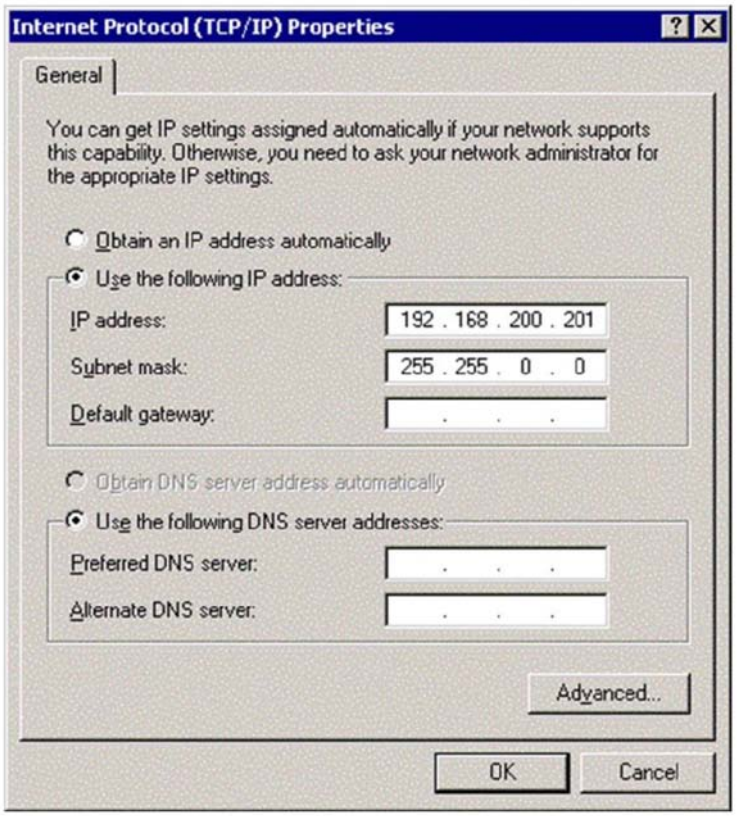


PRIMUS EPIC interface kit for Desktop computer

WARNING: The computer used for the software uploading on the helicopter should not be used for other tasks and an efficient anti-virus software shall be installed and kept updated.

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COMPUTER PREPARATION

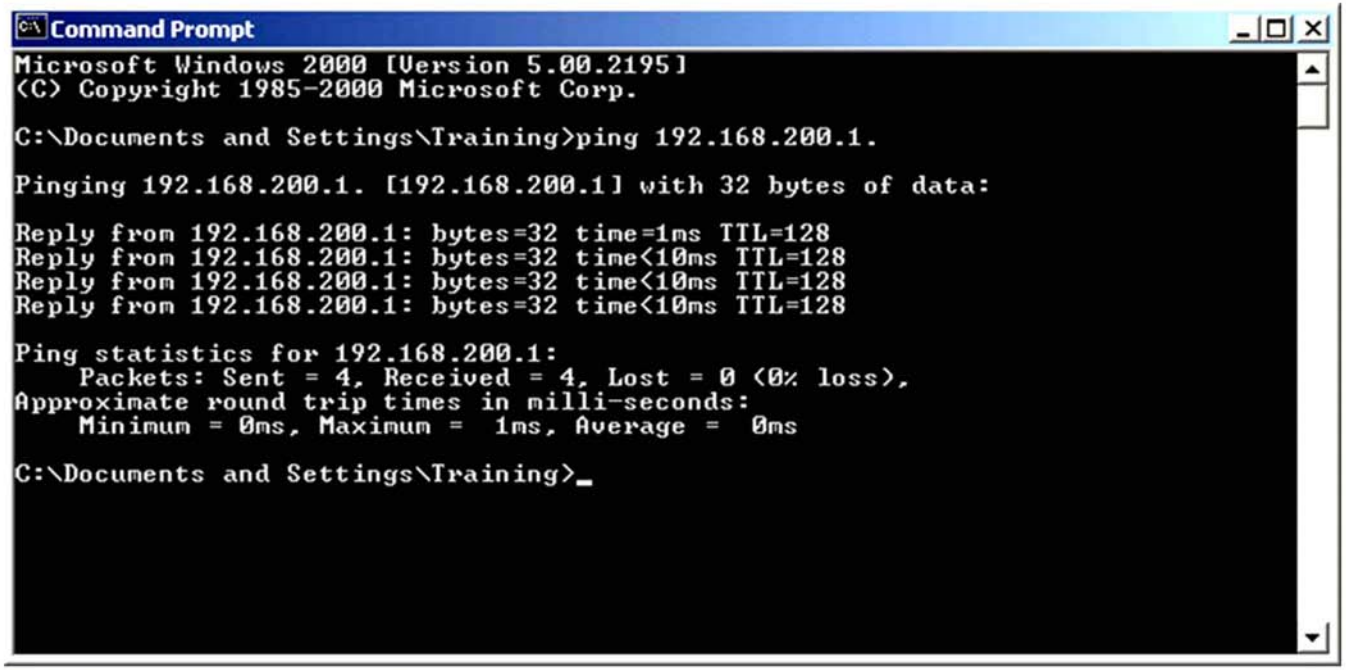


you can give the value of 196 to 220 as necessary

ICN-39-A-454500-G-A0126-00599-A-01-1

TCP/IP SET-UP PROTOCOL

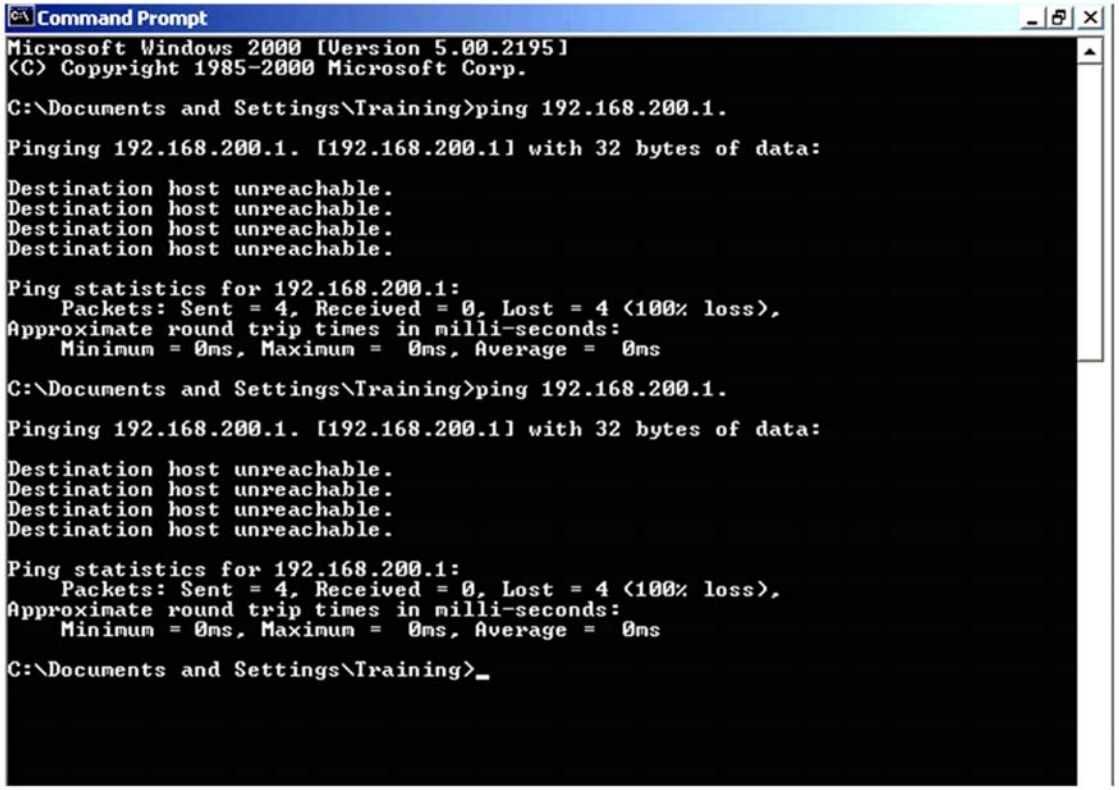
Test of the connection



LAN works properly

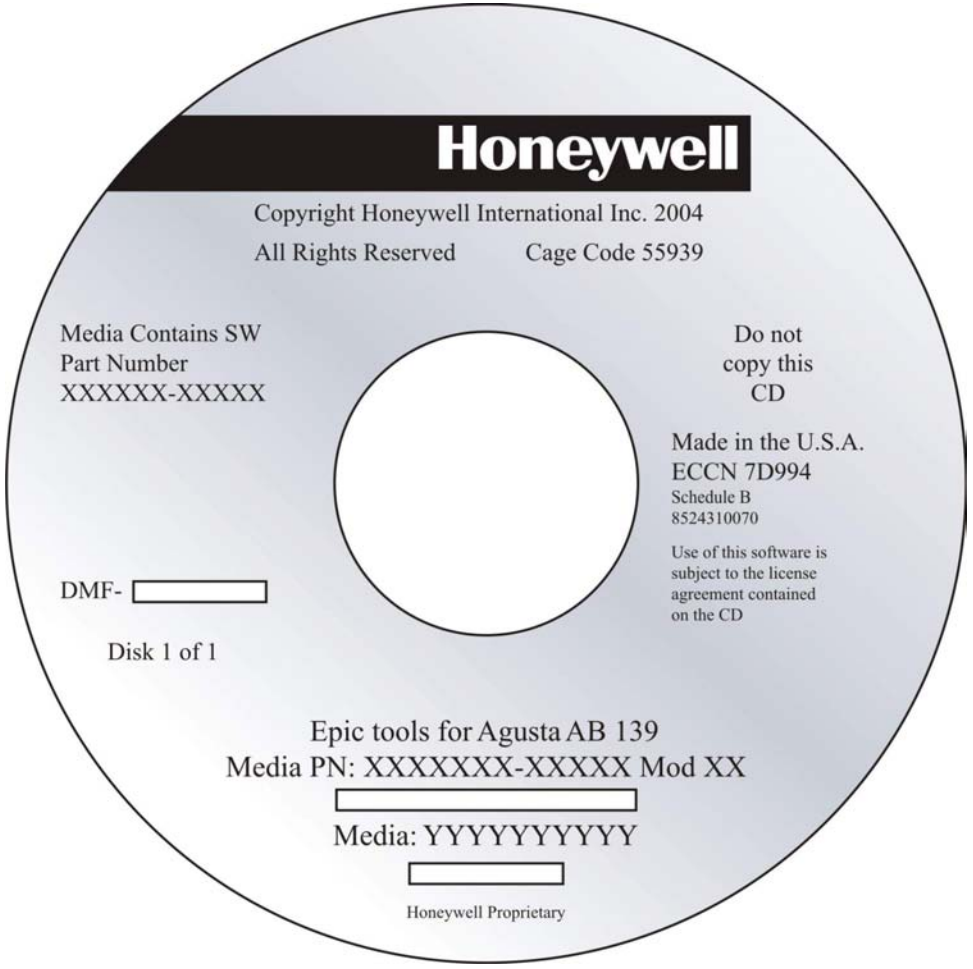
ICN-39-A-454500-G-A0126-00600-A-01-1

DOS COMMAND WINDOW (SHEET 1 OF 2)



ICN-39-A-454500-G-A0126-00601-A-01-1

DOS COMMAND WINDOW (SHEET 2 OF 2)



ICN-39-A-454500-G-A0126-00602-A-01-1

PRIMUS EPIC TOOLS FOR AGUSTA AB139 CD-ROM



CMC REMOTE TERMINAL TOOL.lnk



APM SETTINGS TOOL.lnk



APM RESTORATION TOOL.lnk

ICN-39-A-454500-G-A0126-00603-A-01-1

PRIMUS EPIC TOOLS FOR AGUSTA AB139 PROGRAM

DESCRIPTION AND OPERATION OF MAJOR COMPONENTS

CMC MODULE

The CMC module is the main LRM of the CMS. The CMC module is installed in the MAU 1. The MAU backplane supplies power to the CMC module. The CMC module uses the software and hardware that follow:

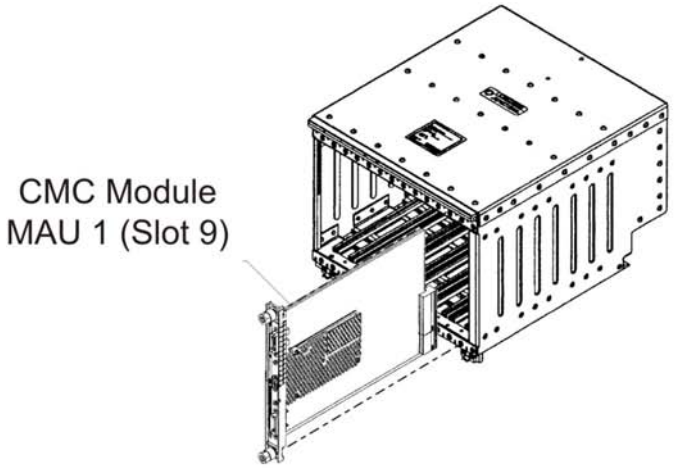
- 333 mHz (minimum speed) Pentium CPU
- 128 megabytes (minimum) of random access memory (RAM)
- 144 megabyte solid-state hard drive
- Windows® NT operating system.

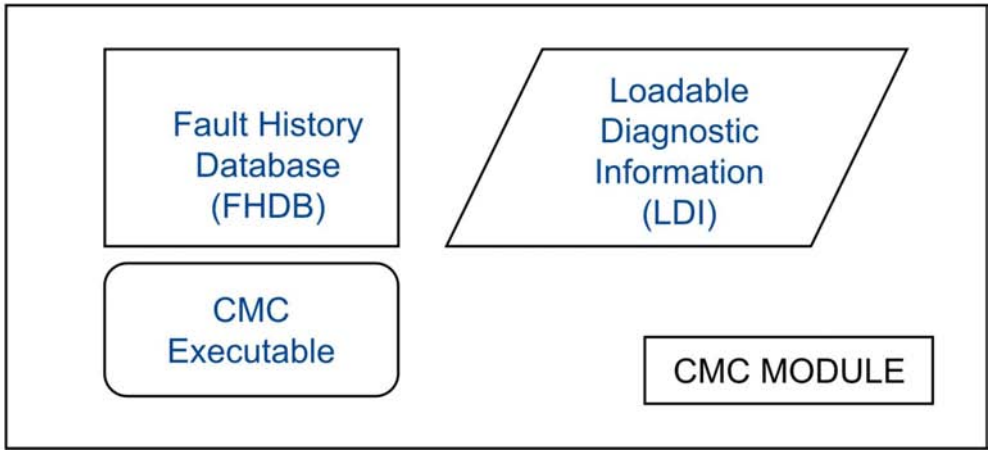
The solid-state hard drive contains the operating system, program files, FHDB, and the LDI. The storage space on the hard drive is divided as follows:

- Operating System — 32 Mbyte.
- Central Maintenance Computer Function (CMCF) Software — 10 Mbyte
- FHDB — 10 Mbyte
- LDI—20Mbyte.

The CMC module supplies a central display of all member systems maintenance data to:

- Give data to the display unit (DU) or CMC-RT
- Do initiated built-in tests (IBIT)
- Download member system storage of fault data
- Give a report of member system SNs and/or PNs
- Monitor continuous BIT
- Monitor power-up BIT
- Monitor the performance parameters of rigging tests.





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CMC INTERNAL FUNCTIONS

NORMAL OPERATION

The CMC has power and is functional during all phases of flight. The CMC function monitors the ASCB-D bus for fault reports that are transmitted in real time from the member systems. Member systems assemble, report to the function, and keep fault data in NVM when equipment failures occur. This is done by the member system continuous BIT and power-up BIT. The data kept in the CMC function is used during maintenance mode. The data kept in the member system NVM is used by the member system supplier for repair of their failed unit. It is necessary to download the member system NVM to the CMC-RT to get access to the data. During normal operation the member system requirements can be defined by the tasks that follow:

- Transmission of the member system faults
- Transmission of the equipment ID, SDI, hardware PN, software PN, and SN data
- Performance of power-up and continuous BIT to find faults.

MAINTENANCE MODE

Maintenance mode is available when the aircraft is on the ground and the proper safety conditions are met. The CMC sends a CMC active command to the member systems to show the maintenance mode of operation. If the aircraft is not on the ground, the maintenance mode functions and access to maintenance screens are locked out by both the CMCF and member systems. If the CMC fails, the member system can not be commanded to any unsafe condition or start tests when not in maintenance mode. The maintenance crew uses this mode to diagnose and repair the aircraft. When the CMCF is in the maintenance mode, the CMC supplies the member system tasks that follow:
During maintenance mode, the CMC supplies the member system tasks that follow:

- Supplies access for member systems to show current fault data
- Supplies systems diagnostics with IBIT and real-time display of member system status
- Downloads fault data that is kept in the member system NVM.
- Transmits the member system current faults, unless the member system has engaged a commanded test or NVM download.
- Transmits the equipment ID, SDI, hardware PN, software PN, and SN data, unless the member system has engaged a commanded test or NVM download.
- Does the power-up function and continuous BIT test to find faults, unless the member system has engaged a commanded test or NVM download.
- When set by the CMC, does the commanded test when necessary and sufficient interlocks are available.
- When set by the CMC, does the NVM download when necessary and sufficient interlocks are available.

SYSTEM CONFIGURATION

The user can find the maintenance configuration of member components installed on the aircraft. The data does not identify the equipment configuration as safe. It is only used to view the list of installed components. The CMC shows the member system data that follows:

- Equipment IDs
- Sources of Destination IDs
- Hardware PNs
- Software PNs
- SNs
- Up to 94 Additional Items.

The system configuration display is supplied by real-time data transmitted from the member system to the CMC.

FILE AND DATA LOADER SYSTEM (DLS)

The data loading system (DLS) function is done through the CMC-RT. This function also supplies support for the FMS database and member software loads. The CMC does file transfers from the CMC-RT to the aircraft ASCB-D member systems. The file transfer process uses COTS software and the Ethernet LAN. For more data about the DLS, refer to Section 2-46 Data Loading System (DLS).

MONITORING AND DIAGNOSTIC

DISPLAY PAGES AND CONTROLS

The maintenance display pages can be selected using the CMC – RT and its cursor pointer device or from the cockpit by using the CCD.
Active menus are highlighted using a green border,selection is made using the < ENTER> key.
Further resolution of in-flight fault data must be done on the ground where full CMC functionality can be put into operation. Fault data from any member system LRU or the CMC can be downloaded to the CMC-RT for later analysis.
Figure 2.43-6 shows a block diagram of the CMC module.

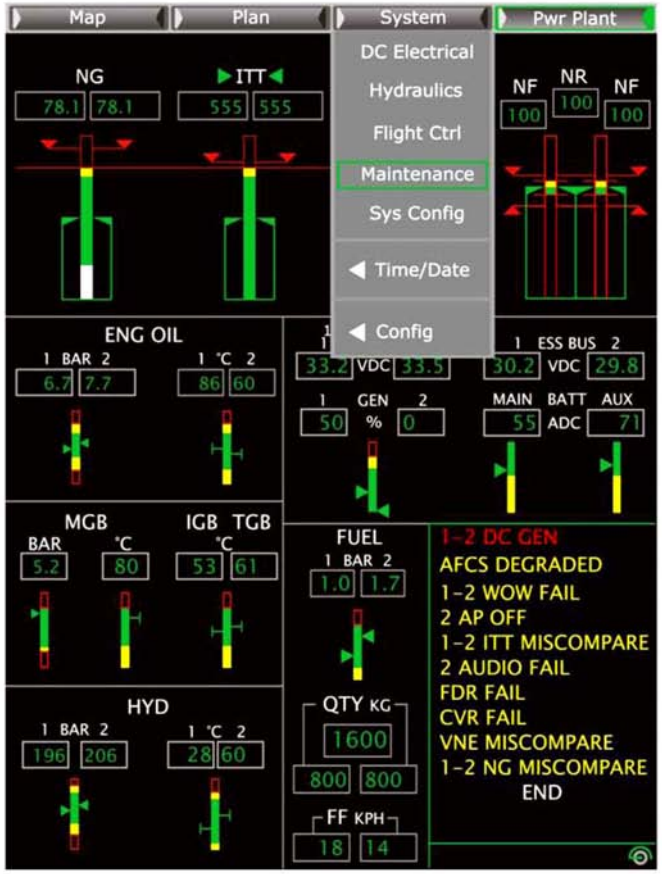
MENU TREE

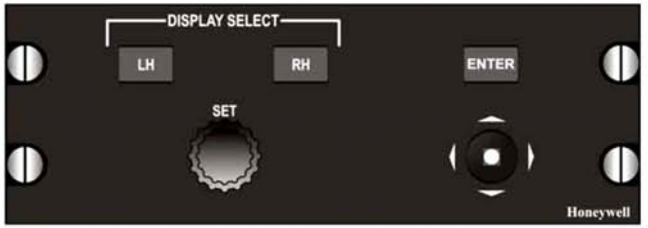
The CMC management is accomplished through a tree of menu pages.
Navigation through the menu is as follows:

- Menus
- CursorBox
- Enter Function
- Bezel Buttons.

MFD SYSTEM MENU

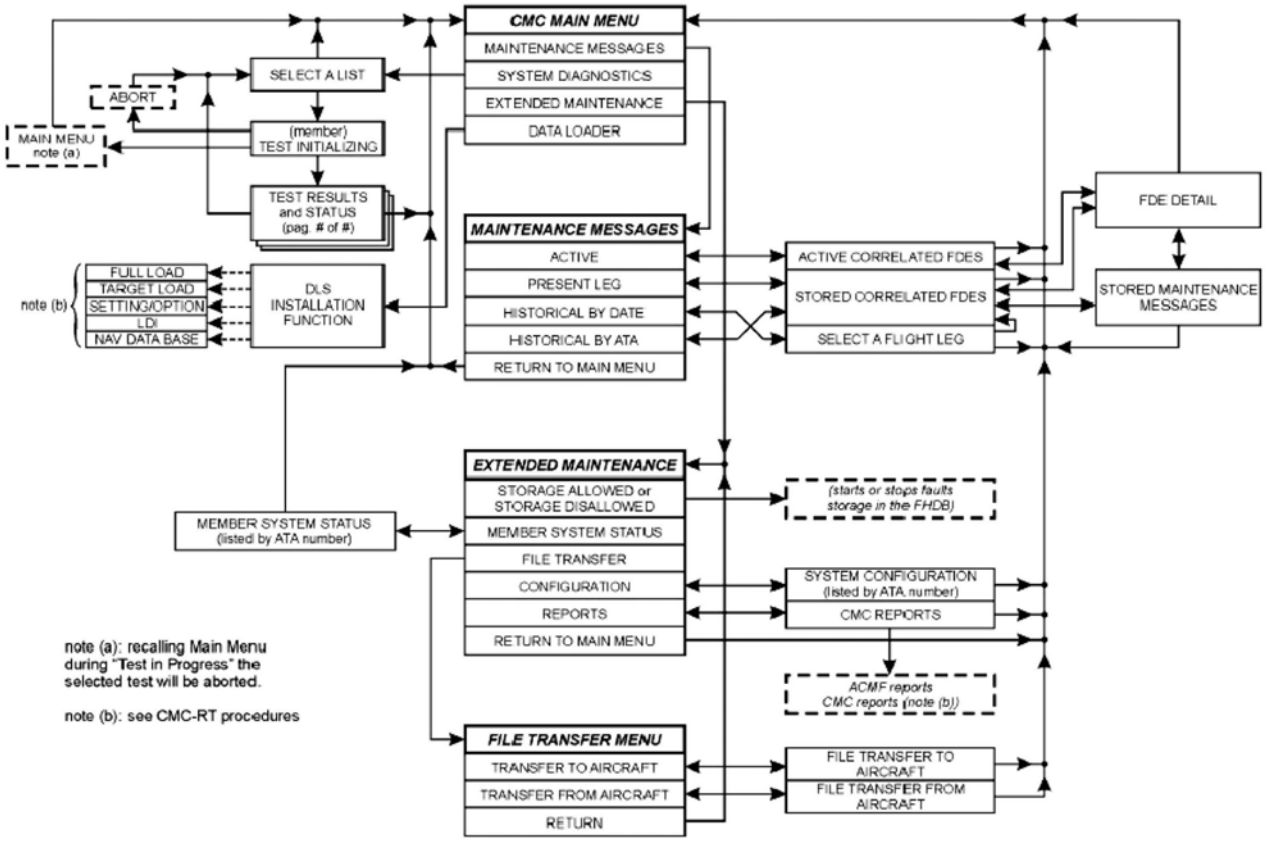
The System button on the menu bar sets and controls the display of the various aircraft system pages on the MFD. When the System menu button is highlighted, selection of the ENTER button on the CCD shows the most recent system page.
Movement of the joystick down while the System menu button is highlighted shows the System dropdown menu. Selection of Maintenance shows the output of the Central Maintenance Computer (CMC) on the MFD in a full screen format.
Selection of Maintenance is only available when the CMC is valid and the aircraft is on the ground. If this menu item is not available for selection, it shows in gray.





ICN-39-A-454500-G-A0126-03010-A-01-1

MAINTENANCE PAGE SELECTION



ICN-39-A-454500-G-A0126-00606-A-01-1

CMC OPERATIVITY DIAGRAM

CMC MAIN MENU

The CMC Main Menu selections let the operator to select and look maintenance messages, do subsystem tests on member systems or look at extended maintenance services.
The menu selections are as follows:

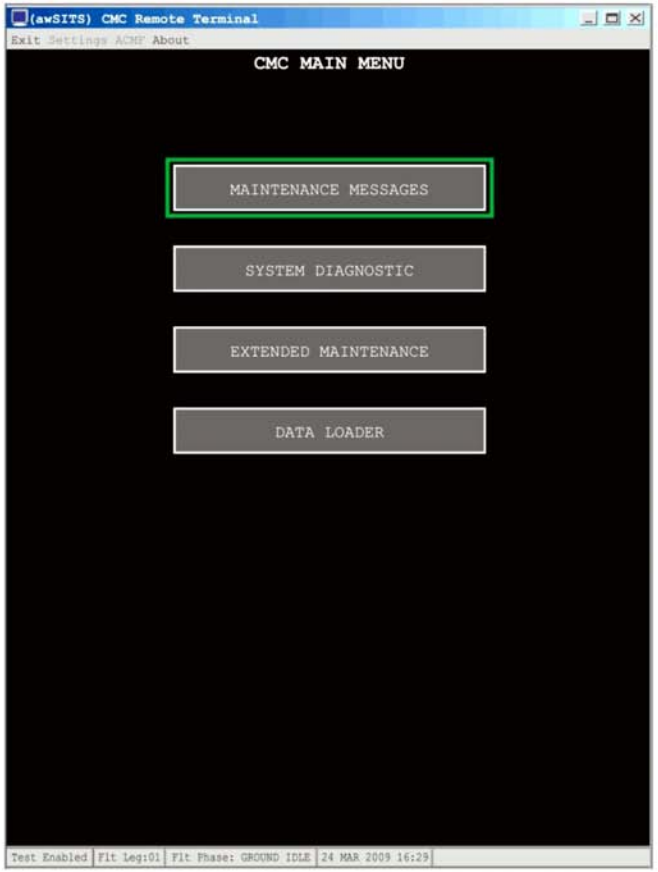
- MAINTENANCE MESSAGES - goes to the maintenance message menu
- SYSTEMS DIAGNOSTICS - goes to the test selection list
- EXTENDED MAINTENANCE - goes to the extended maintenance menu
- DATA LOADER - goes to the data loader menu.



CMC REMOTE TERMINAL TOOL.lnk



COURSE CONTROL DEVICE



ICN-39-A-454500-G-A0126-03012-A-01-1

CMC MAIN MENU PAGE – MAINTENANCE MESSAGES SELECTION

MAINTENANCE MESSAGES MENU

The MAINTENANCE MESSAGES menu shows a selection of current or historical message data.
The menu selections are as follows:

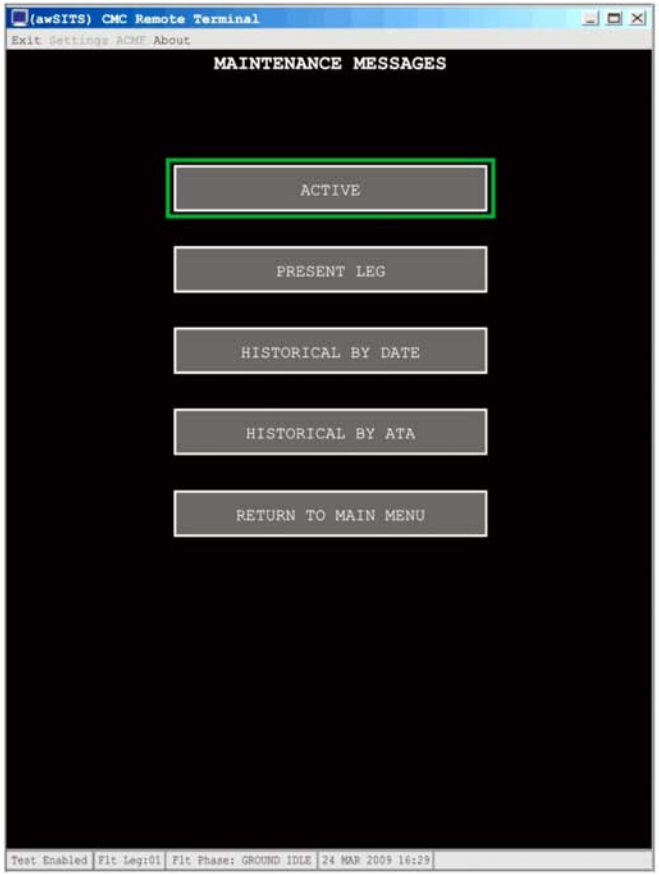
- ACTIVE - goes to the active correlated FDES list
- PRESENT LEG - goes to the stored correlated FDES list
- HISTORICAL BY DATE - goes to the stored correlated FDES list in date order
- HISTORICAL BY ATA - goes to the stored correlated FDES list in ATA order
- RETURN TO MAIN MENU - goes back to the main menu.



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ICN-39-A-454500-G-A0126-03013-A-01-1

MAINTENANCE MESSAGES PAGE

ACTIVE CORRELATED FDES LIST

The ACTIVE CORRELATED FDES shows the current CAS messages and related maintenance messages. When the message is expanded, the message details show.
This display is in sequence by current CAS message groups (warning, caution, advisory, and status). A group is only given in the list if there are messages in that category.
Each group shown can be expanded to show more details related to any current maintenance message.
The bezel buttons are available as follows:

- MAIN MENU - goes to the main menu
- PREV - goes to the previous page
- MAINT MSG - goes to the maintenance message page.



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ICN-39-A-454500-G-A0126-03014-A-01-1

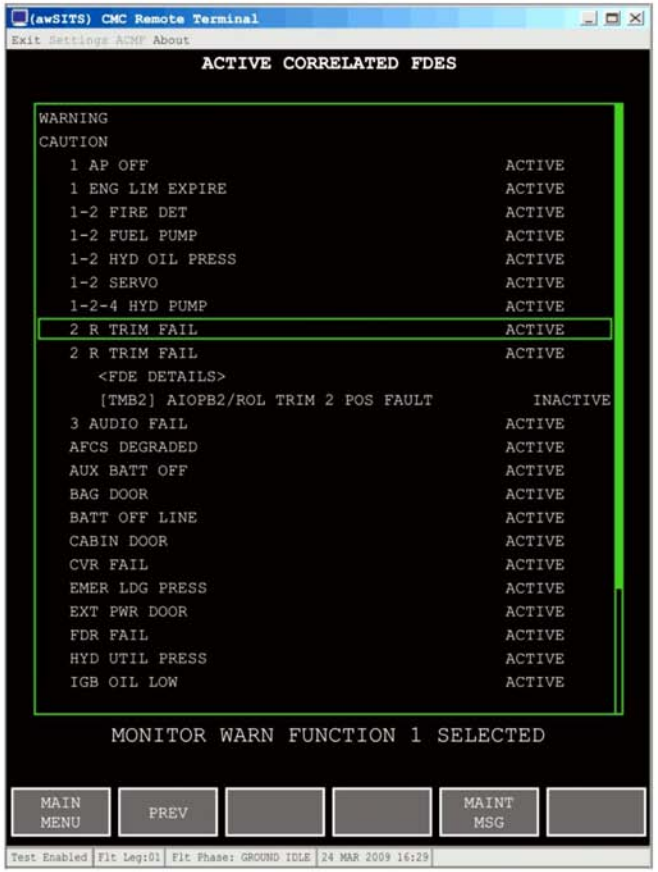
ACTIVE CORRELATED FDES PAGE (SHEET 1 OF 2)



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ICN-39-A-454500-G-A0126-03015-A-01-1

ACTIVE CORRELATED FDES PAGE (SHEET 2 OF 2)

FDE DETAIL PAGE

The FDE DETAIL page shows data about the specific message selection made.
Selection of the PRESENT LEG on the MAINTENANCE MESSAGES menu goes to the stored correlated FDES menu. It contains three fields for added data that are POSSIBLE LRU(S) AT FAULT, SYMPTOMS, and DOCUMENTS.
The bezel buttons are available as follows:

- MAIN MENU - goes to the main menu
- PREV - goes to the previous page.

The FDE DETAIL page has five sections:

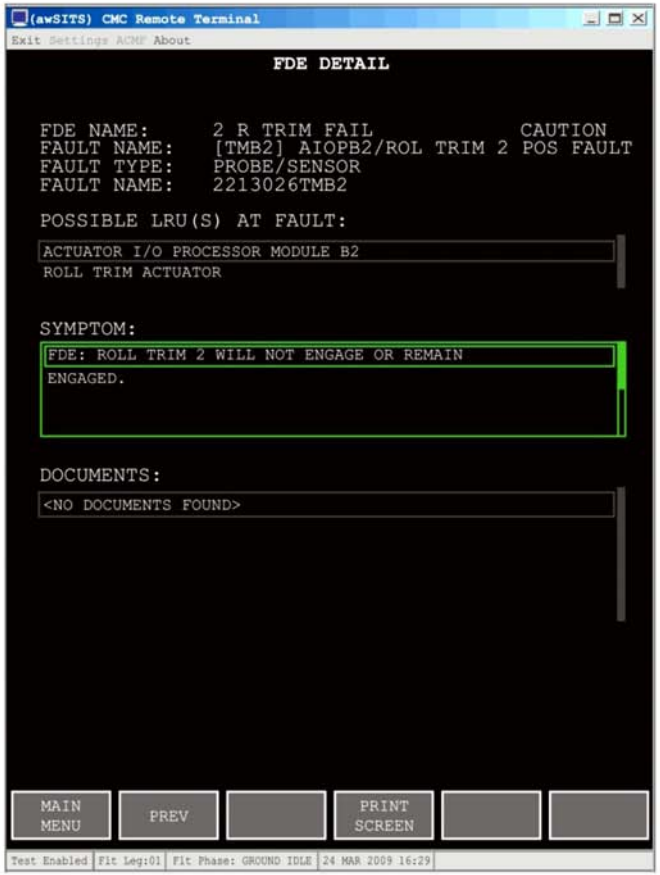
- HEADER - Contains the FDE NAME or maintenance message and its group, which is warning, caution, or advisory. The FAULT NAME, FAULT TYPE and FAULT CODE show, if known.
- POSSIBLE LRU(S) AT FAULT - Shows the possibile LRU that caused the maintenance message. If more than one LRU shows, the more probable LRU is at the top of the list.
- SYMPTOMS - Shows text recorded by the member system supplier that gives the symptoms caused by the identified failure or FDE.
- DOCUMENTS - Shows a list of the documents that the shown FDE or maintenance message is related to. This helps maintenance personnel find more data about the fault or problem.
- MAINTENANCE MESSAGE CORRELATION - Shows the time, date, and flight phase of the changes of the maintenance message.



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COURSE CONTROL DEVICE



ICN-39-A-454500-G-A0126-03016-A-01-1

FDE DETAIL PAGE

HISTORICAL DATA BY DATE LIST

Selection of the HISTORICAL DATA BY DATE button from the main menu shows the SELECT A FLIGHT LEG page.
The SELECT A FLIGHT LEG page lets the user make a selection of a date and flight leg to review. Data is included in the historical message display related to this selection.
The bezel buttons are available as follows:

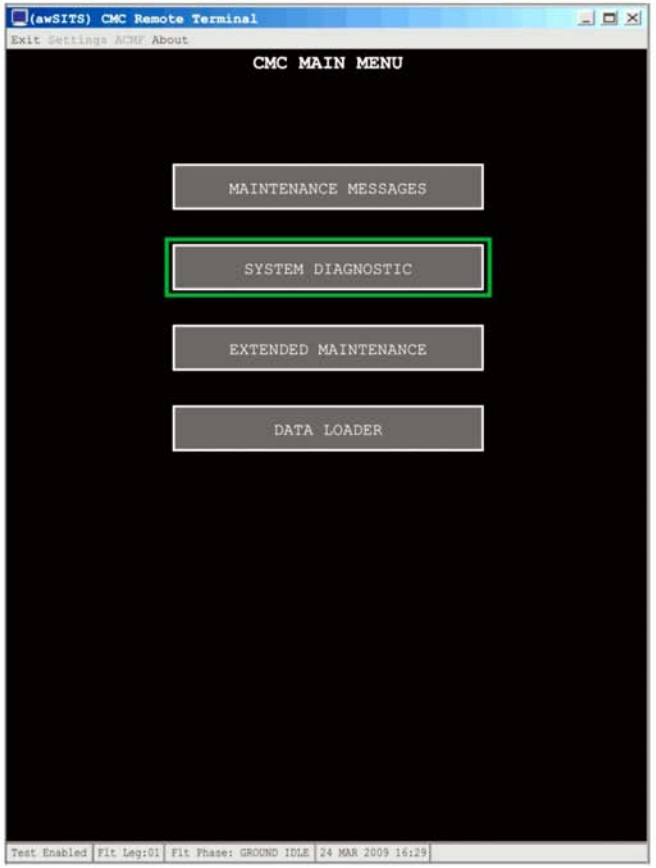
- MAIN MENU - goes to the main menu
- PREV - goes to the previous page.



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COURSE CONTROL DEVICE



ICN-39-A-454500-G-A0126-03017-A-01-1

CMC MAIN MENU PAGE – SYSTEM DIAGNOSTIC SELECTION

STORED MAINTENANCE MESSAGES

Selection of the MAINT MSG button from the STORED CORRELATED FDES page shows the STORED MAINTENANCE MESSAGES page.
The bezel buttons are available as follows:

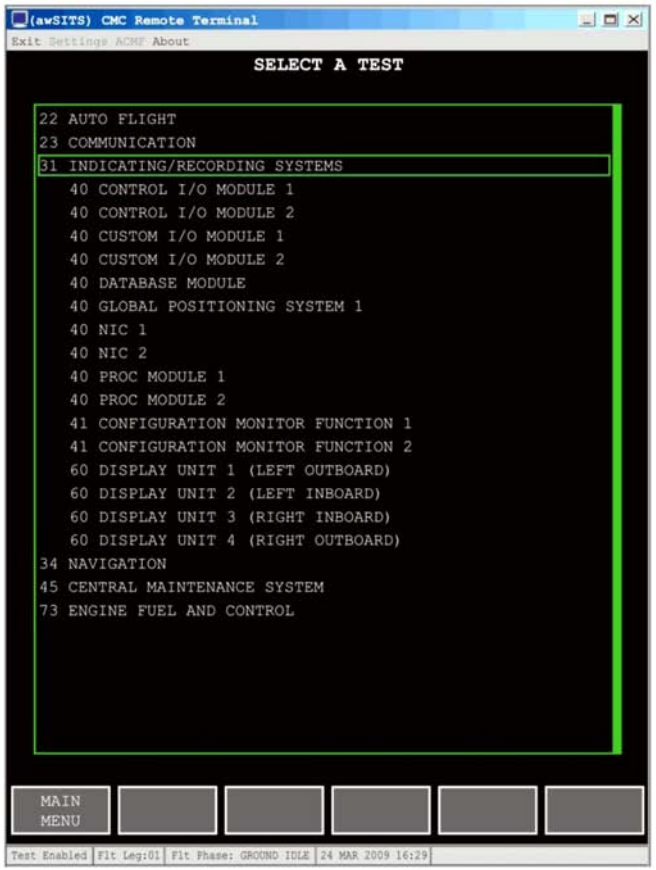
- MAIN MENU - goes to the main menu
- PREV - goes to the previous page
- FDE - goes back to the STORED CORRELATED FDES page.



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COURSE CONTROL DEVICE



ICN-39-A-454500-G-A0126-03018-A-01-1

SELECT A TEST PAGE (SHEET 1 OF 3)

SYSTEM DIAGNOSTICS MENU

When the SYSTEM DIAGNOSTICS button is pushed from the CMC MAIN MENU the SELECT A LIST page is shown. Maintenance personnel use the SELECT A LIST to make a selection of a specific test and then initiate the test pushing the ENTER button on the CCD.
The bezel button is available as follows:

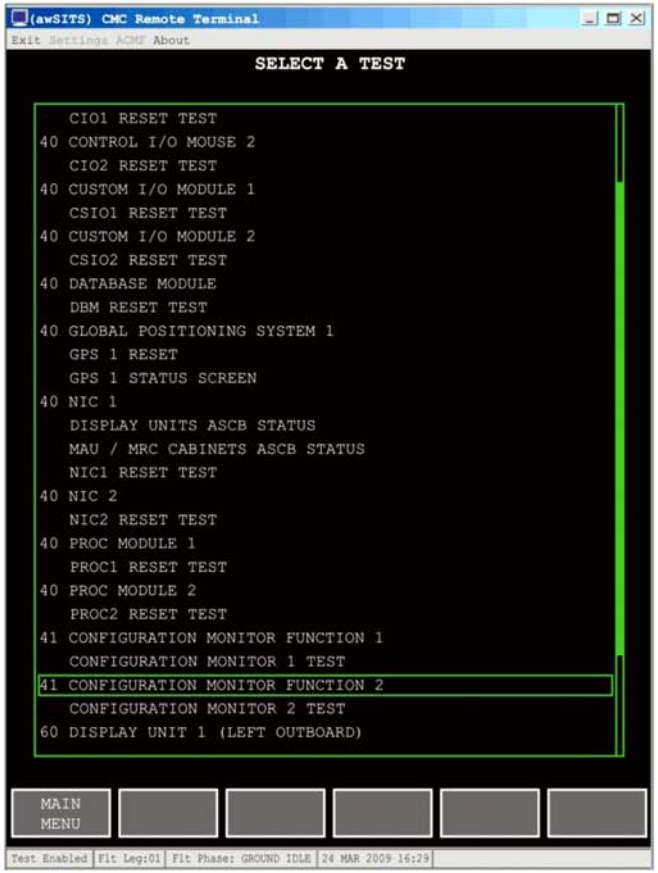
- MAIN MENU - goes to the main menu.



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COURSE CONTROL DEVICE



ICN-39-A-454500-G-A0126-03019-A-01-1

SELECT A TEST PAGE (SHEET 2 OF 3)

TEST INITIALIZING PAGE

When the test starts, the test page shows with TEST INITIALIZING text near the top of the page. The test results and status show on the test page when the test is done.
This page is only shown during test start (initialization) and then is automatically changed to the test results page.
The bezel buttons are available as follows:

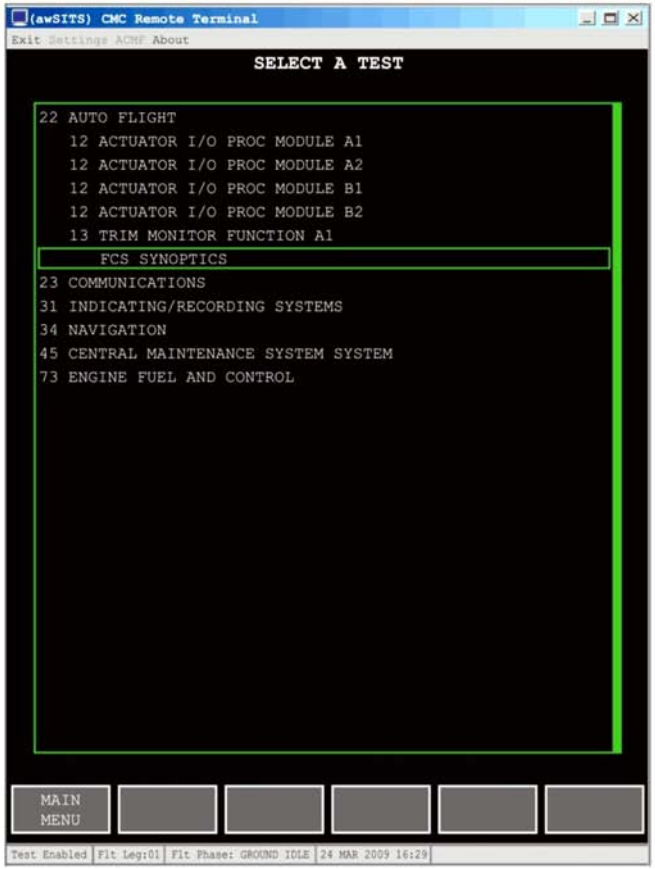
- MAIN MENU - goes to the main menu
- ABORT - stops the test during the initialization procedure.



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COURSE CONTROL DEVICE



ICN-39-A-454500-G-A0126-03020-A-01-1

SELECT A TEST PAGE (SHEET 3 OF 3)

TEST STATUS AND RESULTS PAGES

The TEST STATUS and RESULTS pages show the results of a test on one or more pages. The top of the page shows the name of the test that was done and the centre page shows details relevant to the error sensed.
The bezel buttons are available as follows (The available buttons change relevant to the page that shows.):

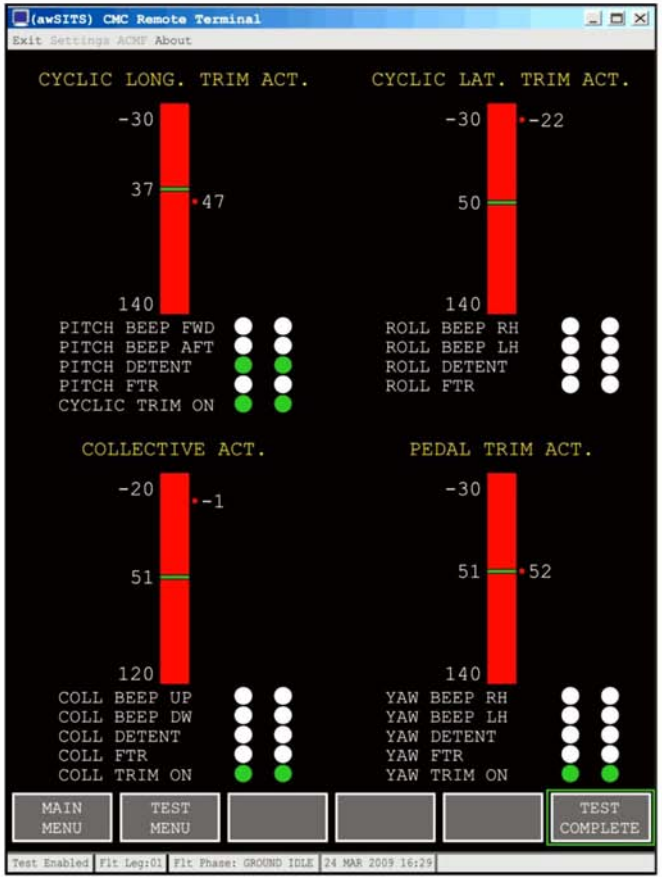
- MAIN MENU - goes to the main menu
- TEST MENU - goes to the test menu
- FWD - shows the next results or status page
- PREV - shows the previous results or status page
- TEST COMPLT - goes back to the SELECT A TEST page.



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COURSE CONTROL DEVICE



ICN-39-A-454500-G-A0126-03021-A-01-1

FCS SYNOPTICS TEST PAGE

EXTENDED MAINTENANCE MENU

Maintenance personnel use the EXTENDED MAINTENANCE page to make a selection of the type of data to show.
The menu selections are as follows:

- STORAGE ALLOWED - this function controls the storage of faults in the FHDB. Selection of the storage label on the EXTENDED MAINTENANCE menu changes the storage label between STORAGE ALLOWED and STORAGE DISALLOWED.
- MEMBER SYSTEM STATUS - goes to the member system status list. This selection, sequenced by ATA chapter, supplies a display of which member systems currently operate.
- FILE TRANSFER - moves data to or from the aircraft.
- CONFIGURATION - goes to the member system configurations list. This selection, sequenced by ATA chapter, shows the member system configuration data source from the member system.
- REPORTS - shows, prints or moves the maintenance reports.
- RETURN TO MAIN MENU - goes back to the main menu.

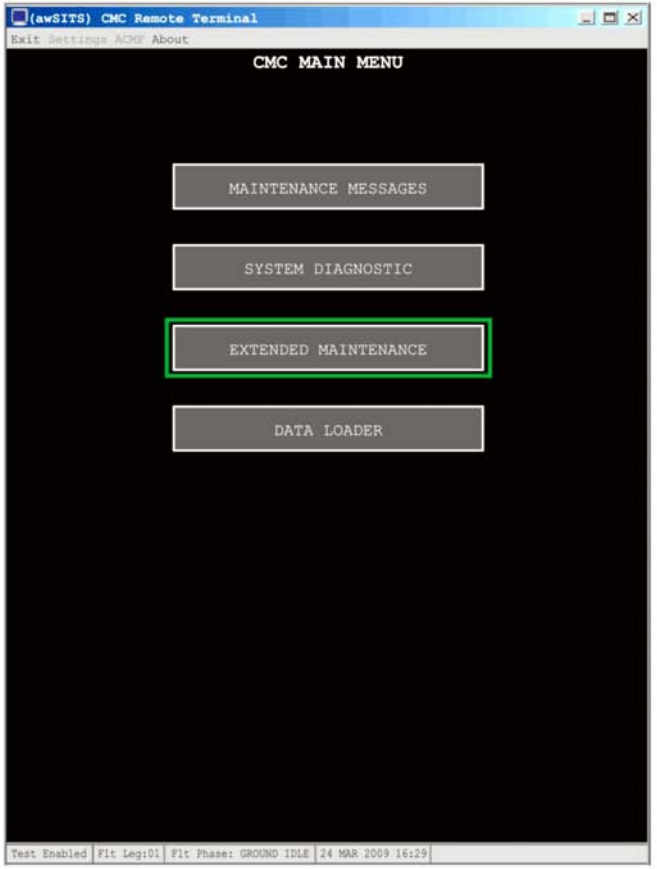
No bezel buttons are available on this menu.



CMC REMOTE TERMINAL TOOL.lnk



COURSE CONTROL DEVICE



ICN-39-A-454500-G-A0126-03022-A-01-1

CMC MAIN MENU PAGE – EXTENDED MAINTENANCE SELECTION

MEMBER SYSTEM STATUS PAGE

Maintenance personnel use the MEMBER SYSTEM STATUS page to get the status from a specified unit or function.
Member systems are listed by ATA number. To get the status, set the necessary line and push the ENTER key to expand it.
The status shows below the line selection.
Chapters with all specified member systems that report as OPERATIONAL show in white. If an ATA chapter has one or more specified member systems that are not OPERATIONAL, the ATA chapter name is in cyan. When the ATA chapter is expanded, the system that does not operate shows with a short text message in cyan.
The bezel buttons are available as follows:

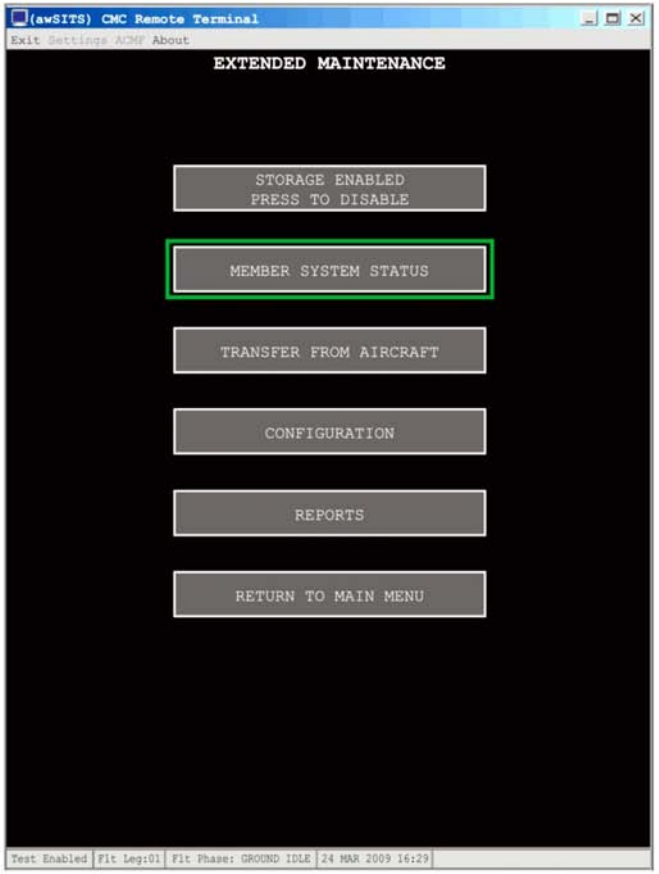
- MAIN MENU - goes to the main menu
- PREV - goes to the previous page.



CMC REMOTE TERMINAL TOOL.lnk



COURSE CONTROL DEVICE



ICN-39-A-454500-G-A0126-03023-A-01-1

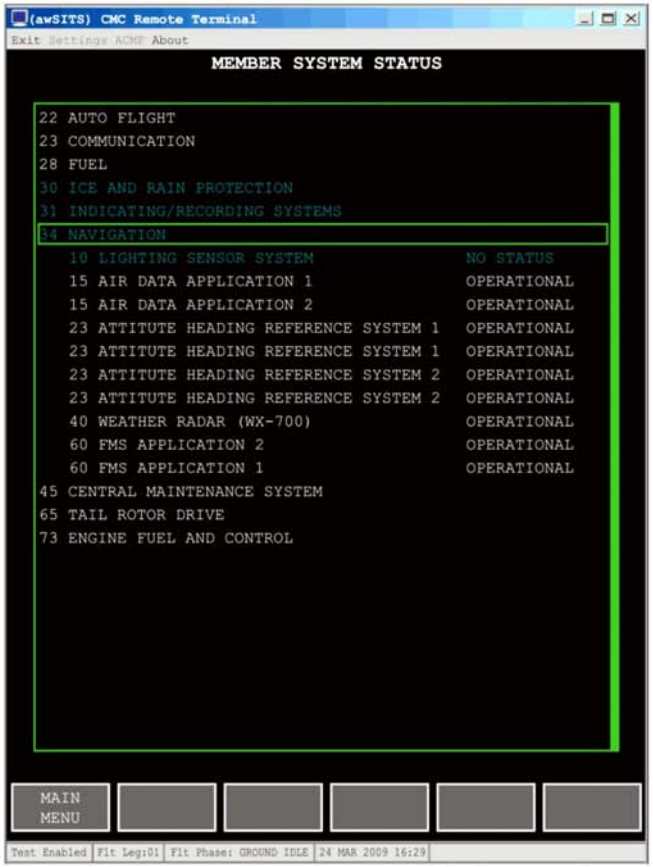
EXTENDED MAINTENANCE PAGE – MEMBER SYSTEM STATUS SELECTION



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COURSE CONTROL DEVICE



ICN-39-A-454500-G-A0126-03024-A-01-1

MEMBER SYSTEM STATUS PAGE

FILE TRANSFER MENU

Use the FILE TRANSFER MENU to make a selection of the type of data movement. Data can be sent to or from the aircraft.
The menu selections are as follows:

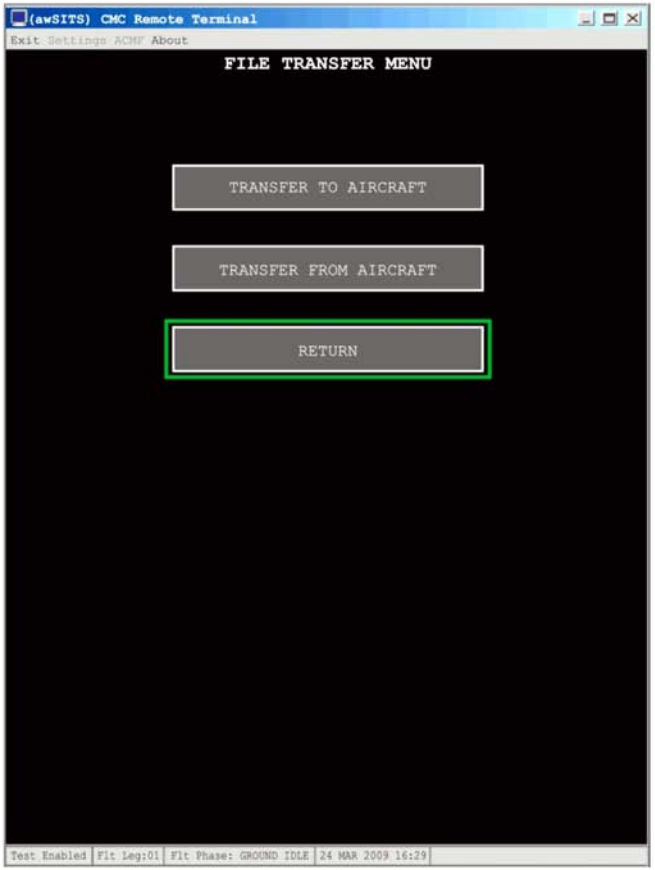
- TRANSFER TO AIRCRAFT - goes to the file transfer to aircraft list
- TRANSFER FROM AIRCRAFT - goes to the file transfer from aircraft list
- RETURN - goes back to the extended Maintenance menu.
- No bezel buttons are available on this menu.



CMC REMOTE TERMINAL TOOL.lnk



COURSE CONTROL DEVICE



ICN-39-A-454500-G-A0126-03025-A-01-1

FILE TRANSFER MENU PAGE

FILE TRANSFER TO AIRCRAFT PAGE

Use the FILE TRANSFER TO AIRCRAFT page to make a selection of the data source and file to be moved . The file selection is copied from the CMC-RT and sent to the CMS.
The bezel buttons are available as follows:

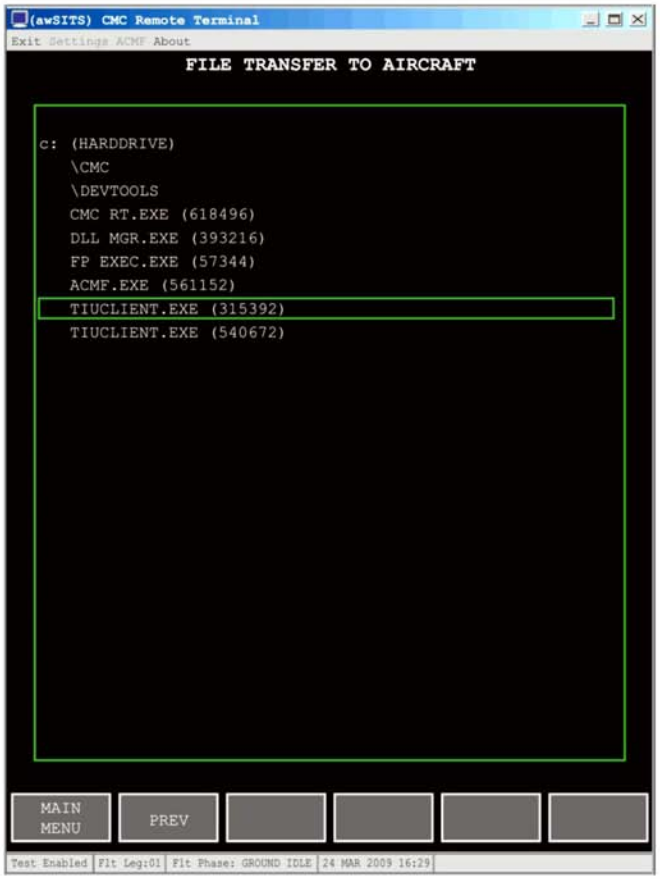
- MAIN MENU - goes to the main menu
- PREV - goes to the previous page.



CMC REMOTE TERMINAL TOOL.lnk



COURSE CONTROL DEVICE



ICN-39-A-454500-G-A0126-03026-A-01-1

FILE TRANSFER TO AIRCRAFT PAGE

FILE TRANSFER FROM AIRCRAFT PAGE

Use the FILE TRANSFER FROM AIRCRAFT page to make a selection of the data source (file) to be sent to the CMC-RT. There is a wide green bar on the right of the larger green cursor box. This bar shows that more data is available than shows in the boxed window. Move the green cursor box to the upper or lower limits of the page to move the page contents.
The green bar shows the direction and relative amount of data that does not show.
The bezel buttons are available as follows:

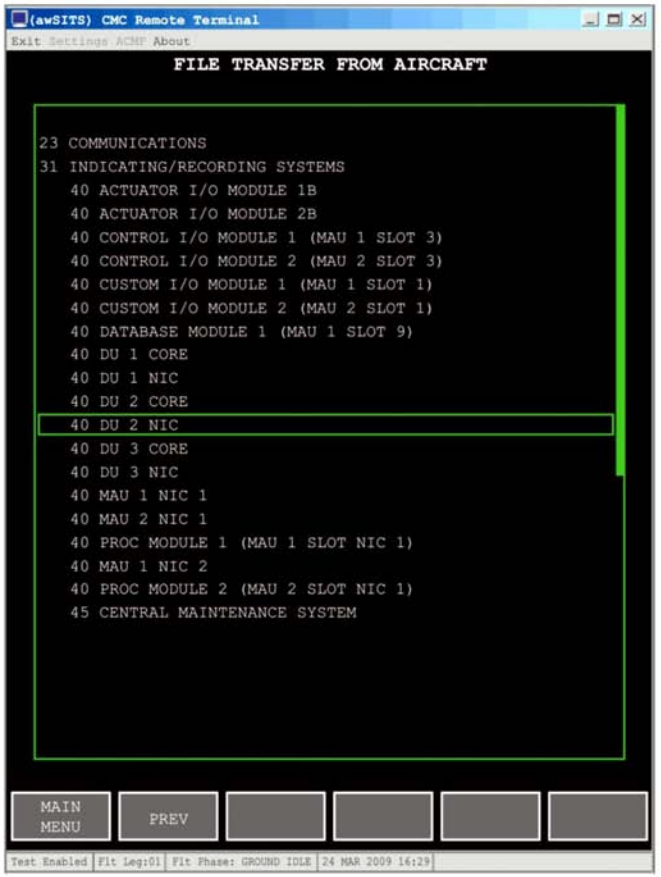
- MAIN MENU - goes to the main menu
- PREV - goes to the previous page.



CMC REMOTE TERMINAL TOOL.lnk



COURSE CONTROL DEVICE



ICN-39-A-454500-G-A0126-03027-A-01-1

FILE TRANSFER FROM AIRCRAFT PAGE

SYSTEM CONFIGURATION PAGE

The SYSTEM CONFIGURATION page gets data about the member systems connected to the maintenance system.
The bezel buttons are available as follows:

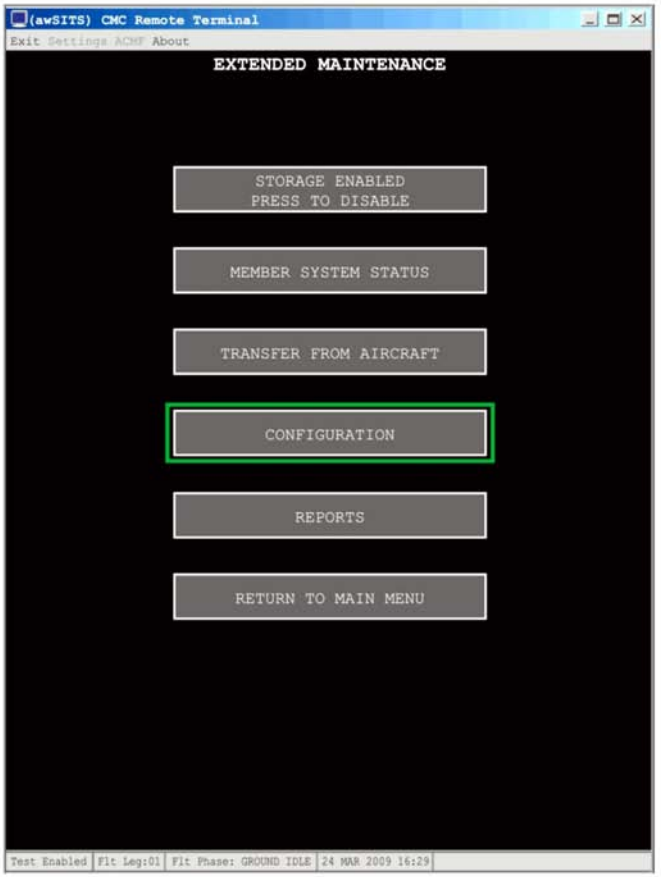
- MAIN MENU - goes to the main menu
- PREV - goes to the previous page.



CMC REMOTE TERMINAL TOOL.lnk



COURSE CONTROL DEVICE



ICN-39-A-454500-G-A0126-03028-A-01-1

EXTENDED MAINTENANCE PAGE – CONFIGURATION SELECTION



CMC REMOTE TERMINAL TOOL.lnk



COURSE CONTROL DEVICE



ICN-39-A-454500-G-A0126-03029-A-01-1

SYSTEM CONFIGURATION PAGE (SHEET 1 OF 2)



CMC REMOTE TERMINAL TOOL.lnk



COURSE CONTROL DEVICE



ICN-39-A-454500-G-A0126-03030-A-01-1

SYSTEM CONFIGURATION PAGE (SHEET 2 OF 2)

CMC REPORTS PAGE

Use the CMC REPORTS page to set, print, and move reports from a member to storage or an output device.
Report selections are as follows:

- ACMFREPORTS
- CMCREPORTS.

When the CMC REPORTS selection is expanded, the options are available that follow:

- ACTIVE MAINTENANCE MESSAGE REPORT
- CURRENT LEG FAILURE REPORT
- CONFIGURATION REPORT
- EXPORTFAULTHISTORY.

Each report can be sent to local storage or sent to the printer. The bezel buttons are available as follows:

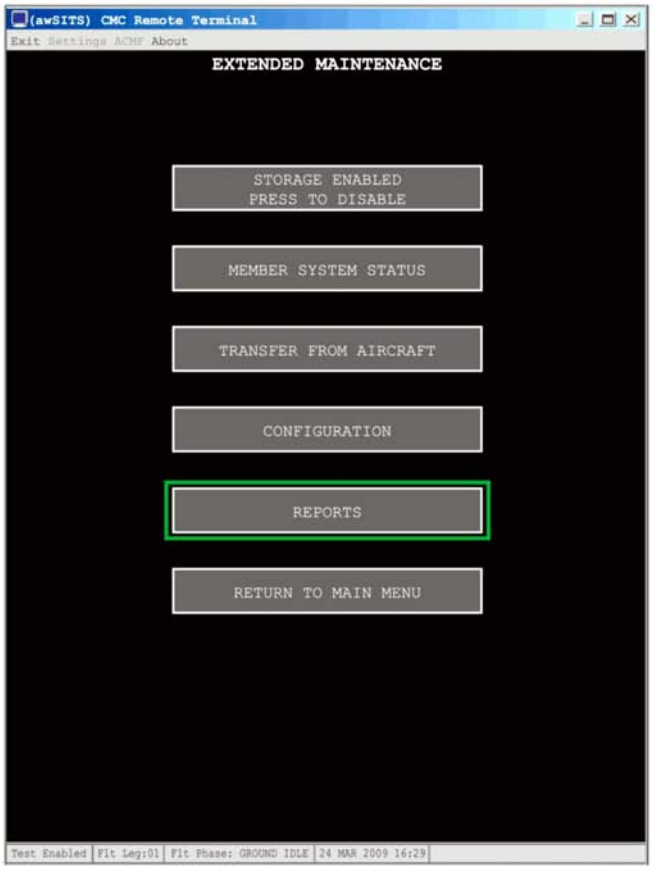
- MAIN MENU - goes to the main menu
- PREV - goes to the previous page.



CMC REMOTE TERMINAL TOOL.lnk



COURSE CONTROL DEVICE



ICN-39-A-454500-G-A0126-03031-A-01-1

EXTENDED MAINTENANCE PAGE – REPORTS SELECTION



CMC REMOTE TERMINAL TOOL.lnk



COURSE CONTROL DEVICE



ICN-39-A-454500-G-A0126-03032-A-01-1

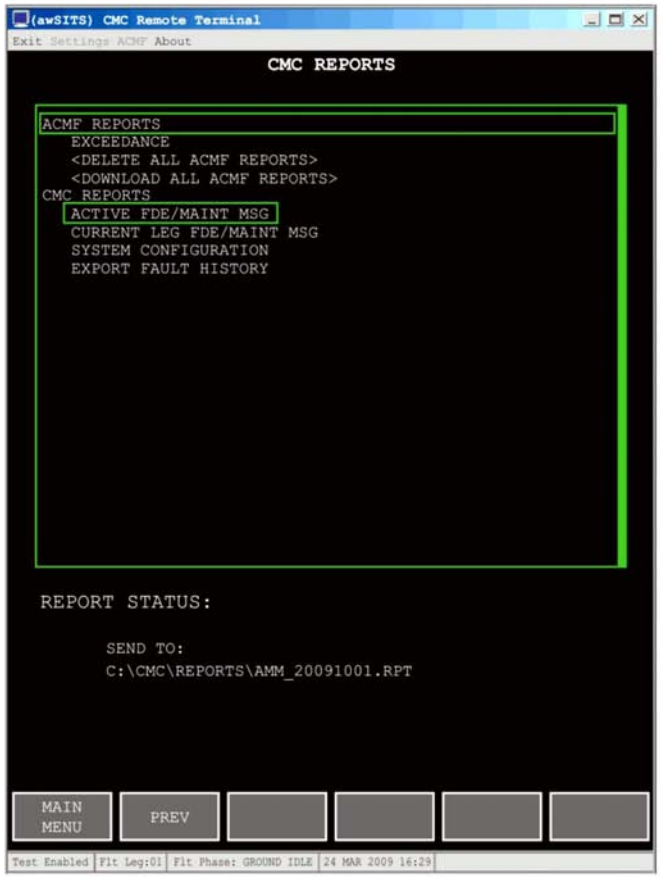
CMC REPORTS PAGE (SHEET 1 OF 2)



CMC REMOTE TERMINAL TOOL.lnk



COURSE CONTROL DEVICE



ICN-39-A-454500-G-A0126-03033-A-01-1

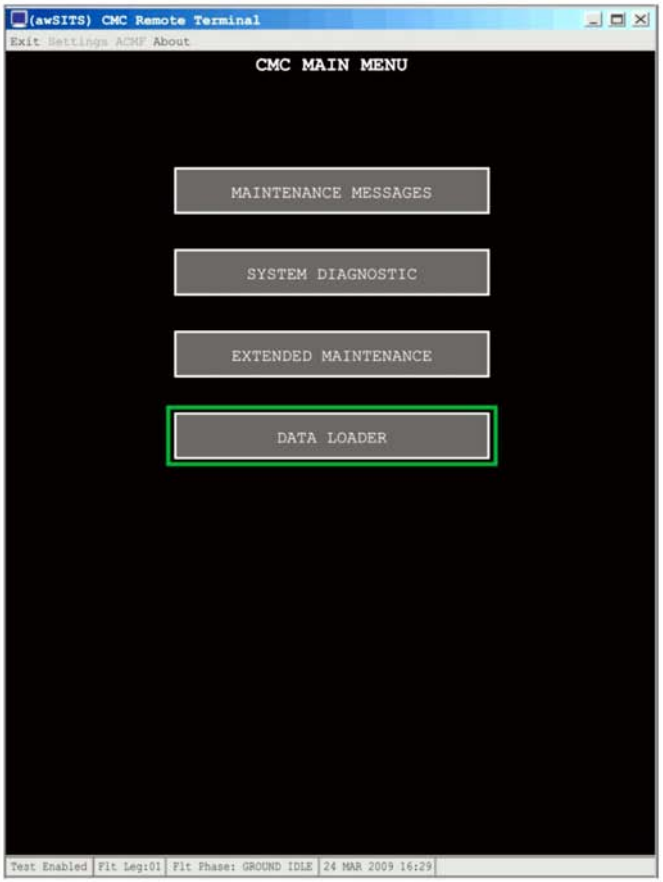
CMC REPORTS PAGE (SHEET 2 OF 2)



CMC REMOTE TERMINAL TOOL.lnk



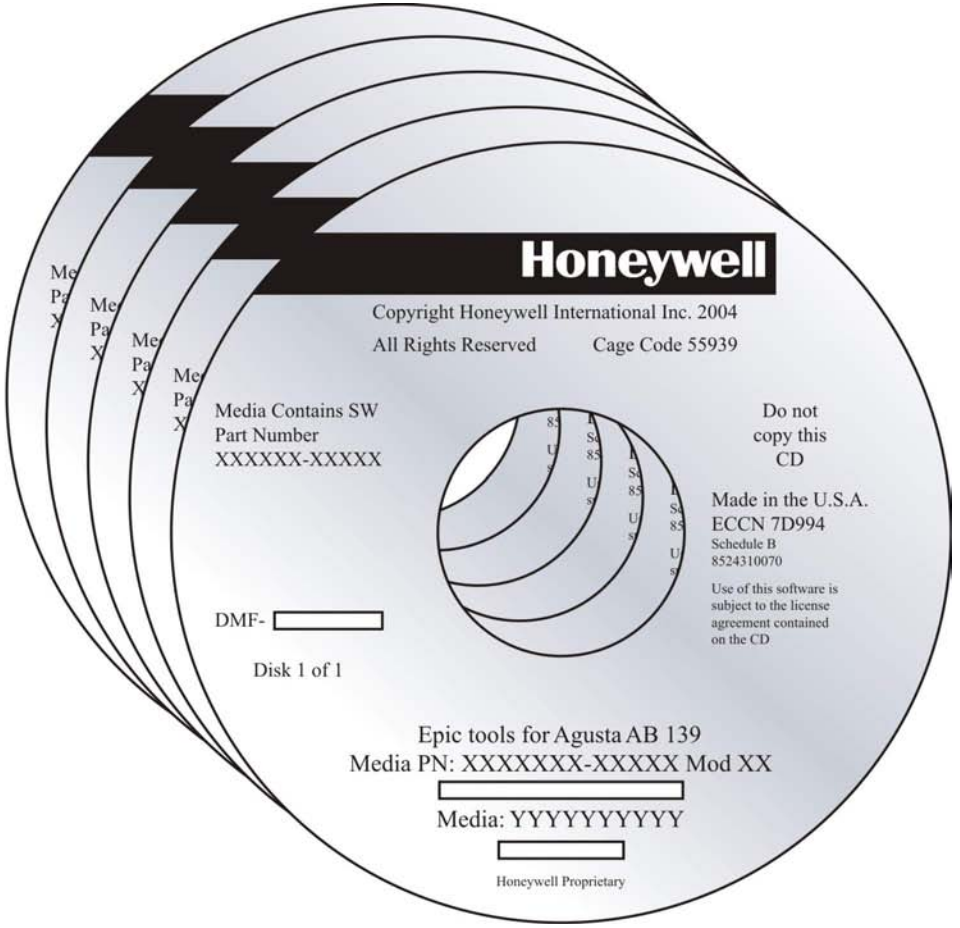
COURSE CONTROL DEVICE



ICN-39-A-454500-G-A0126-03034-A-01-1

CMC MAIN MANU PAGE - DATA LOADER SELECTION

- EPIC tools for Agusta AW139
- Operational Flight software
- Option Files
- Operational software NAV_AC_DB
- Operational software LDI



ICN-39-A-454500-G-A0126-00630-A-01-1

PRIMUS EPIC CD-ROM



CMC REMOTE TERMINAL TOOL.lnk



COURSE CONTROL DEVICE



ICN-39-A-454500-G-A0126-03035-A-01-1

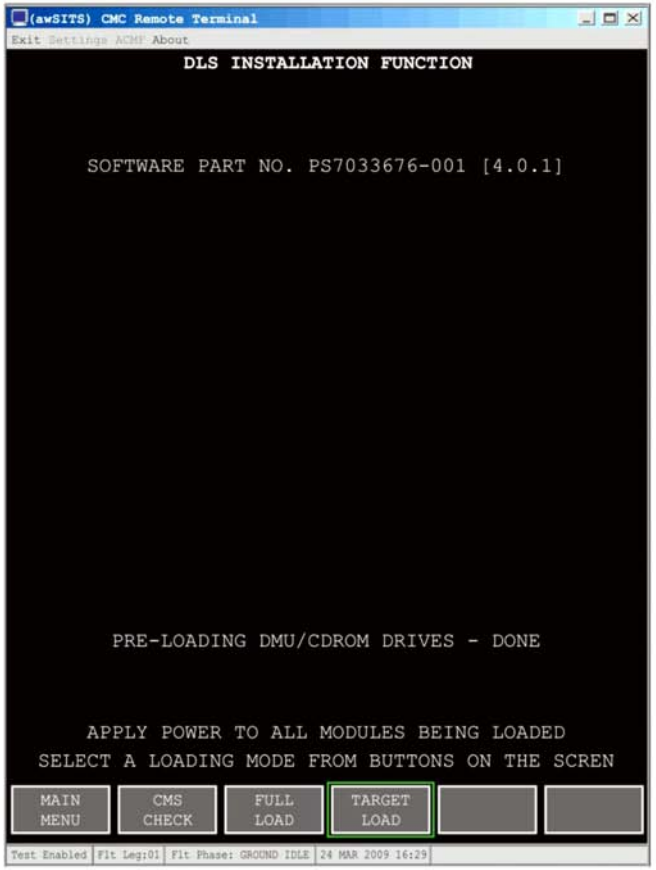
DLS INSTALLATION FUNCTION PAGE (SHEET 1 OF 2)



CMC REMOTE TERMINAL TOOL.lnk



COURSE CONTROL DEVICE

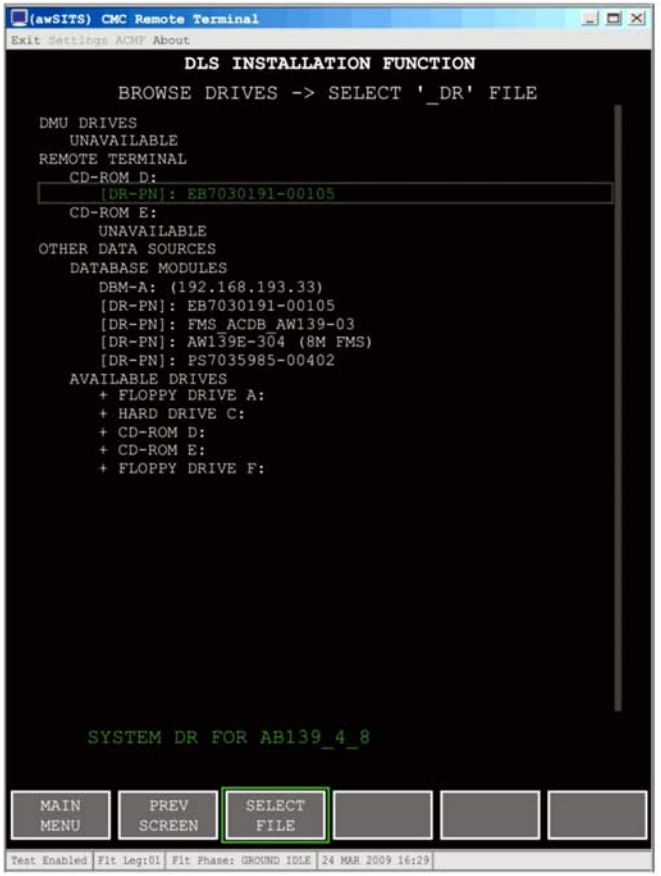


ICN-39-A-454500-G-A0126-03036-A-01-1

DLS INSTALLATION FUNCTION PAGE (SHEET 2 OF 2)



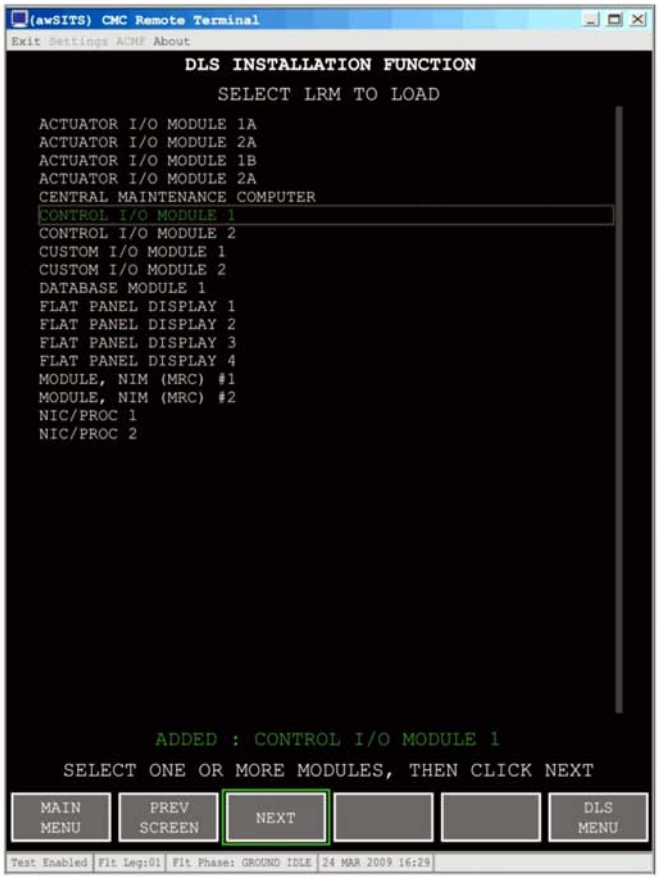
CMC REMOTE TERMINAL TOOL.lnk



ICN-39-A-454500-G-A0126-03037-A-01-1

DLS – SELECT [DR-PN] FILE PAGE



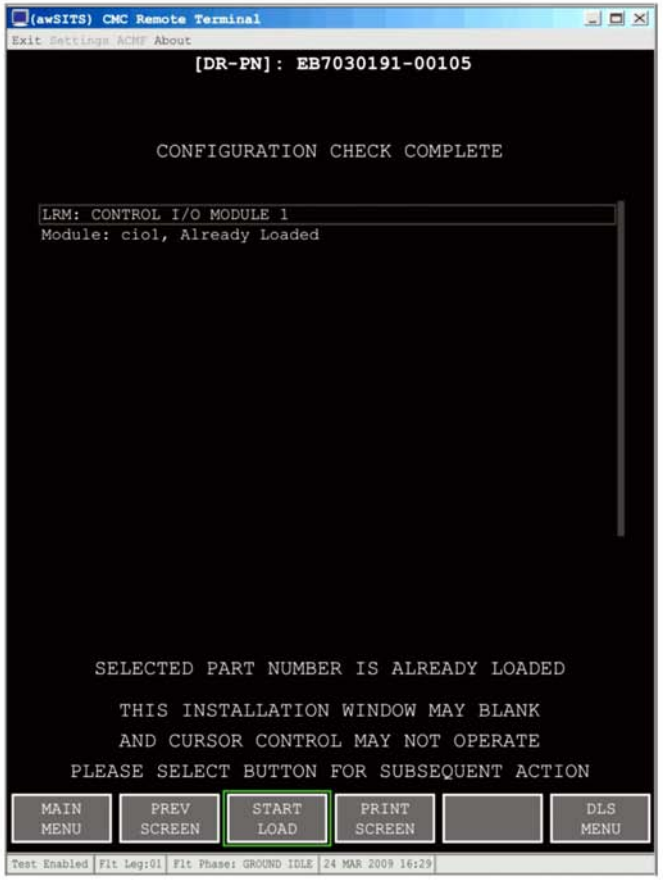


ICN-39-A-454500-G-A0126-03038-A-01-1

DLS – SELECT LRM TO LOAD PAGE



CMC REMOTE TERMINAL TOOL.lnk

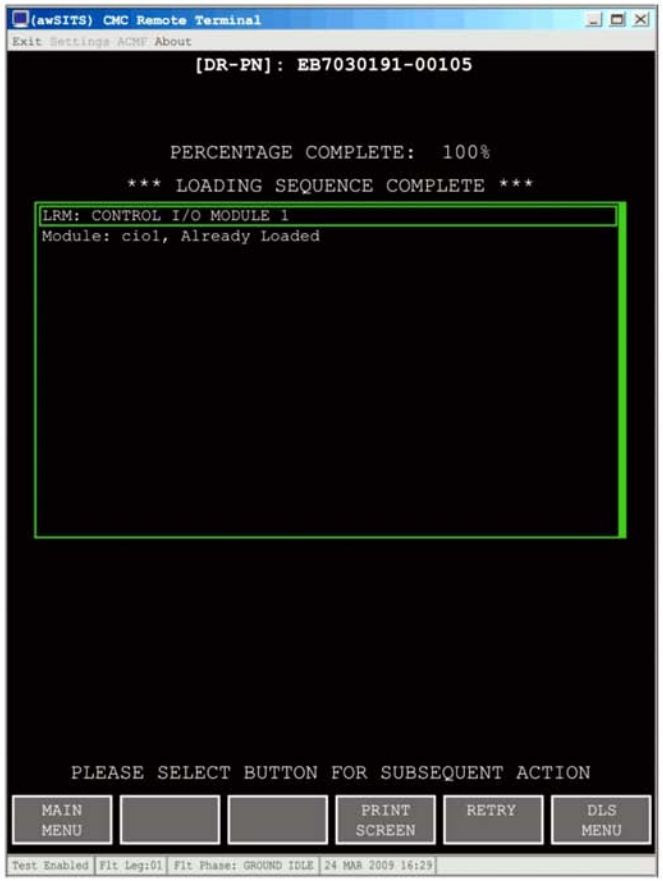


ICN-39-A-454500-G-A0126-03039-A-01-1

DLS – CONFIGURATION CHECK PAGE



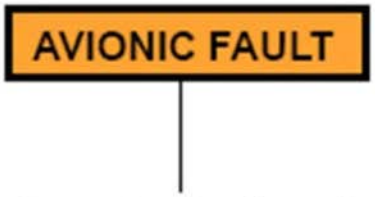
CMC REMOTE TERMINAL TOOL.lnk



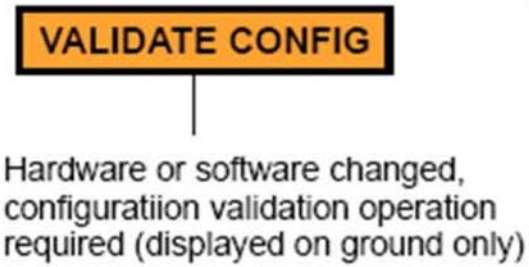
ICN-39-A-454500-G-A0126-03040-A-01-1

DLS – LOADING PAGE

Set the electrical power off to reset the modules. Then start again after 4 minutes. Make sure that the caution messages that follow are not shown in the CAS window of the MFD:

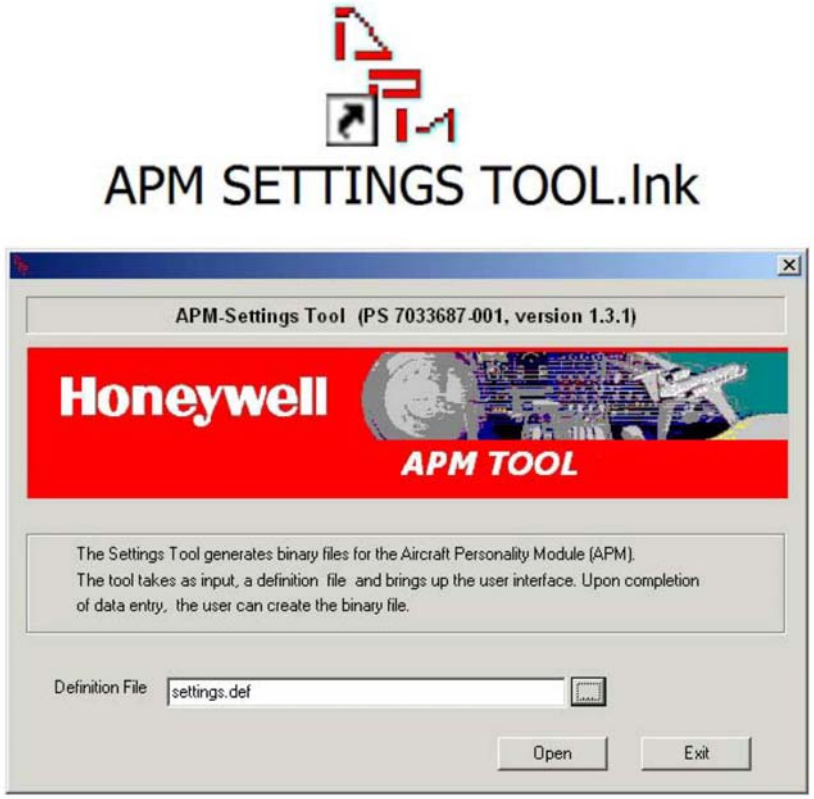






ICN-39-A-454500-G-A0126-00637-A-01-1

AFTER LOADING PROCEDURES



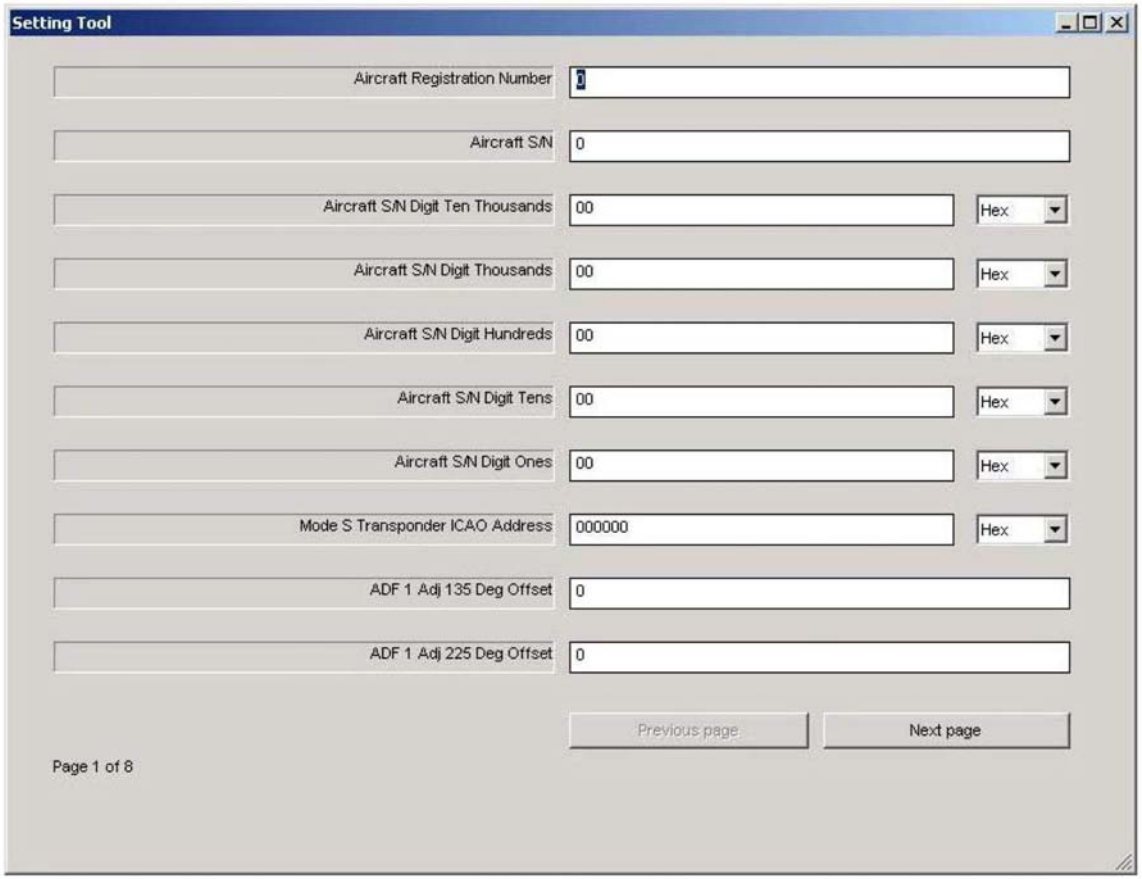
ICN-39-A-454500-G-A0126-00638-A-01-1

APM SETTING TOOL PROGRAM



ICN-39-A-454500-G-A0126-00639-A-01-1

APM – SELECT DEFAULT DATA SOURCE PAGE



ICN-39-A-454500-G-A0126-00640-A-01-1

APM – SETTING TOOL PAGE 1 OF 8

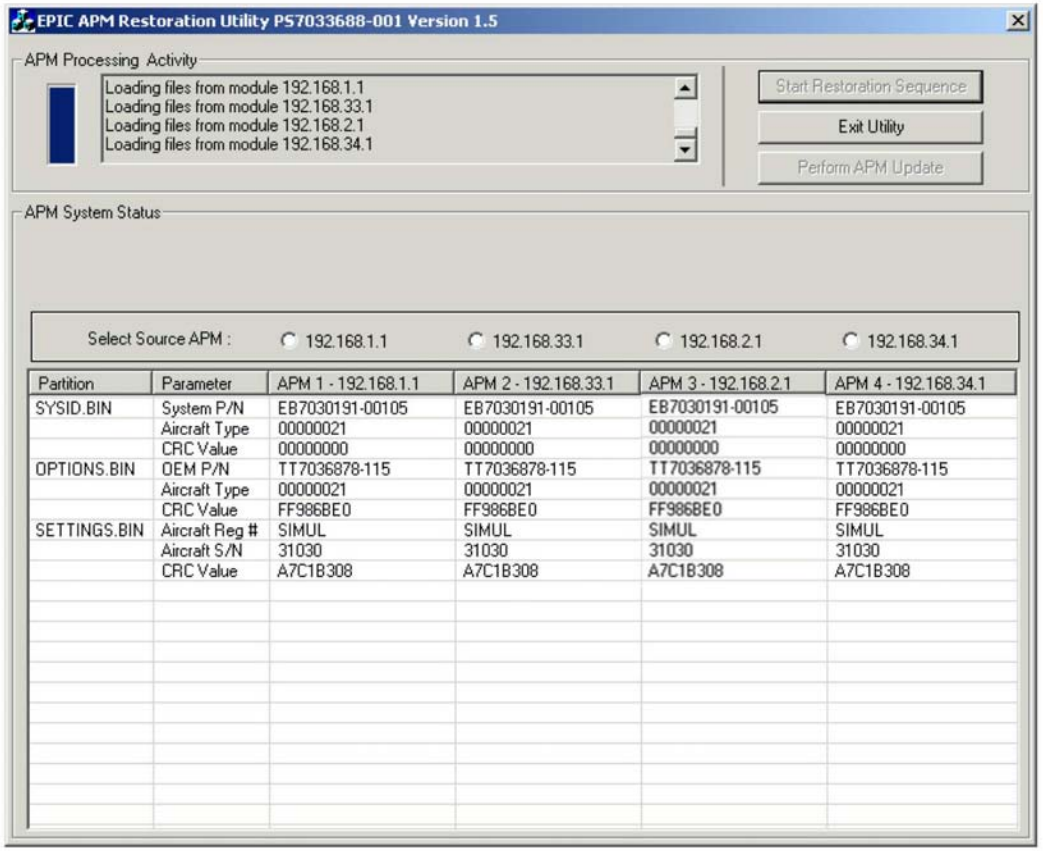


APM RESTORATION TOOL.lnk



ICN-39-A-454500-G-A0126-00641-A-01-1

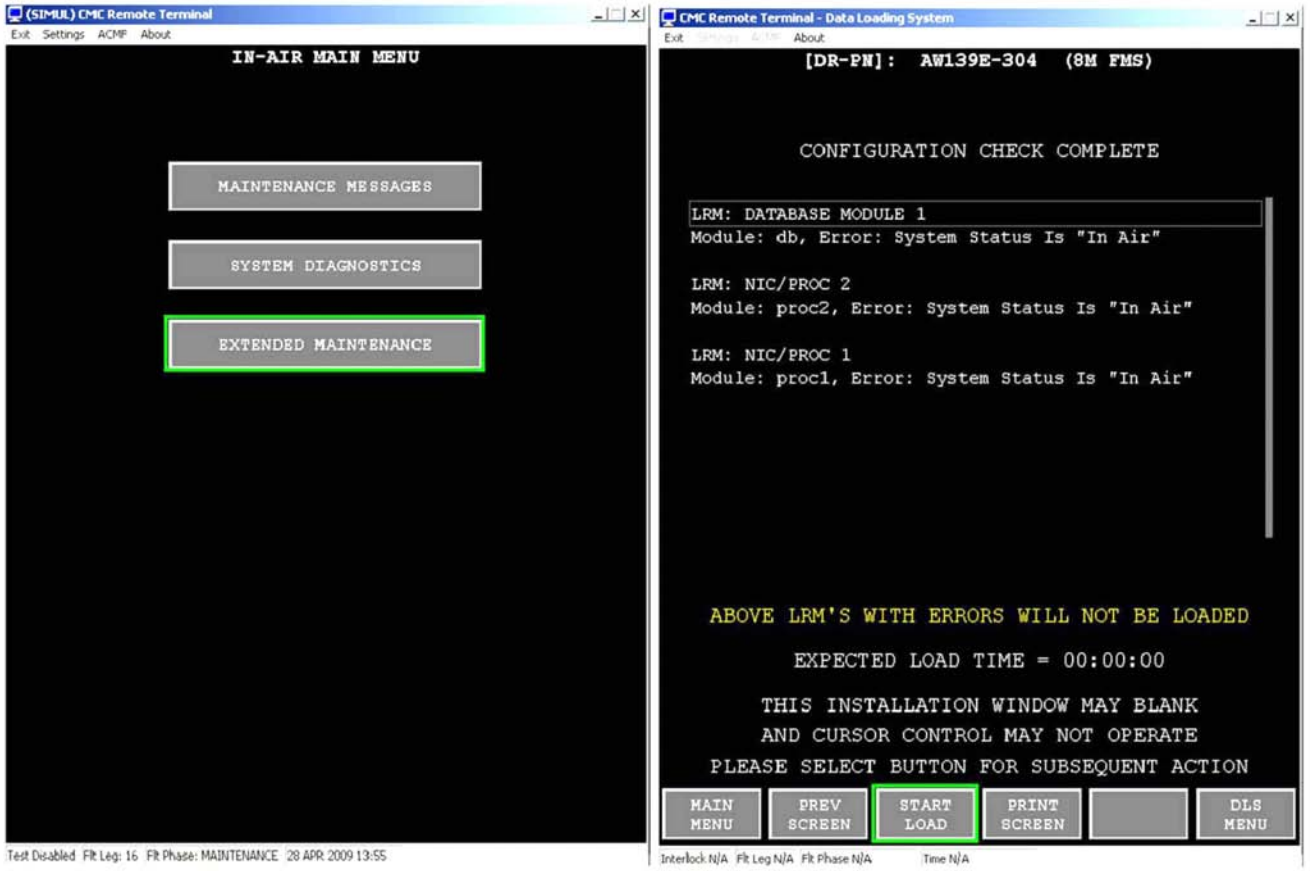
APM – RESTORATION TOOL PROGRAM



Partition	Parameter	APM 1 - 192.168.1.1	APM 2 - 192.168.33.1	APM 3 - 192.168.2.1	APM 4 - 192.168.34.1
SYSID.BIN	System P/N	EB7030191-00105	EB7030191-00105	EB7030191-00105	EB7030191-00105
	Aircraft Type	00000021	00000021	00000021	00000021
	CRC Value	00000000	00000000	00000000	00000000
OPTIONS.BIN	OEM P/N	TT7036878-115	TT7036878-115	TT7036878-115	TT7036878-115
	Aircraft Type	00000021	00000021	00000021	00000021
	CRC Value	FF986BE0	FF986BE0	FF986BE0	FF986BE0
SETTINGS.BIN	Aircraft Reg #	SIMUL	SIMUL	SIMUL	SIMUL
	Aircraft S/N	31030	31030	31030	31030
	CRC Value	A7C1B308	A7C1B308	A7C1B308	A7C1B308

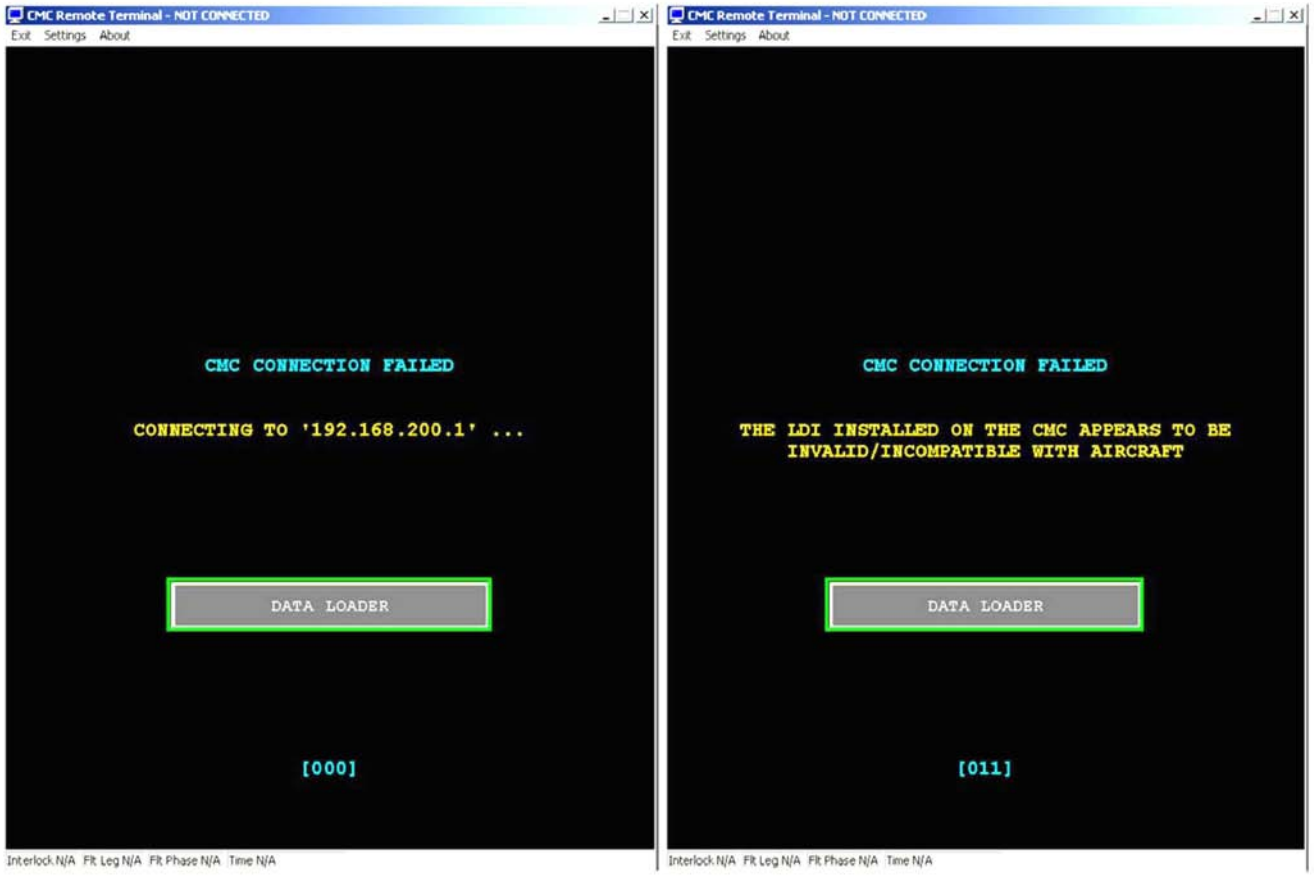
ICN-39-A-454500-G-A0126-00642-A-01-1

APM – RESTORATION TOOL PAGE



ICN-39-A-454500-G-A0126-00643-A-01-1

CMS - CMC IN-AIR CONDITION PAGES



ICN-39-A-454500-G-A0126-00644-A-01-1

CMS - CMC CONNECTION FAILED PAGES

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CHAPTER 46

SYSTEM INTEGRATION AND DISPLAY

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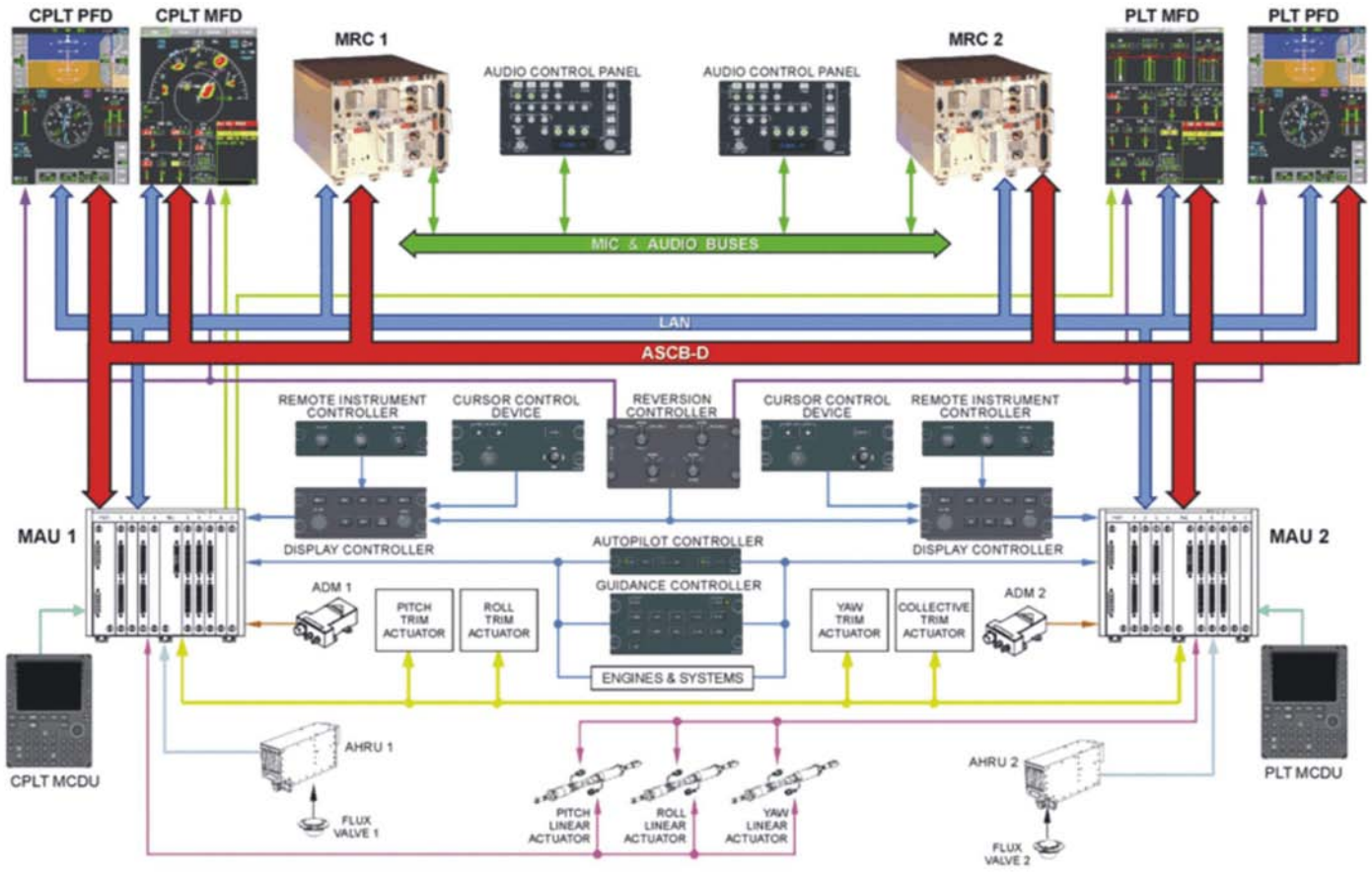
SECTION 00 - SYSTEMS INTEGRATION AND DISPLAY

GENERAL DESCRIPTION

The helicopter uses the Honeywell PRIMUS EPIC system for the integration of the different avionics systems. PRIMUS EPIC is a US-registered trademark of Honeywell. The PRIMUS EPIC integrated avionics system is referred to in this manual as the integrated avionics system.
The integrated avionics system contains the systems and functions necessary to operate the helicopter. These systems and functions are:

- Auto flight (39-A-22-00-00-00A-000A-A)
- Communications (39-A-23-00-00-00A-000A-A)
- Indicating/recording (39-A-31-00-00-00A-000A-A)
- Navigation (39-A-34-00-00-00A-000A-A)
- Central maintenance (39-A-45-00-00-00A-000A-A).

The integrated avionics system has modular avionics units. The modular avionics units contain the hardware units that make the subsystems and functions necessary to operate the helicopter. The integrated avionics system uses flat-panel display-units to show data in the cockpit. It uses mechanical and virtual controllers to control the helicopter. It includes a maintenance system to monitor the operation of the helicopter.
The Integrated Avionics interfaces all the helicopter system sensors for status and parameter values display.



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INTEGRATED AVIONICS SYSTEM BLOCK DIAGRAM

ARCHITECTURE

The architecture of the integrated avionics system includes two primary structures:

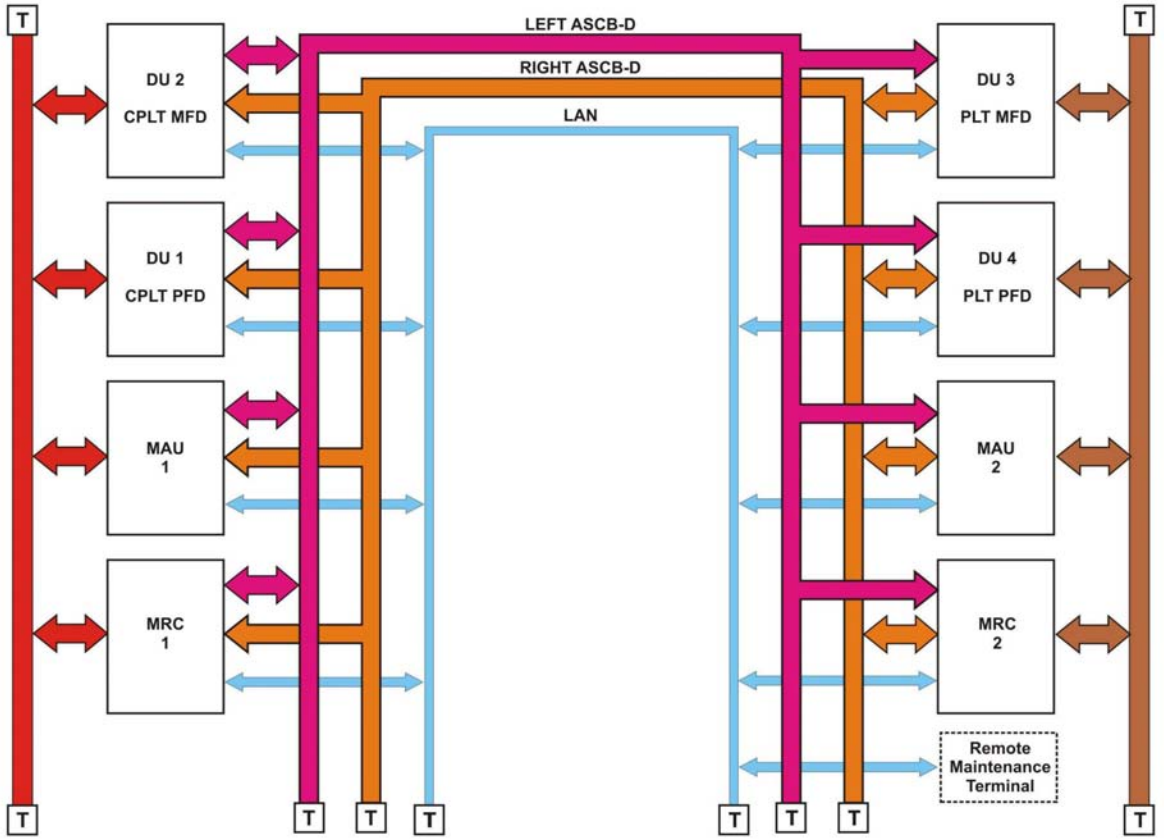
- The virtual backplane network
- The modular avionics unit (MAU) architecture.

VIRTUAL BACKPLANE NETWORK

The units in the integrated avionics system use the virtual backplane network to send data to each other. The virtual backplane network includes:

- The Avionics Standard Communication Bus, version-D (ASCB-D), which is a high-integrity, bus network that goes to all zones of the helicopter
- The software and hardware in the units that transmit and receive data on the ASCB-D.

The MAUs, the display units and the Modular Radio Cabinets (MRC) are directly connected to the ASCB-D.
The ASCB-D network lets these units transmit and receive data at the highest possible speed, as a function of data availability and reliability. Each ASCB-D bus uses tapped twin axial wire. Data are moved through the bus at 10 Mb/s, thus large volumes of data can be moved quickly. The ASCB-D network includes four buses that are the same, and make sure that no data is lost. Thus the reliability of the system is increased. The protocols of the ASCB-D keep all four buses synchronized at the ASCB-D transmission rates that are: 80, 40, 10, 5 and 1 Hz.
A Local Area Network (LAN) is also supplied together with the ASCB-D network. The LAN is related to Ethernet 10-Base-2 and is available to all the units that connect to the ASCB-D. The integrated avionics system uses the LAN for data upload/download, and the maintenance and test functions. The maintenance is done on the ground by connection of a standard laptop computer to the LAN. The configuration of the laptop computer must be that of a maintenance terminal: 10-Base-2 hardware and proprietary software. The maintenance terminal and its CD or 3.5 inch disk drive are used to upload/download the data. Database files and/or software are uploaded to the system. Stored data (memory) or the fault history files are downloaded.
Each unit that connects to the ASCB-D uses a Network Interface Controller (NIC). The NIC gives the interface between the unit and the ASCB-D. This interface lets the internal modules or circuit cards of the unit transmit to, and receive data from the ASCB-D. All the modules that connect to the ASCB-D must be backplane compatible. The NIC can also do configuration checks and monitor the fan.



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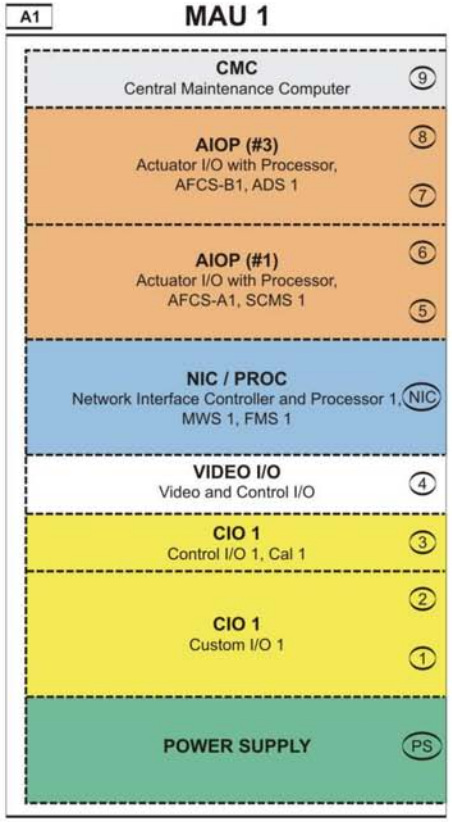
ASCB-D AND LAN DATA BUS

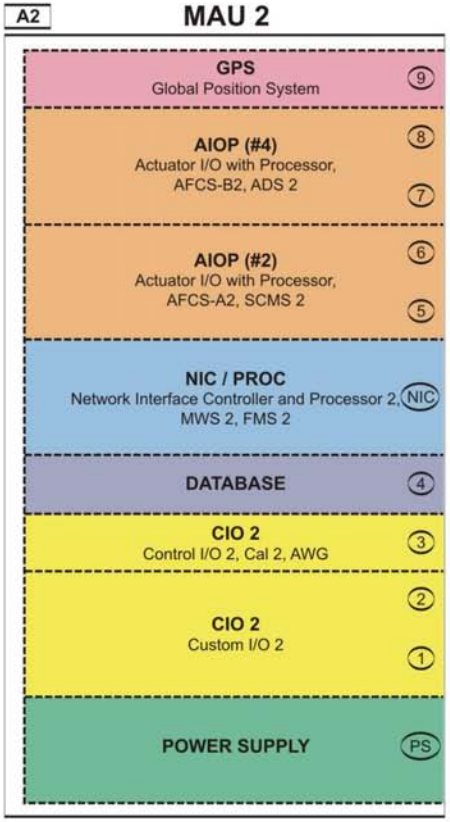
MAU ARCHITECTURE

The MAUs in the helicopter have Digital Engine Operating System (DEOS) compliant processing, and include the modules that follow:

- Power Supply Module (PSM)
- Network Interface Controller with Processor (NIC/PROC)
- Custom I/O (CSIO) module
- Control I/O (CIO) module
- Actuator I/O Processor (AIOP) module
- Central Maintenance Computer (CMC)
- Database Module (DBM)
- Global Positioning System (GPS) (optional)
- Video CIO (optional).

The MAUs transmit and receive the data through the ASCB-D bus and the LAN bus. The Custom I/O (CSIO), Control I/O (CIO) and the other modules send the data from the sensors and the systems to the processor modules. The processor modules process the data to control and monitor the helicopter.
The NIC transmits and receives data from the ASCB-D and the LAN, and makes them available to the other modules in the MAU. The data are made available through a special Peripheral Component Interconnect (PCI) backplane that has this name: Virtual back Plane Peripheral Component Interconnect (VbPCI) bus. The VbPCI is a 32-bit wide passive, parallel backplane bus. The system uses this bus to connect the NIC to all the modules in the MAU on the ASCB-D. All module data in these units (I/O included) are available to all other modules or to the ASCB-D in the integrated avionics system.
A standard hardware interface the name of which is Backplane Interface Controller (BIC) supplies the connection to the MAU VbPCI. The BIC is installed on the MAU modules. The BIC stores and sends the ASCB-D and LAN data it receives from the NIC to the modules in sequence. But the transmission occurs only when the modules are prepared to receive the data. The hardware mechanism to store, transmit and receive the data in sequence is the BIC frame buffer. The BIC frame buffer has dual ports and uses Random Access Memory (RAM). The NIC and the MAU can read and write to the BIC frame buffer.
To decode and encode the data to and from the BIC, the MAU modules use a software function of the module processing circuit. This is a standard function the name of which is Periodic Device Driver (PDD). The PDD is a high-integrity, table-driven routine. It applies ASCB-D data to software functions available on the modules that are synchronized with the ASCB-D network.
Each module in the MAU can read and write to the primary and backup buses on the same ASCB-D channel. Each module can only read from the primary ASCB-D channel on the opposite side.





ADS = Air Data System

AFCS = Automatic Flight Control System

SCMS = Software Configuration Management System

AWG = Aural Warning Generator

FMS = Flight Management System

MWS = Monitor Warning System

NIC = Network Interface Controller

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MAU 1 AND MAU 2 CONFIGURATION

DEOS

Many of the hardware components in the integrated avionics system use the same software components. These software components supply the operating system and the software routines that are necessary for the hardware component to process the data.
The DEOS is a high-integrity avionics operating system that supplies a standard set of functions for the software programs that are on the processors.

The DEOS, together with the hardware, supplies the necessary functions for time and space partitioning. Multiple functions of all the very important levels can be on the same processor.

CORE SOFTWARE COMPONENTS

Many of the processors in the system also use the core software components included in the Table.

SYSTEM CORE SOFTWARE COMPONENTS	
COMPONENT	**DESCRIPTION**
Boot	Software to start and stop the processor
LAN device driver	Software on the unit processor that manages the interface to the helicopter LAN
Periodic device driver	Software on the unit processor that manages the interface to the periodic part of the ASCB-D
Aperiodic device driver	Software on the unit processor that manages the interface to the aperiodic part of the ASCB-D
File system	Software on the unit processor that manages the storage (memory) and retrieval of data files
Core Built-in Test (BIT)	Software on the unit processor that does the BIT
Fault history manager	Software on the unit processor that manages the storage (memory) of fault conditions
File Transfer Protocol (FTP)	Software on the unit processor that does the FTP to upload or download the files
Supplemental libraries	Software libraries that do core routines (math library, for example)

DATA BUS SYSTEM

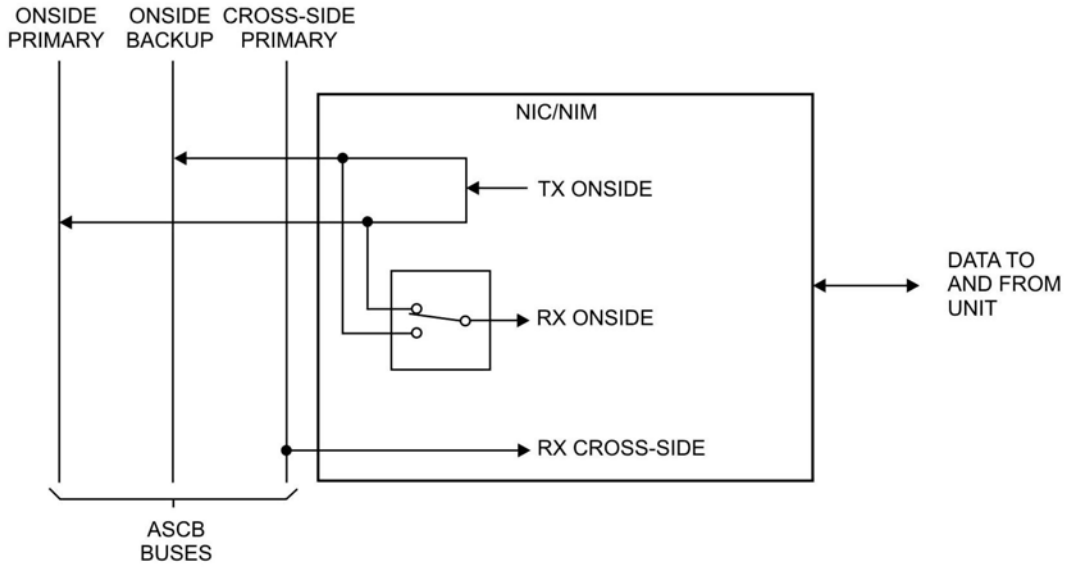
The data bus system is the primary system for the transmission of data between the subsystems, the units and the modules. The data buses supply the electrical circuits and connections necessary for data transmission between the subsystems, the units and the modules.
The most important function of the avionics system is the transfer of data between subsystems and/or units or module. Some of these data are discrete data. The discrete data move on one wire only and usually change between + 28 V dc and open, or between ground and open. The subsystems, units and modules use the discrete data for the annunciators and warnings, and all the times when these data are sufficient. Most of the data transfer is between the subsystems, and through the use of the digital data buses.
The data bus system has the buses that follow:

- Avionics Standard Communication Bus - Version D (ASCB-D)
- Local Area Network (LAN)
- Virtual backplane Peripheral Component Interconnect (VbPCI)
- Remote Image Bus (RIB)
- Control Area Network (CAN)
- Weather Radar Picture Data (WXPD)
- Digital audio bus
- Digital microphone bus
- Radio Control Bus (RCB)
- ARINC 429 data bus
- RS-422 control bus.

All units and modules in the data bus system have different addresses. The data transmitted on the ASCB-D and RSB include these addresses. When a unit or a module reads its address, it transmits data on the bus during a specified time. After a unit or a module completes the transmission of the data, it goes back to the monitor only mode.

ASCB-D DATA BUS

The ASCB-D data bus is the primary interface between the MAUs and the MRCs. The ASCB-D data bus is a high-speed data bus of the fail-operational type that includes four data buses: the left primary, the right primary, the left backup and the right backup data buses. The fail-operational data buses are those that continue to operate also if one or more units or modules connected to them become unserviceable. Each data bus is made of sets of twisted wires that are electrically isolated, and have a shield and resistor terminations. Each data bus transmits the serial data in two directions (one direction at a time) at a rate of 10 MHz. The data configuration is equivalent to that of a high-level data-link control. The isolation bus couplers isolate the four ASCB-D buses from each other. Thus a short circuit on one bus has no effect on the other buses.
A NIC/PROC module in the MAUs and a NIM in the MRCs controls the data transmitted on the ASCB-D. Each NIC/PROC module and NIM that is attached to the ASCB-D is related to one side (right or left), and connected to three buses: the onside primary, the onside backup and the cross-side primary buses. Each NIC/PROC module and NIM transmits data on the onside buses at the same time.
Each NIC/PROC module and NIM receives data from one of the onside buses (through a switching algorithm), and also from the cross-side bus.



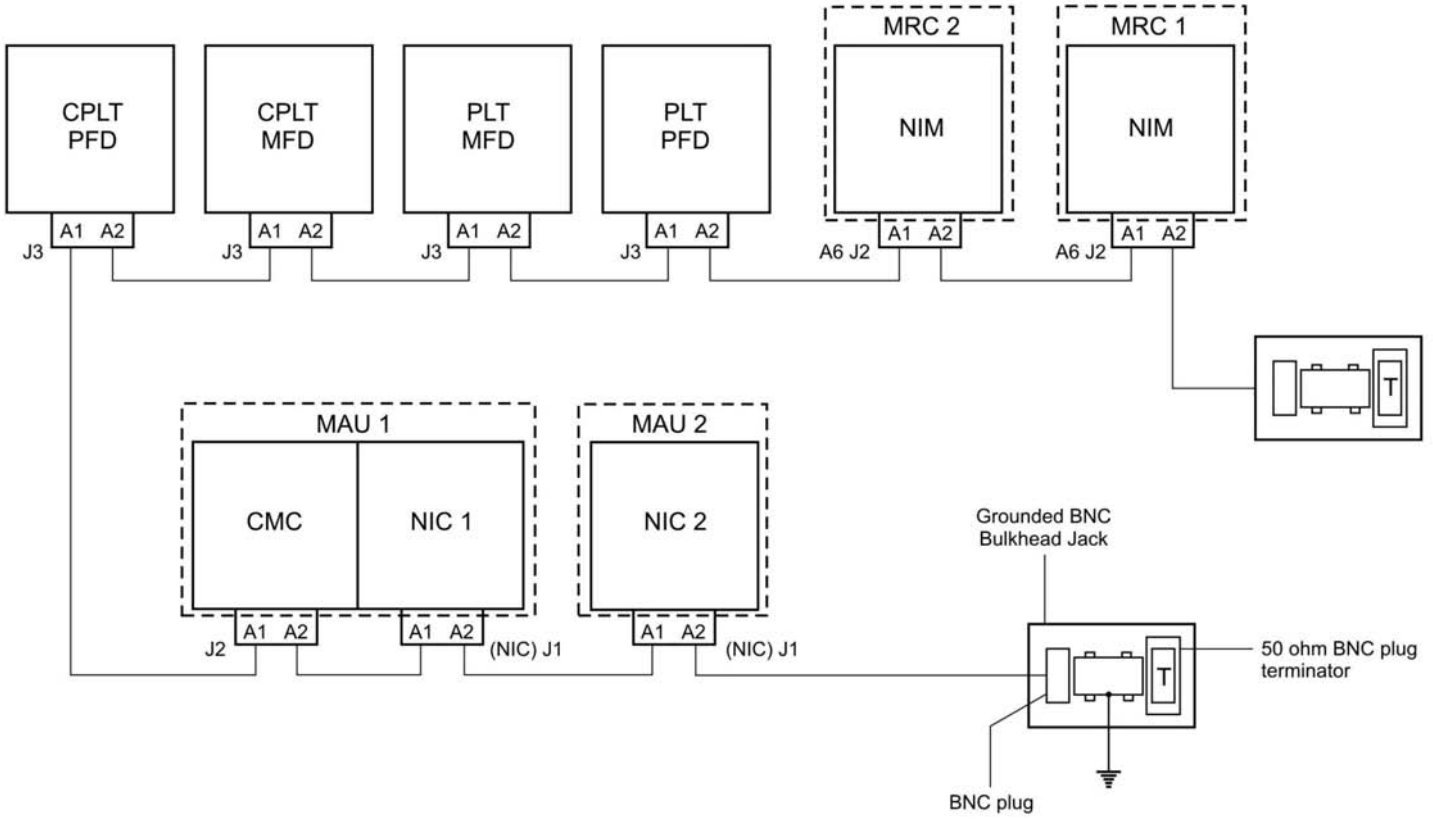
ASCB	Avionic Standard Communication Bus
NIC	Network Interface Controller
NIM	Network Interface Module
RX	Receive
TX	Transmit

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SYSTEM NETWORK BUS INTERFACES

LAN DATA BUS

The LAN is available to all units that connect to the ASCB-D. The LAN is a high-speed, non-essential bus without redundancy that is used for the maintenance and test functions. It is also used to upload the data and software in the MAU modules and the NIMs of the MRCs. The maintenance is done on the ground by connection of a standard laptop computer to the LAN. The configuration of the laptop computer must be that of a remote maintenance terminal (RMT) to the LAN: Ethernet 10-Base-2 hardware and proprietary software. The maintenance terminal and its CD or 3.5 inch disk drive are used to upload/download the data. Database files and/or software are uploaded to the system. Stored data (memory) or the fault history files are downloaded.
The LAN is connected to the same NIC/PROC as the ASCB-D. Each NIC/PROC includes a hardware LAN connector. All MAU modules transmit and receive on the LAN through the backplane bus. The NIC/PROC operates as an interface between the virtual LAN and the LAN hardware. The LAN hardware is a thin coaxial cable connected point-to-point to the NIC/PROC, NIM and CMC modules. The LAN is continuous internally to the same modules.

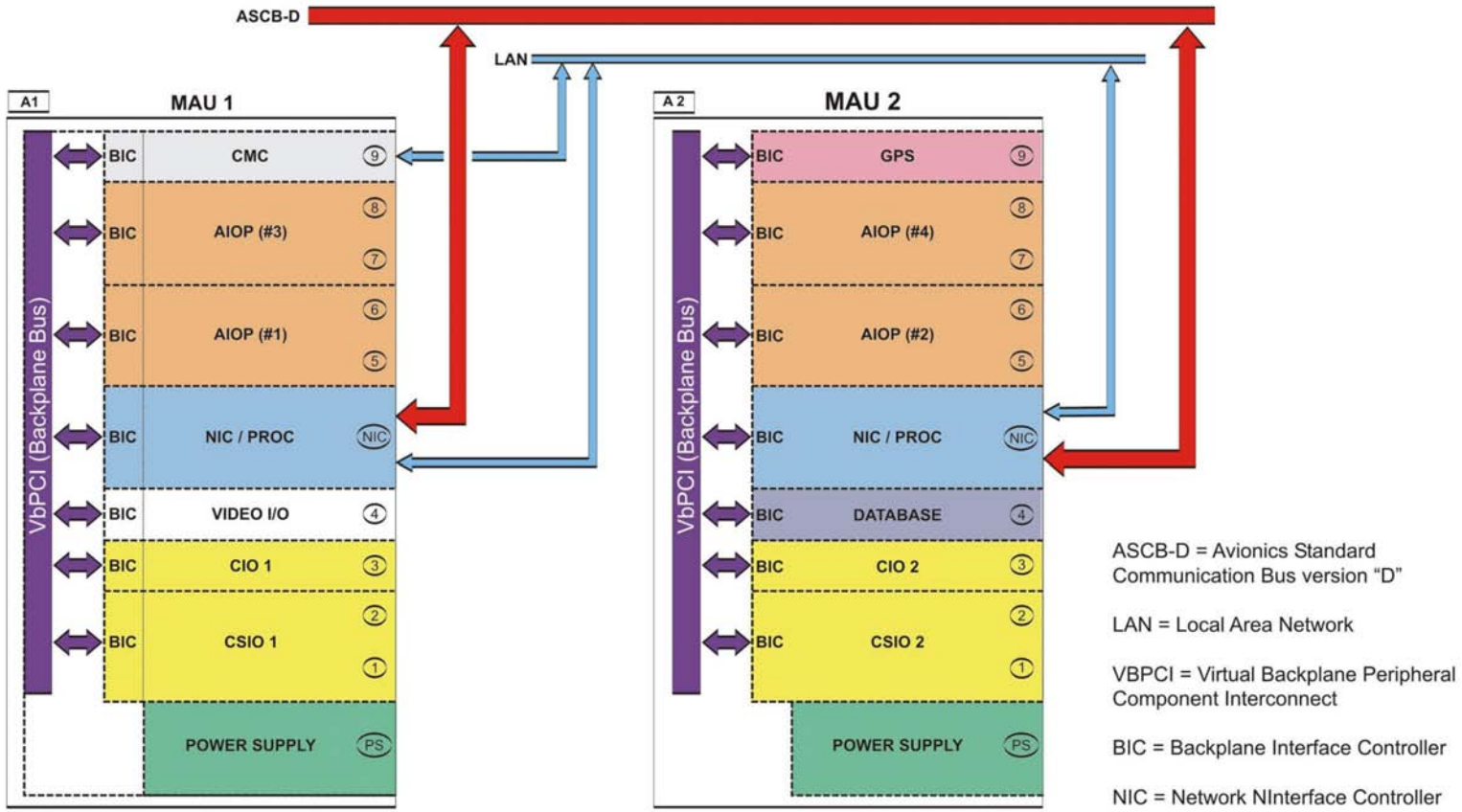


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LOCAL AREA NETWORK (LAN) DATA BUS

VBPCI DATA BUS

The VbPCI is a parallel bus that operates at high-speed and transmits in two directions. It connects the NIC/PROC to most MAU modules, but not to the power supply module and the GPS module. The bus structure is the same as that of the standard PCI bus you have in personal computers. Each channel of an MAU has a VbPCI that operates independently. The NIC/PROC modules control the transmission of data on the VbPCI. These NIC/PROC modules operate as the VbPCI masters.
A standard hardware interface the name of which is BIC gives the connection to the backplane of the MAU VbPCI. The BIC is installed on the client modules. It lets the NIC/PROC move the data received from the ASCB-D and LAN to the memory of the client module. It also lets the NIC/PROC retrieve the ASCB-D and LAN data the client module wants to transmit. The hardware mechanism that does these operations is the BIC frame buffer. The BIC frame buffer is a dual-port RAM that can be read and written to by the NIC/PROC and the client module.



ICN-39-A-460000-G-A0126-00578-A-01-1

VIRTUAL BACKPLANE PERIPHERAL COMPONENT INTERCONNECT (VBPCI) DATA BUS

REMOTE IMAGE BUS (RIB)

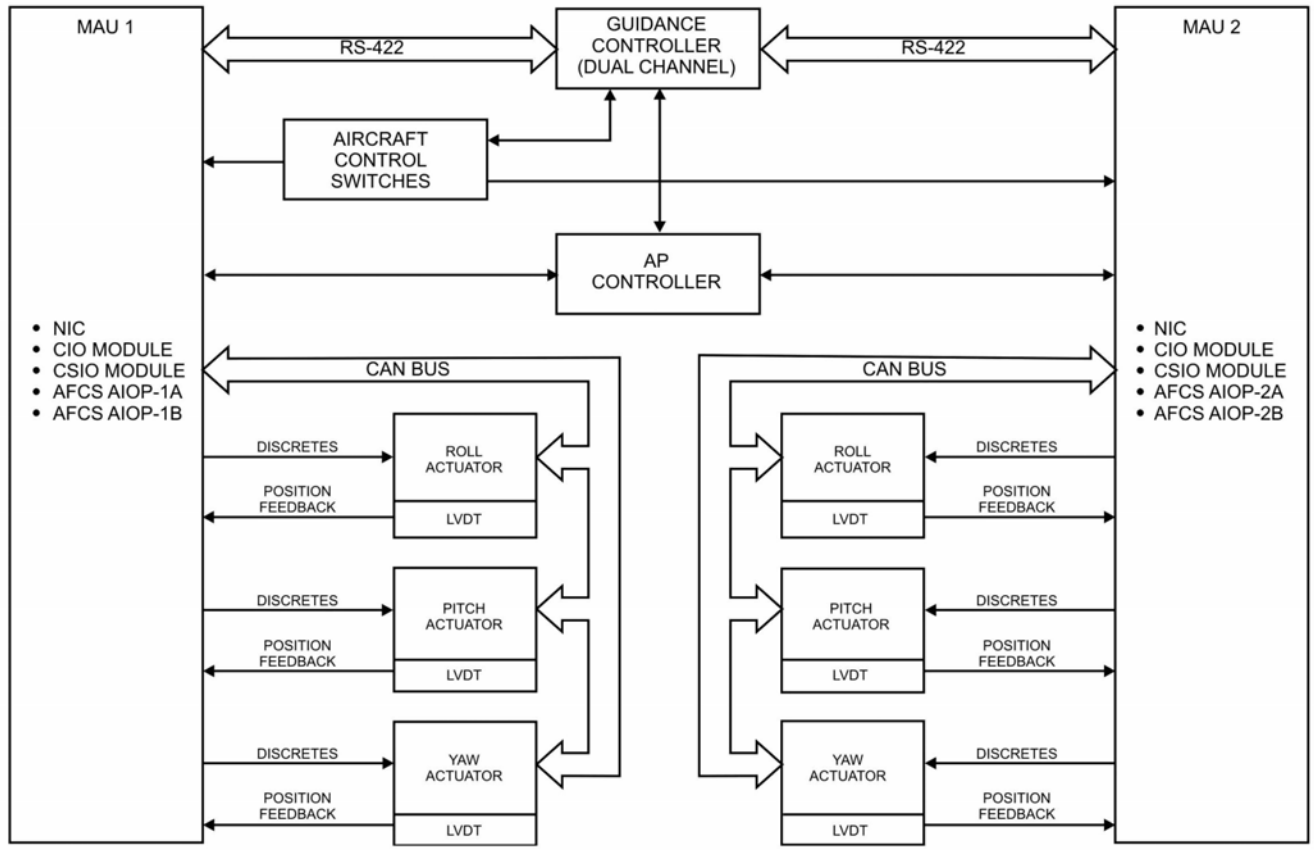
The video data are transmitted on the RIB from the video I/O module to the display units in the CDS. The RIB is a point-to-point bus that transmits in one direction. It is a wire bundle contained in a sheath and made of four sets of two twisted wires that have a shield.

CAN DATA BUS

The data are transmitted to and from the AIOP modules in the MAUs and the linear actuators through the CAN bus. The CAN bus is the interface between each MAU and the related triaxial set of series actuators. The NIC/PROC puts all data transmitted to and from the MAU modules on the ASCB-D thus the data can be available to the other systems.
The CAN bus is of the CAN-industry standard type and uses controller integrated circuits that operate at 500 kHz. The CAN bus transmits in two directions and has the same type of wire and harness as the ASCB-D.

WXPD BUS

The video data from the WXR system are transmitted on the WXPD bus. The WXR system has two buses that transmit in one direction only. Their names are: left WXPD bus and right WXPD bus. Each bus connects to the display unit in the CDS. The WXPD transmits weather-radar picture-data as serial Manchester II encoded-data at a rate of 1 MHz. Data transmission occurs at 8.33 +/-0.050 ms intervals.



ICN-39-A-460000-G-A0126-00579-A-01-1

CAN CONTROL AREA NETWORK DATA BUS

DIGITAL AUDIO BUS

The radio audio signals are transmitted from the MRC through the digital audio buses to the audio panels. Each MRC has one bus output and each audio panel has two bus inputs, one for each MRC. Each digital audio bus is a high-speed serial bus that transmits in one direction. It multiplexes multiple audio channels to two twisted wires that have a shield. The audio signals are made digital in the MRCs for all internal and external radios. The signals are then changed back to analog in the audio panels. The NIM in each MRC controls the transmission and reception of data on the digital audio bus for that MRC.

DIGITAL MICROPHONE BUS

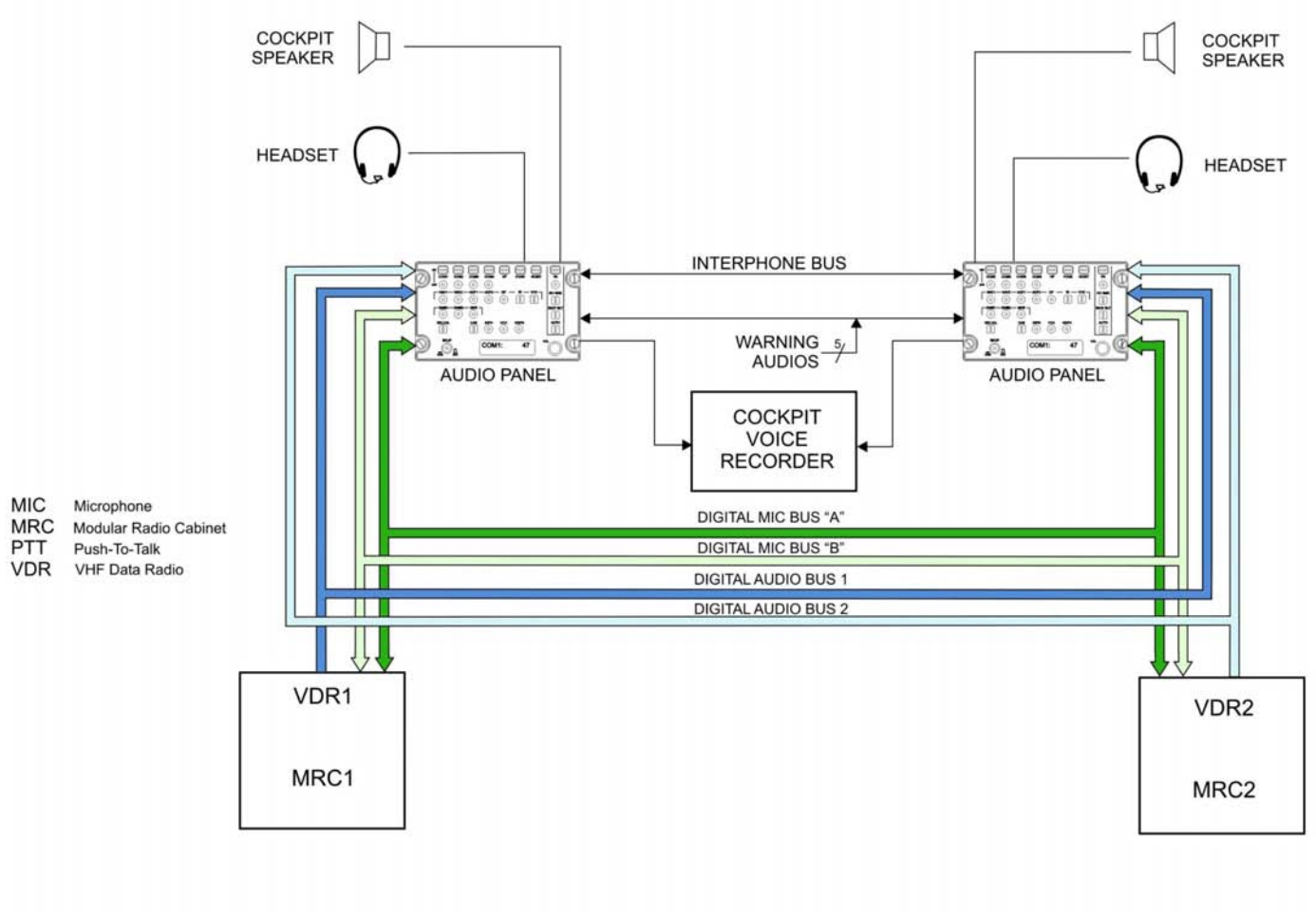
There are two digital microphone buses in the system. They transmit the audio signals from an audio panel to the MRCs and the other audio panels. Each digital microphone bus is a high-speed serial bus that transmits in two directions. It multiplexes the many microphone channels to one set of two twisted wires that have a shield. The microphone channels are supplied for one pilot's microphone (MIC) (mask or boom) and two maintenance MICs for each audio panel. The maximum permitted number of audio panels is eight. The audio signals are made digital in the audio panels. There they are changed back to analog for the interphones and the MRCs for radio transmissions. The MRCs supply analog MIC outputs for the external radios, if installed. The audio panels control the transmission and reception of data on the digital microphone buses.

RCB

The RCB connects the NIM in the MRCs to the internal modules of the MRCs. Each module receives and transmits through the serial receive (RX) RCB data bus and the serial transmit (TX) data bus to the NIM. The serial-receive data bus and the serial-transmit data bus are not connected to each other.
The RCB also connects an optional receiver of the microwave landing system to the NIM.

Each RCB bus is a set of two twisted wires with a shield for each module. One set of wires is for reception, the other for transmission.
The data rates are:

ADF module:	187.5 kBaud
VIDL module:	187.5 kBaud
VDR module:	200 kBaud
ATC XPDR module:	200 kBaud
DME module:	375 kBaud



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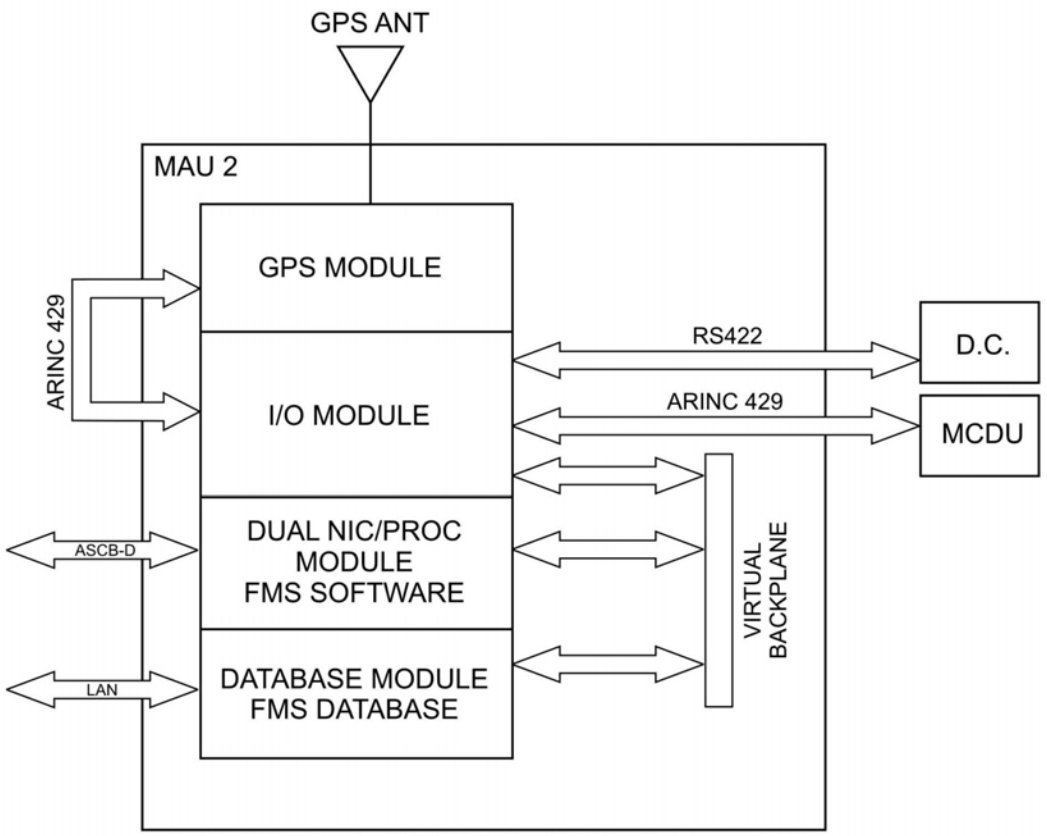
AUDIO AND MICROPHONE DATA BUSES

ARINC 429 DATA BUS

The ARINC 429 data bus system is made of transmitters and receivers connected through sets of two twisted wires that have a shield. The data are transmitted by one transmitter to one receiver or to a group of as many as 20 receivers connected in parallel. Each ARINC 429 bus transmits the data in one direction only. Two sets of transmitters, receivers and two twisted wires with a shield are necessary to transmit the data in two directions (to and from two units or modules).
The ARINC 429 data transmissions contain words of 32 bits. These words are transmitted at 12.5 kHz (low-speed) or 100 kHz (high-speed). Bit number 1 is always the first bit transmitted and bit number 32 is always the last bit transmitted. Bits 1 thru 8 are the octal label, which identifies the type of data contained in the word. Bits 9 and 10 usually are the source/destination identifier, which shows what source unit or module transmitted the word. Bits 9 and 10 can also be used as data bits in high-resolution data words. Bits 11 thru 29 are the data field. Bit 11 is the least significant bit and bit 29 is the most significant bit. Bits 30 and 31 usually are the sign-status matrix, which identifies the sign and validity of the data. Bits 30 and 31 can be used as data bits in high-resolution data words, as occurs with bits 9 and 10. Bit 32 is used for parity.

RS-422 CONTROL BUS

The RS-422 buses are made of sets of two twisted wires that have a shield. The controllers in the other avionics systems use the RS-422 buses to transmit data. These controllers include the guidance panel, display controller and the WX control bus (the MAUs supply this bus). The data transmitted on an RS-422 data bus are for control of the flight director. The display controller transmits RS-422 data for the manipulation of the display data on the PFD and MFD.



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ARINC 429 AND RS-422 DATA BUSES

CAPTIONS

The failure messages, relevant to the EPIC system, shown in the CAS messages window are as following:

CAS Message	Due to	Enabling Logic/Message Conditions
(1-2) MAU OVHT	MAU is overheating	Always enabled. When the MAU is overheating the NIC sets an over temperature bit on ASCB.
AVIONIC FAULT	ASCB BUS FAIL	Enabled on ground. Triggered by one or more NICs indicating same ASCB failure.
AVIONIC FAULT	APM MISCOMPARE	Always enabled. Triggered by APM Status = Miscompare OR Invalid.
AVIONIC FAULT	CMS FAIL	Always enabled. Triggered by 3-4 APMs reporting APM Status = Not Available. Lower level function messages should be inhibited.
AVIONIC FAULT	(1-2) DISP CNTL FAIL	Always enabled. The Control Function sets bit on ASCB to indicate no Display Controller activity.
AVIONIC FAULT	MAU FAN FAIL	Always enabled. Triggered by the failure of one or more MAU FANS.
AVIONIC FAULT	(1-2)MAU Fault	Always enabled. Triggered by CORE BIT detection of MAU module inactivity, invalidity of PROC Module functions, or Module BIT failures.

CAS Message	Due to	Enabling Logic/Message Conditions
AVIONIC FAULT	A429/RS422 BUS	Always enabled. Triggered by loss of communications to a single MAU for the following AHRS, CHIP Detector, EEC1, EEC2, Fuel Computer left and Fuel Computer right.
AVIONIC FAULT	MRC Fault	Always enabled. Triggered by CORE BIT detection of an MRC NIM module inactivity.
MAINTENANCE	CMC status	Enabled on-ground. Triggered when the parameter group output from CMC module is invalid or not fresh reference. Also, triggered when the CMC sets a bit on ASCB indicating maintenance type fault has occurred in the aircraft.
(1-2) MCDU OVHT	MCDU is overheating	Always enabled. Triggered when the MCDU temperature status bit indicates an over-temperature condition.
(1 2 3 4) DU DEGRADED	DU is in critical condition	Always enabled. Triggered by “critical wrap failure” from Monitor Warning function to displays. Lower level function messages should be inhibited
(1 2 3 4) DU OVHT	DU is overheating	Always enabled. Triggered by a DU Overheat condition. DU sets bit on ASCB indicating over-temperature status. Lower level function messages should be inhibited
(1-2) MRC OVHT	NIM is overheating	Always enabled. Triggered by over heat condition in radio NIM, NIM sets temperature status to over-temperature.
AWG FAIL	AW function not valid	Always enabled. MW monitors validity and freshness of ASCB transmissions from AW function.

CAS Message	Due to	Enabling Logic/Message Conditions
SYS CONFIG FAIL	System Configuration not valid	Enabled on Ground. Triggered after 5 second debounce by the failure of SCMS to validate system configuration.
VALIDATE CONFIG	HW/SW validity	Triggered by a change in EPIC HW and/or SW.

CHAPTER

52

DOORS

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SECTION 00 - DOORS

DOORS – GENERAL

Doors allow the crew and the passengers to enter or exit the aircraft. From the pilot point of view, the most important doors are:

- two cockpit doors (left and right)
- two cabin doors (left and right)
- one baggage door (located on the left side; the right baggage door is an optional)
- one nose door
- one external power door

The pilot can verify if the doors are fully closed and locked:

- by a direct visual inspection of the locking mechanism
- by monitoring the absence of caution messages on the CAS window of the MFD. The micro-switches are installed on each door allowing to check the position of the locking mechanism alerting the crew in case that the door are not closed.

COCKPIT DOORS

The cockpit doors provide access to both sides of the cockpit. The front side of the doors is hinged to the fuselage to allow them to be open freely. The other side of the doors is equipped with a handle.
Both cockpit doors can be opened from outside by turning the handle counter-clockwise and pulling the door; vice versa pilot and co-pilot doors can be opened from inside by pressing and turning the handle counter-clockwise and pushing the door.
An actuator installed in the upper part of the fuselage maintains the door opened.
The cockpit doors are provided with microswitches that generate a message of caution in case one door (left or right) is not closed.

CABIN DOORS

The cabin doors are located on the left and right sides of the center fuselage. Two rails with four rollers located two on the top and two on the bottom of the door allowing to slide the door backwards and to access to the cabin.
Cabin doors open from outside by pulling and rotating the lever and sliding the door backwards; vice versa to open from inside need to be pressed the safety pin, near the handle and sliding the door backwards.
The cabin doors are provided with microswitches that generate a message of caution in case one door (left or right) is not closed.

BAGGAGE DOOR

The baggage door is located on the left side of the rear fuselage and is hinged to the structure on the upper side A handle located at the bottom of the door allows to open the door. The right baggage door is furnished as an option.
The left baggage door is provided with a microswitch that generates a caution message in case the door is not properly closed. (Also on the right baggage door is installed a microswitch when the right baggage door is installed).

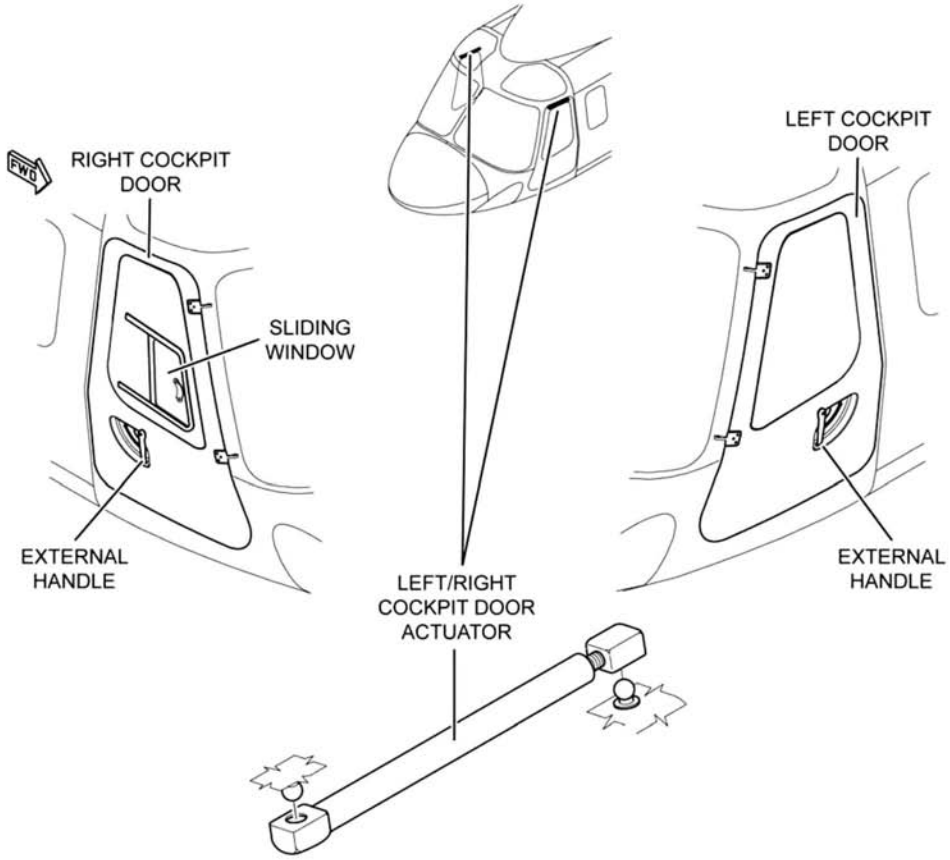
NOSE DOOR

The nose door provides access to the relevant compartment. The door is hinged to the fuselage on the bottom side. Centering pins installed on the upper side, secure the door when closed.
On the nose door no microswitches are installed.

EXTERNAL POWER DOOR

The external power receptacle is provided with a door that closes the access to the receptacle.
The door is provided with a microswitch that generates a caution message in case the door is not properly closed.





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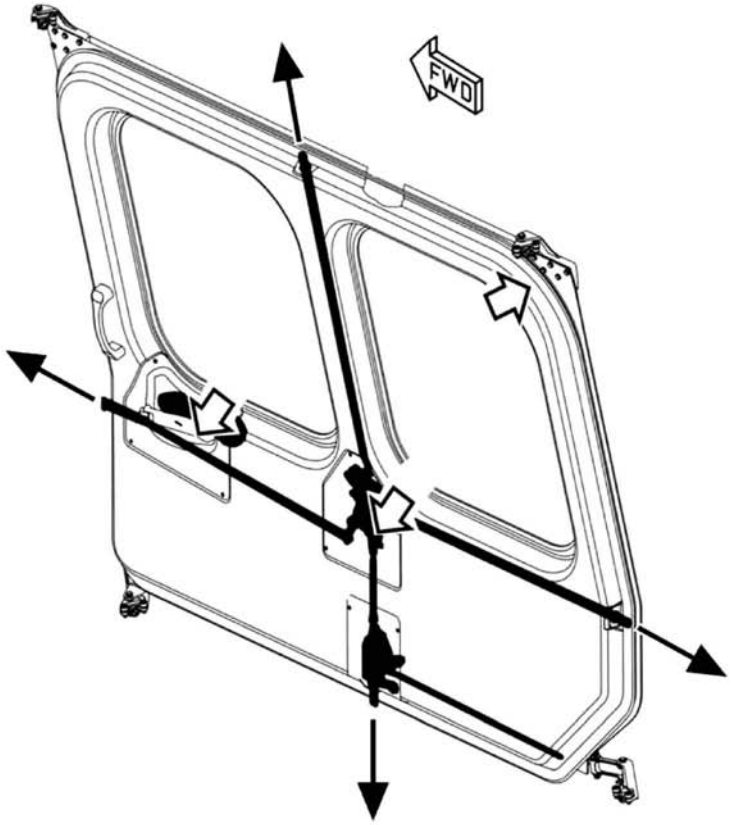
COCKPIT DOORS

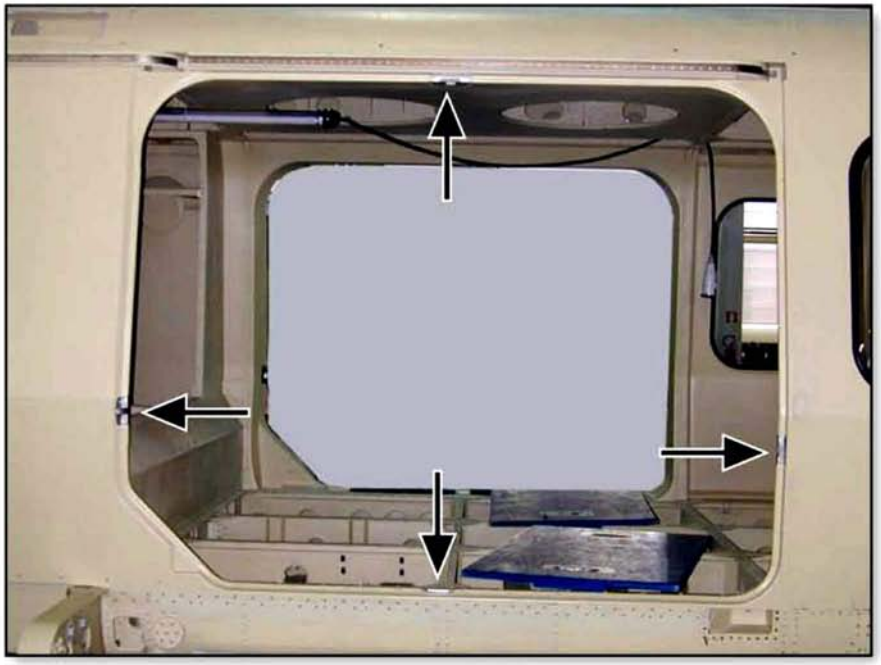




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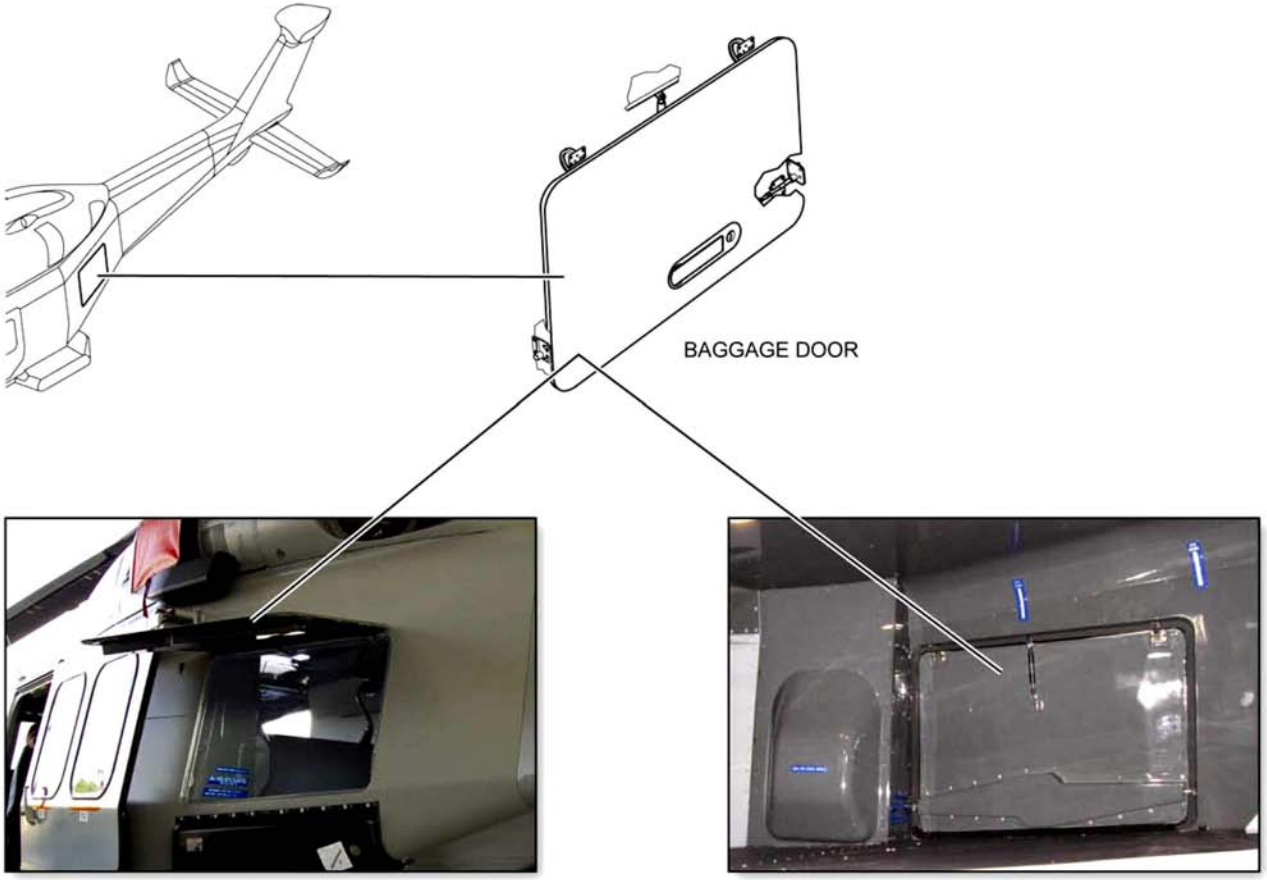
CABIN DOORS





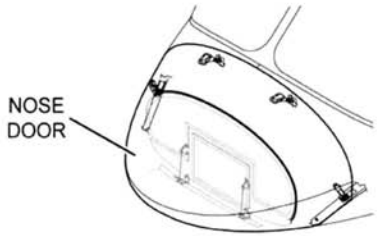
ICN-39-A-510000-G-A0126-02062-A-01-1

CABIN DOOR LOCKING

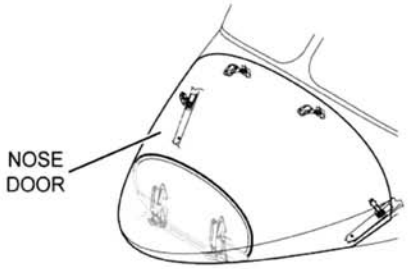


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BAGGAGE DOOR



SHORT NOSE CONFIGURATION



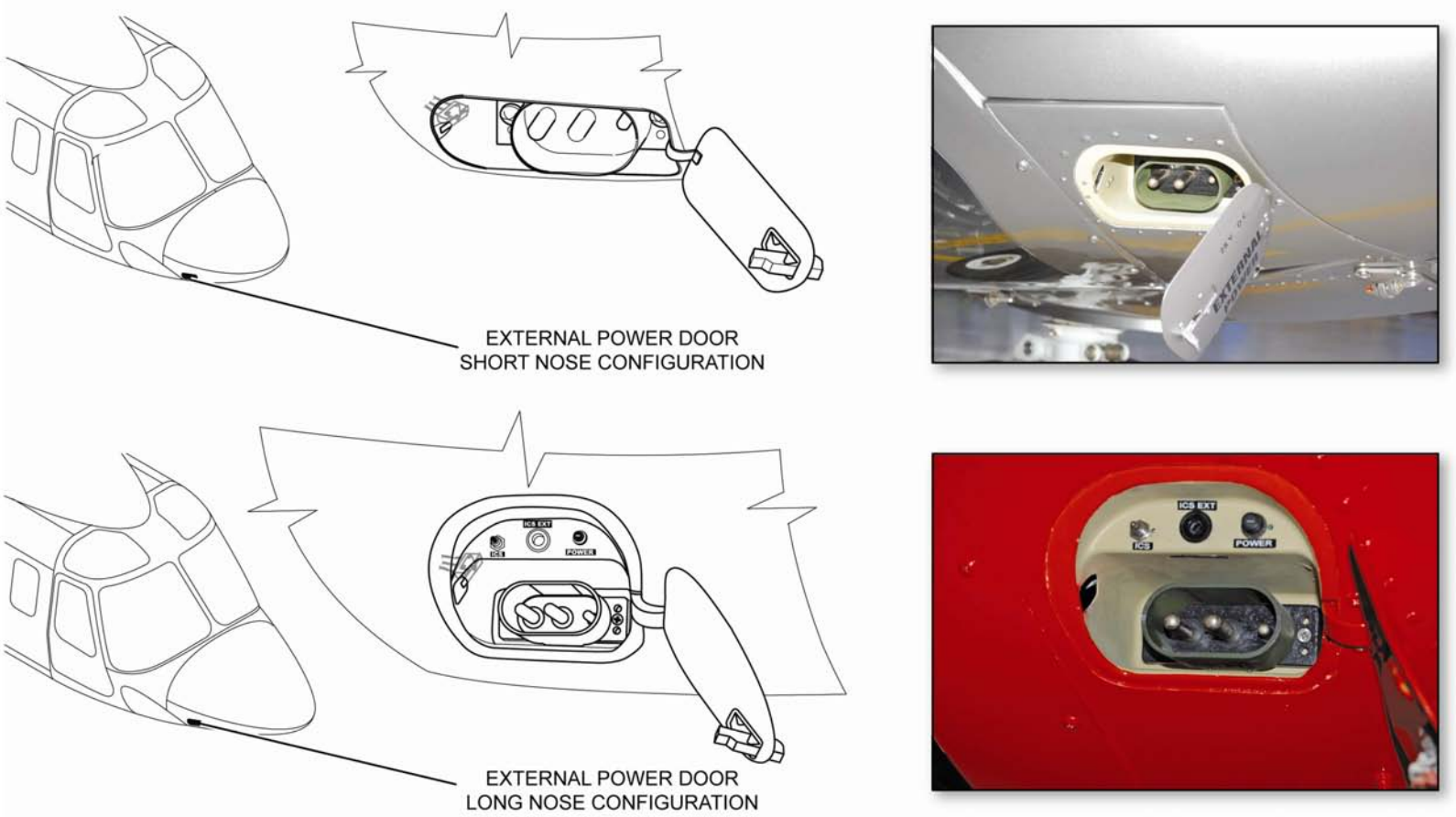
LONG NOSE CONFIGURATION





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NOSE DOOR



ICN-39-A-240000-G-A0126-01729-A-01-1

EXTERNAL POWER DOOR

SECTION 70 - DOOR WARNING

DESCRIPTION

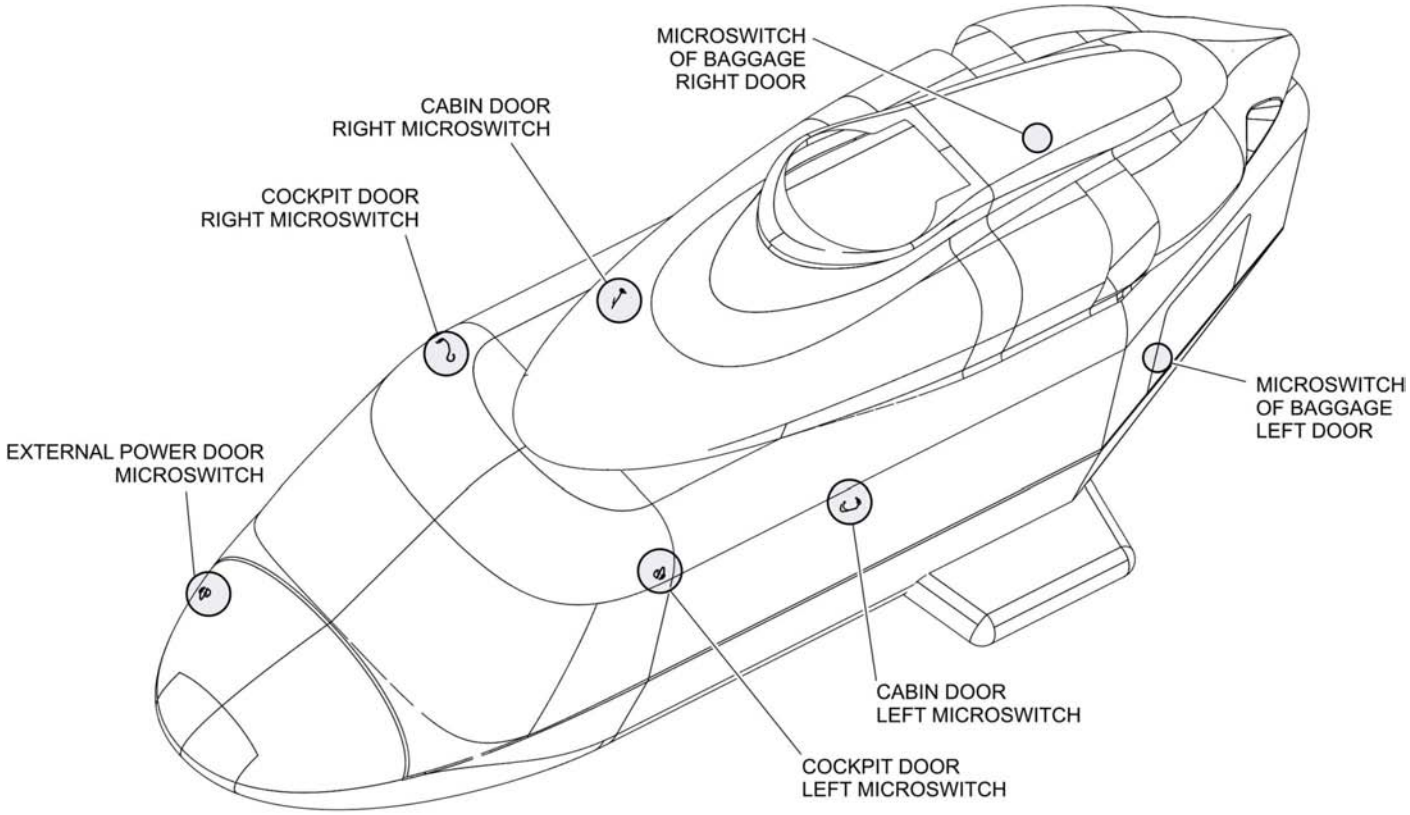
The helicopter is equipped with:

- One external power receptacle door.
- Two baggage doors.
- Two cockpit doors.
- One cabin doors.

There are two ways to determine if each door locking mechanism are fully locked.
One by direct visual inspection of the locking mechanism and the other by the visual indication, showed on the CAS display, for the crew member.
To comply this last way a microswitch is installed on each door to monitor the position of the locking mechanism and to warn the crew member if the door is not secured in the closed position.

OPERATION

On each door, a microswitch is fitted and connected electrically in series with the microswitch of the other door to send a discrete ground signal to the MAUs.
If a least one door is open, the MAUs detect an open signal and caution will be generated.



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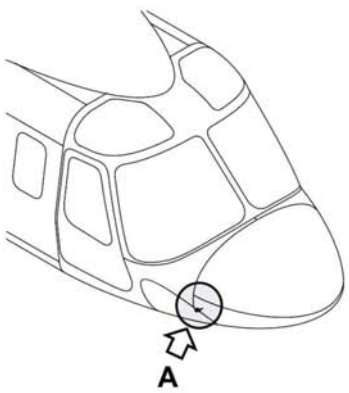
DOOR MICROSWITCHES INSTALLATION

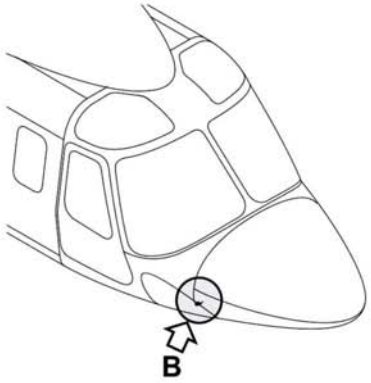
EXTERNAL POWER RECEPTACLE ACCESS DOOR WARNING INSTALLATION

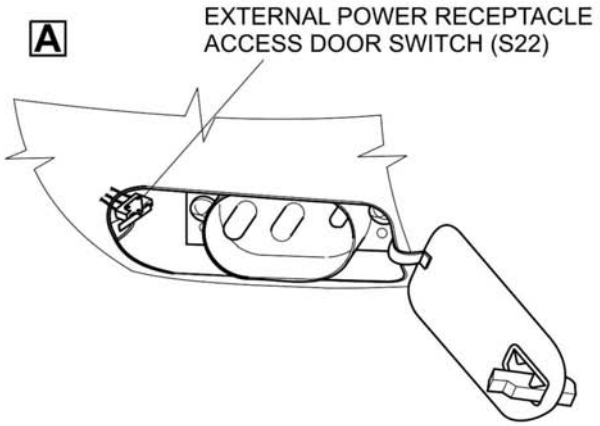
The primary components of the warning installation of the access door of the external power receptacle is the switch.

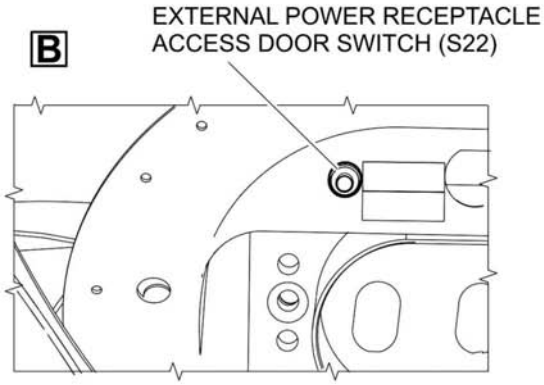
EXTERNAL POWER RECEPTACLE ACCESS DOOR SWITCH

The switch (S22) of the access door of the external power receptacle is installed in the housing of the external power receptacle. It is installed adjacent to the outer edge of the housing.
The access door is on the right side of the fuselage.
Two screws, two washers and two nuts attach the access door switch to the nose structure in the helicopters [1J]. Two nuts attach the access door switch to the support in the nose structure in the helicopters [1L]. Three electrical cables connect the switch to the ground and to the Number 1 Modular Avionic Unit (MAU1).
The access door switch monitors the position of the access door of the external power receptacle.
When the access door of the external power receptacle is open, or unlocked, the switch closes its contacts. In this condition, the switch sends a signal to the MAU1. Then the EXT PWR DOOR caution message comes in view on the Multifunction Flight Display (MFD) in the cockpit.









ICN-39-A-527000-G-A0126-00703-A-01-1

EXTERNAL POWER RECEPTACLE ACCESS DOOR WARNING INSTALLATION

BAGGAGE COMPARTMENT DOOR WARNING INSTALLATION

The primary components of the warning installation of the baggage compartment door are:

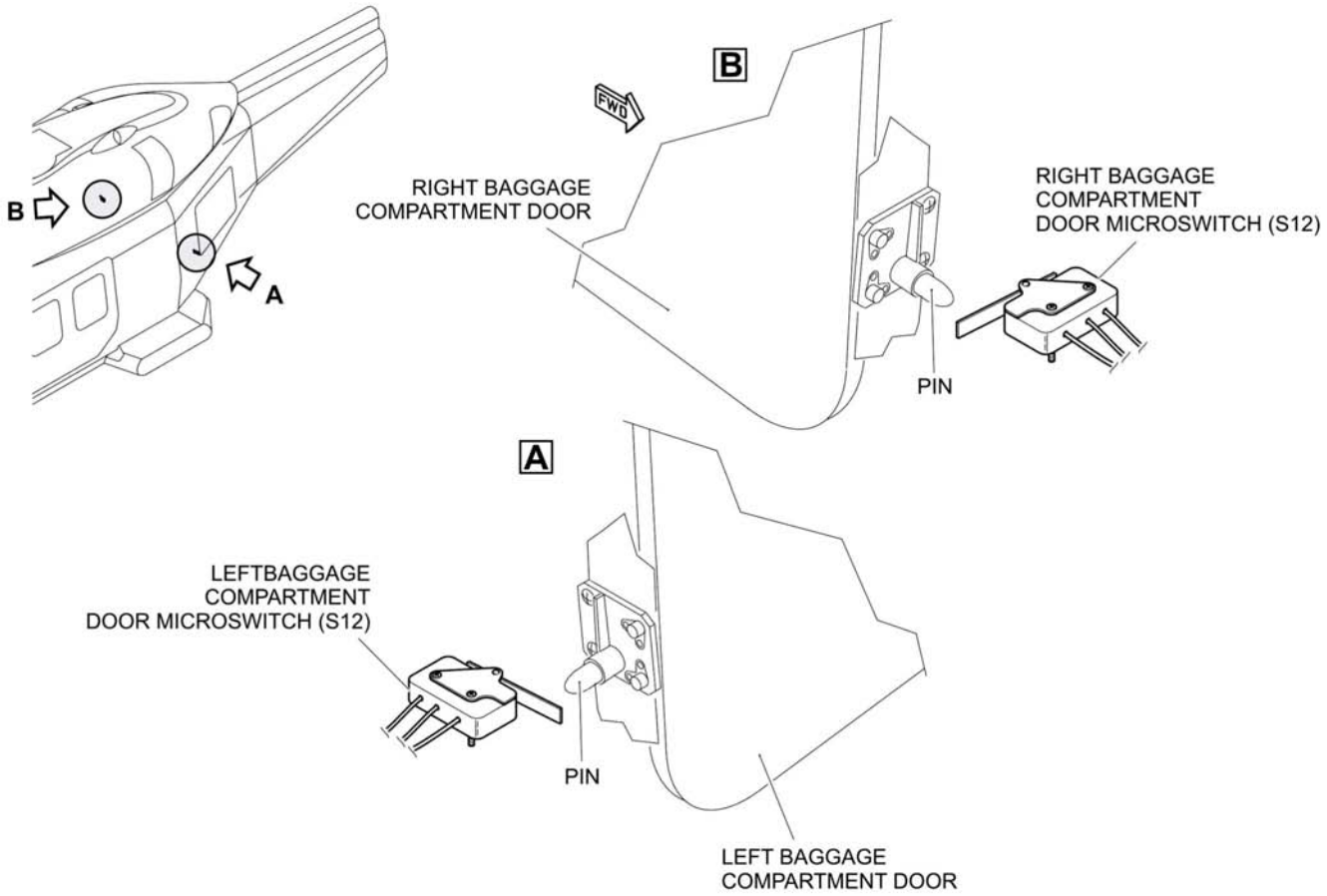
- The microswitch of the left baggage-compartment door.
- The microswitch of the right baggage-compartment door.

LEFT BAGGAGE COMPARTMENT DOOR MICROSWITCH

The microswitch (S37) of the left baggage-compartment door is installed on the forward outer edge of the door rabbet. Two screws attach the microswitch of the left baggage compartment door to the structure.
When the baggage-compartment access door is open, or unlocked, the microswitch closes its contacts. In this condition, the microswitch sends a signal to the Number 1 Modular Avionic Unit 1 (MAU1). Then the BAGGAGE DOOR OPEN 2 caution message comes on on the Multifunction Flight Display (MFD).
A pin is installed on the front of the left baggage-compartment door. When the door is closed and locked, the pin opens the contacts of the microswitch and the MFD does not show the related caution message.

RIGHT BAGGAGE COMPARTMENT DOOR MICROSWITCH

The microswitch (S12) of the right baggage-compartment door is installed on the forward outer edge of the door rabbet. Two screws attach the microswitch of the right baggage-compartment door to the structure.
When the baggage-compartment access door is open, or unlocked, the microswitch closes its contacts. In this condition, the microswitch sends a signal to the Modular Avionic Unit 2 (MAU2).
Then the BAGGAGE DOOR OPEN 2 caution message comes on on the Multifunction Flight Display (MFD) in the cockpit.
A pin is installed on the front of the right baggage-compartment door.
When the door is closed and locked, the pin opens the contacts of the microswitch and the MFD does not show the related caution message.



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BAGGAGE COMPARTMENT DOOR WARNING INSTALLATION

COCKPIT DOOR WARNING INSTALLATION

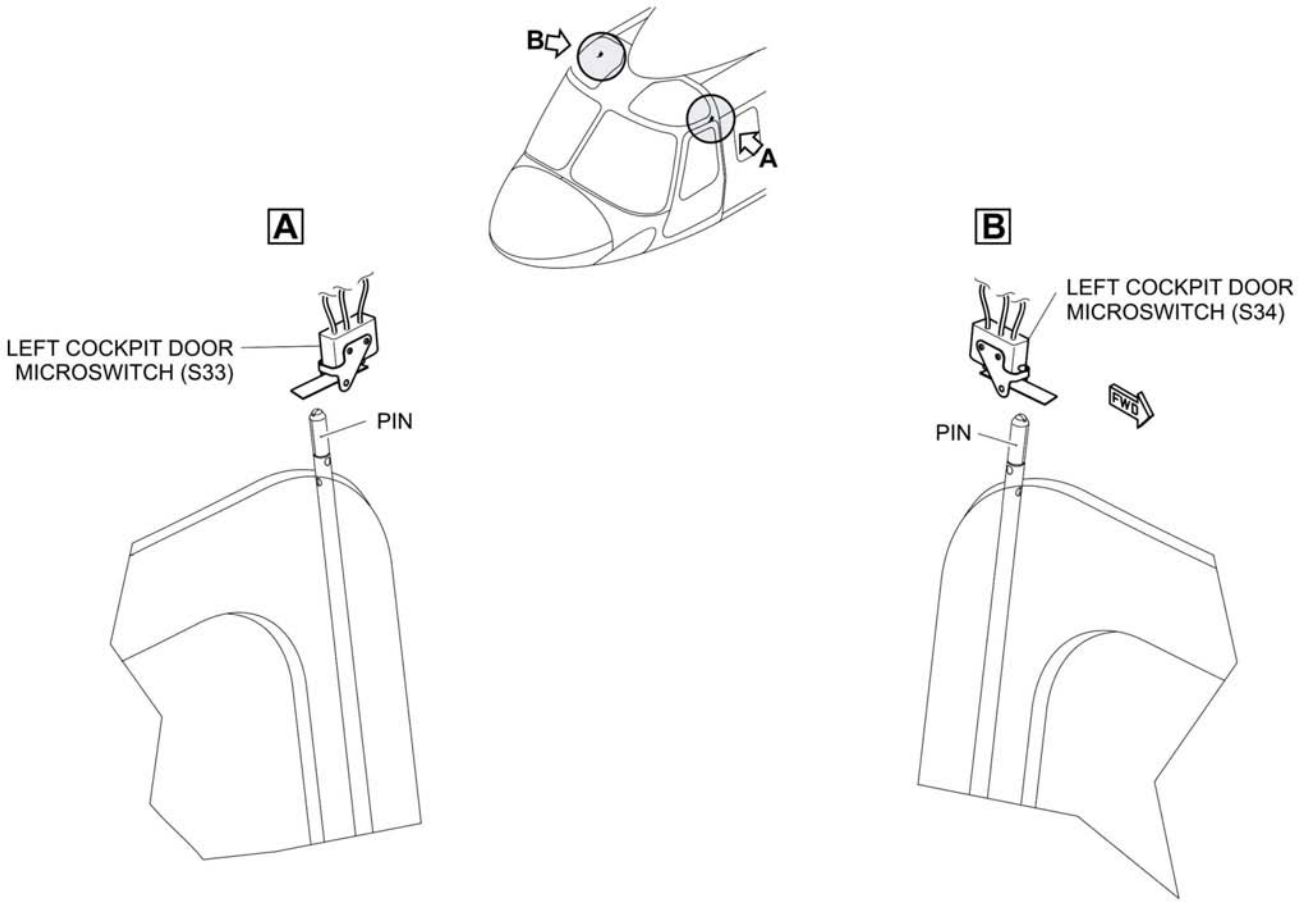
The primary components of the cockpit door warning installation are the left cockpit-door microswitch and the right cockpit-door microswitch.

LEFT COCKPIT DOOR MICROSWITCH

The left cockpit-door microswitch (S33) is installed on the left side of the cockpit ceiling adjacent to the edge of the left cockpit door.
Two screws attach the microswitch to the structure. Three electrical wires connect the left-cockpit door microswitch to the Number 1 Modular Avionic Unit (MAU1) and to the ground.
The left cockpit-door microswitch monitors the position of the left cockpit door. When the left cockpit door is open or not locked in the closed position the microswitch closes its contacts. Then the microswitch sends the related signal to the to the MAU1.
This causes the COCKPIT DOOR OPEN 1 caution message to come in view on the Multifunction Flight Display (MFD) in the cockpit.

RIGHT COCKPIT DOOR MICROSWITCH

The right cockpit-door microswitch (S34) is installed on the right side of the cockpit ceiling adjacent to the edge of the right cockpit door.
Two screws attach the microswitch to the structure. Three electrical wires connect the right cockpit door microswitch to the Number 2 Modular Avionic Unit (MAU2) and to the ground.
The right cockpit-door microswitch monitors the position of the right cockpit door.
When the right cockpit door is open or not locked in the closed position the microswitch closes its contacts. Then the microswitch sends the related signal to the MAU2.
This causes the COCKPIT DOOR OPEN 2 caution message to come in view on the Multifunction Flight Display (MFD) in the cockpit.



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COCKPIT DOOR WARNING INSTALLATION

CABIN DOOR WARNING INSTALLATION

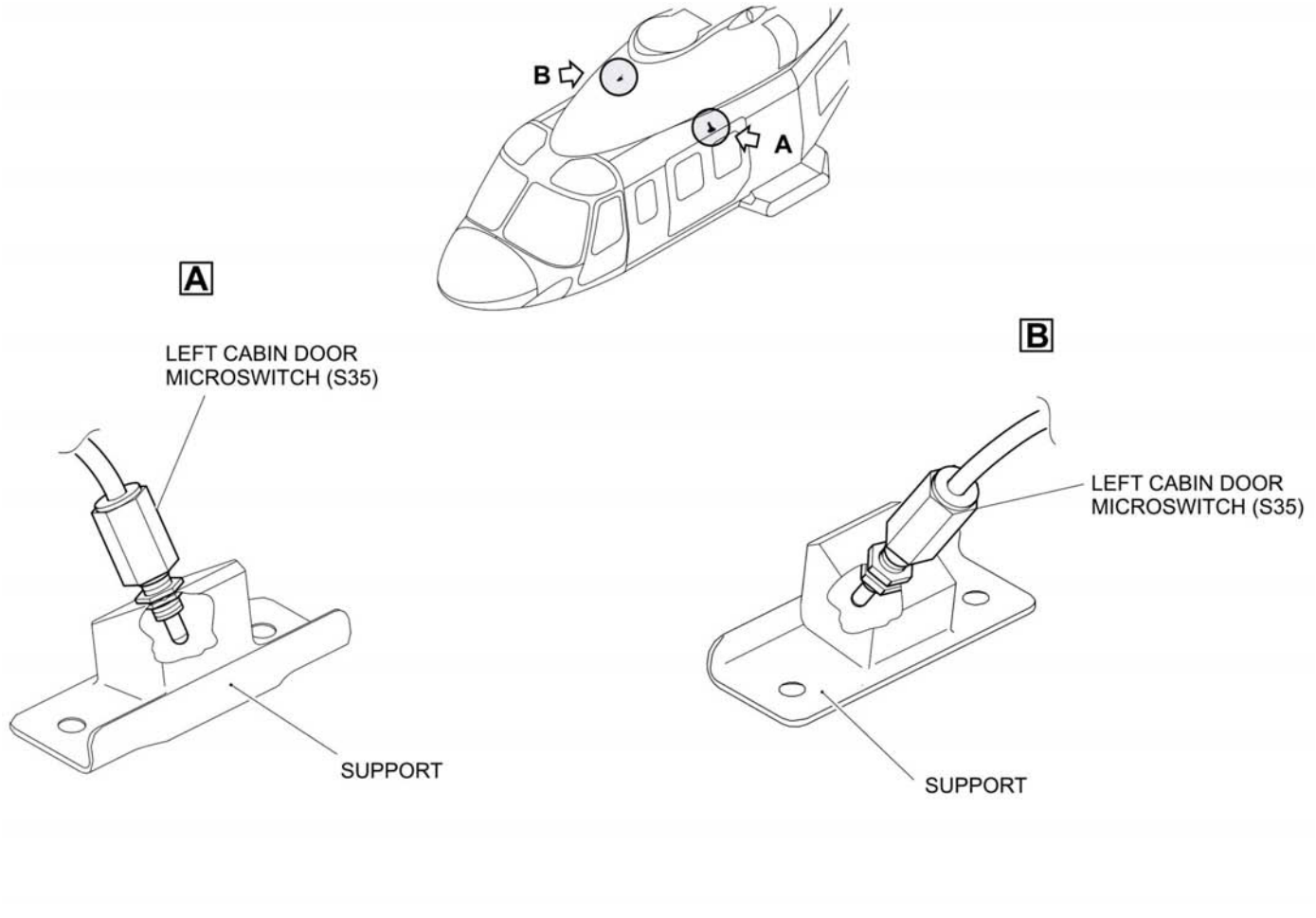
The primary components of the cabin door warning installation are the left cabin-door microswitch and the right cabin-door microswitch.

LEFT CABIN DOOR MICROSWITCH

The left cabin-door microswitch (S35) is installed on the left side of the cabin ceiling, at STA 4812. Its position is adjacent to the edge of the left cabin door. A nut attaches the left-cabin door microswitch to its support. Two electrical wires connect the left cabin-door microswitch to the Number 1 Modular Avionics Unit (MAU1) and to the ground.
The left cabin-door microswitch monitors the position of the left cabin door. When the left cabin door is open or not locked in the closed position the microswitch closes its contacts. Then the microswitch sends a signal to the MAU1. This causes the CABIN DOOR OPEN 1 caution message to come in view on the Multifunction Flight Display (MFD) in the cockpit.

RIGHT CABIN DOOR MICROSWITCH

The right cabin-door microswitch (S36) is installed on the right side of the cabin ceiling, at STA 4812. Its position is adjacent to the edge of the right cabin door. A nut attaches the right cabin-door microswitch to its support. Two electrical wires connect the left cabin-door microswitch to the Number 2 Modular Avionic Unit (MAU2) and to the ground.
The right cabin-door microswitch monitors the position of the right cabin door. When the right cabin door is open or not locked in the closed position the microswitch closes its contacts. Then the microswitch sends a signal to the MAU2. This causes the CABIN DOOR OPEN 2 caution message to come in view on the Multifunction Flight Display (MFD) in the cockpit.



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CABIN DOOR WARNING INSTALLATION

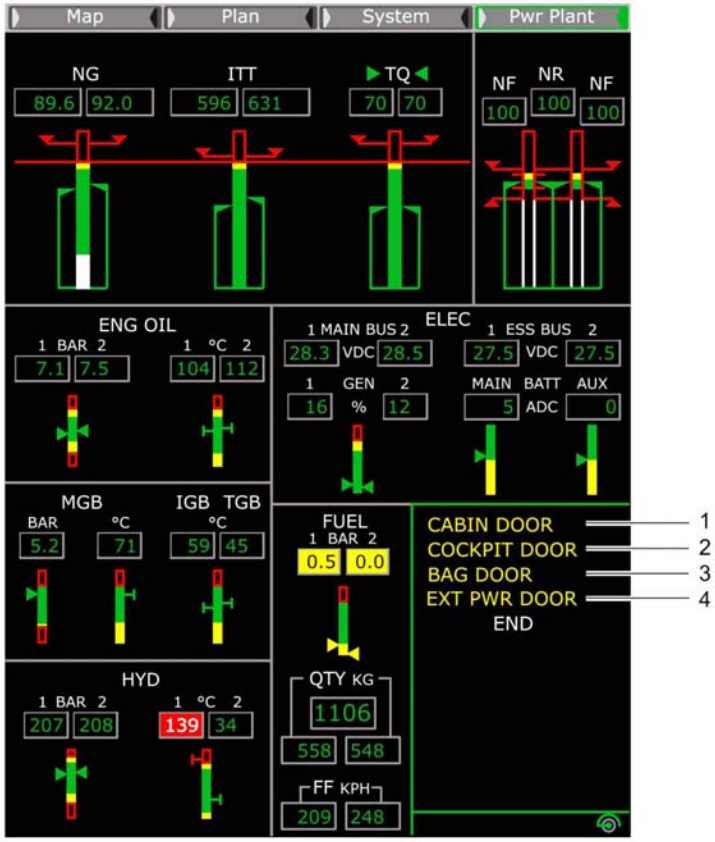
DOOR WARNING SYSTEM INDICATING

All the microswitches are installed on the fuselage.

CAPTIONS

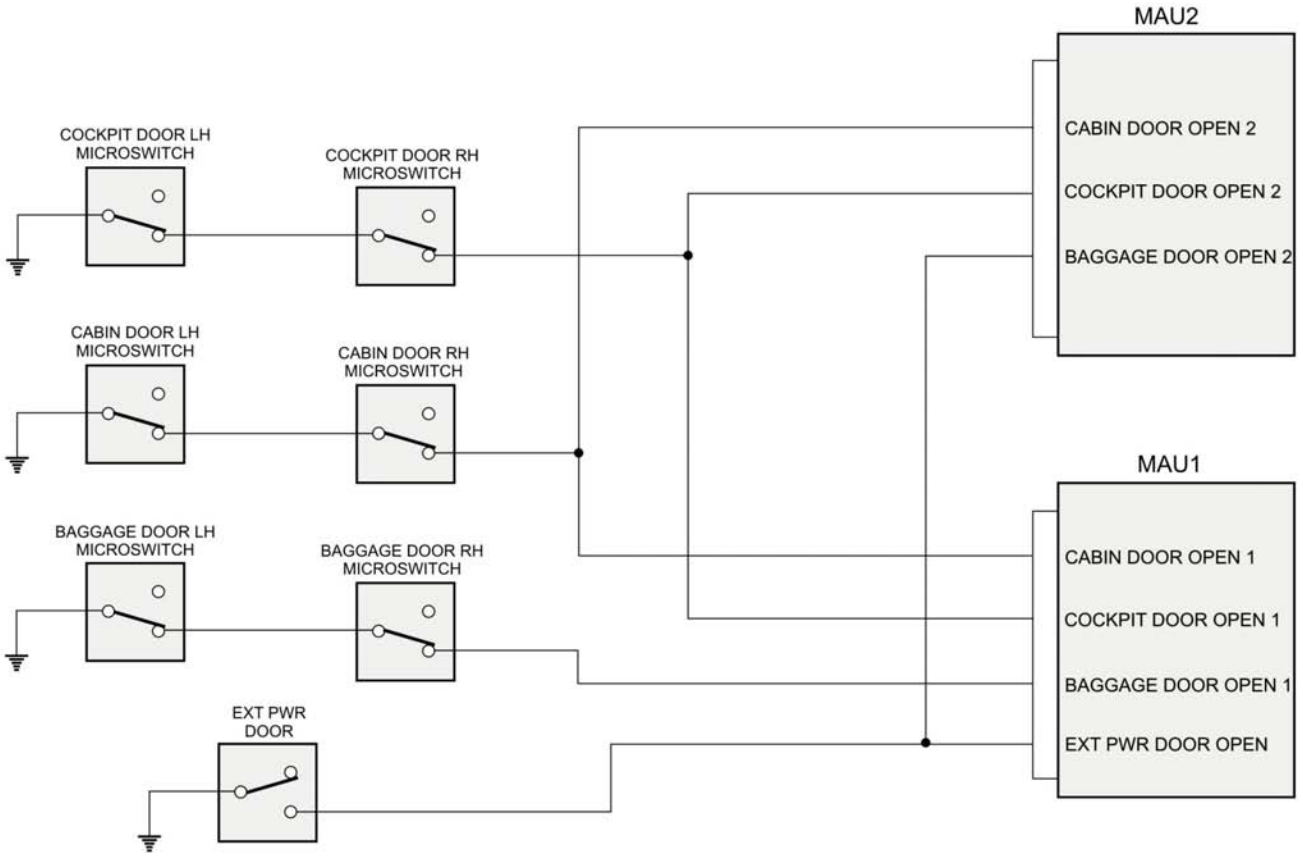
The caution messages provided in the CAS message window is as follows:

1	CABIN DOOR	when either cockpit door is not secured in the closed position.
2	COCKPIT DOOR	when either cabin door is not secured in the closed position.
3	BAG DOOR	when either baggage door is not secured in the closed position.
4	EXT PWR DOOR	when the external power access door is open or unlocked.



ICN-39-A-527000-G-A0126-00707-A-01-1

DOOR WARNING SYSTEM INDICATING



ICN-39-A-527000-G-A0126-00708-A-01-1

DOOR MICROSWITCHES ELECTRICAL SCHEMATIC

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CHAPTER

53

FUSELAGE

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SECTION 10 - FORWARD SECTIONS

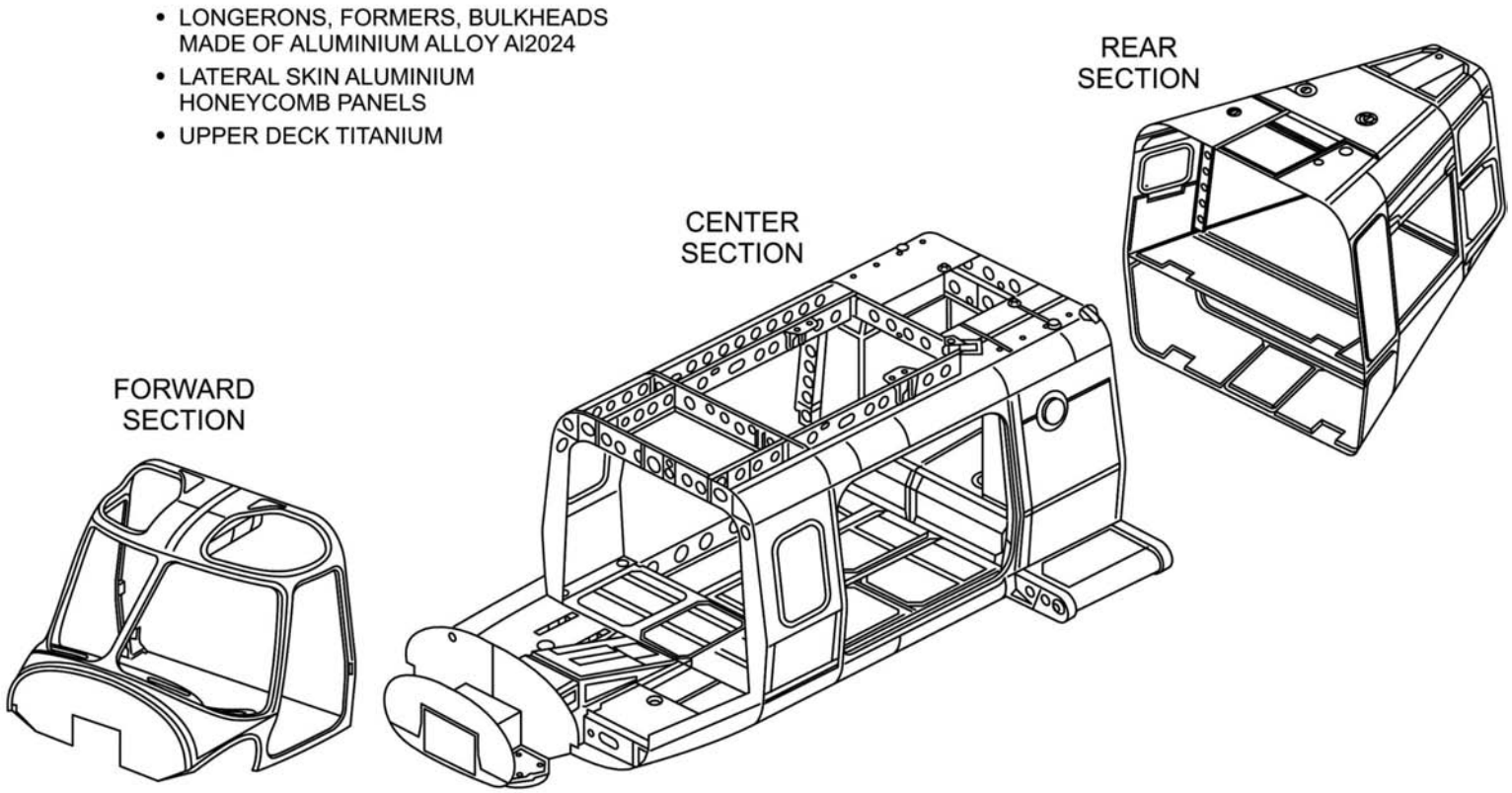
DESCRIPTION

The forward section is the front part of the primary structure.
The forward section is a metallic construction made with longerons, frames and sandwich panels; it mainly provides the pilots cabin floor, interface with cockpit, nose landing gear housing and supports the canopy assembly.
The forward fuselage is part of the whole primary structure which includes also the center and rear fuselage; subdivision is intended for convention only.
The forward section is interfaced with the following main systems:

- Instrument panel/overhead panel/interseat console
- Landing gear
- Electric power system
- Electric system
- Engine controls
- Hydraulic system
- ECS
- Flight controls
- Avionics
- Equipment and furnishing
- Doors
- Windows.

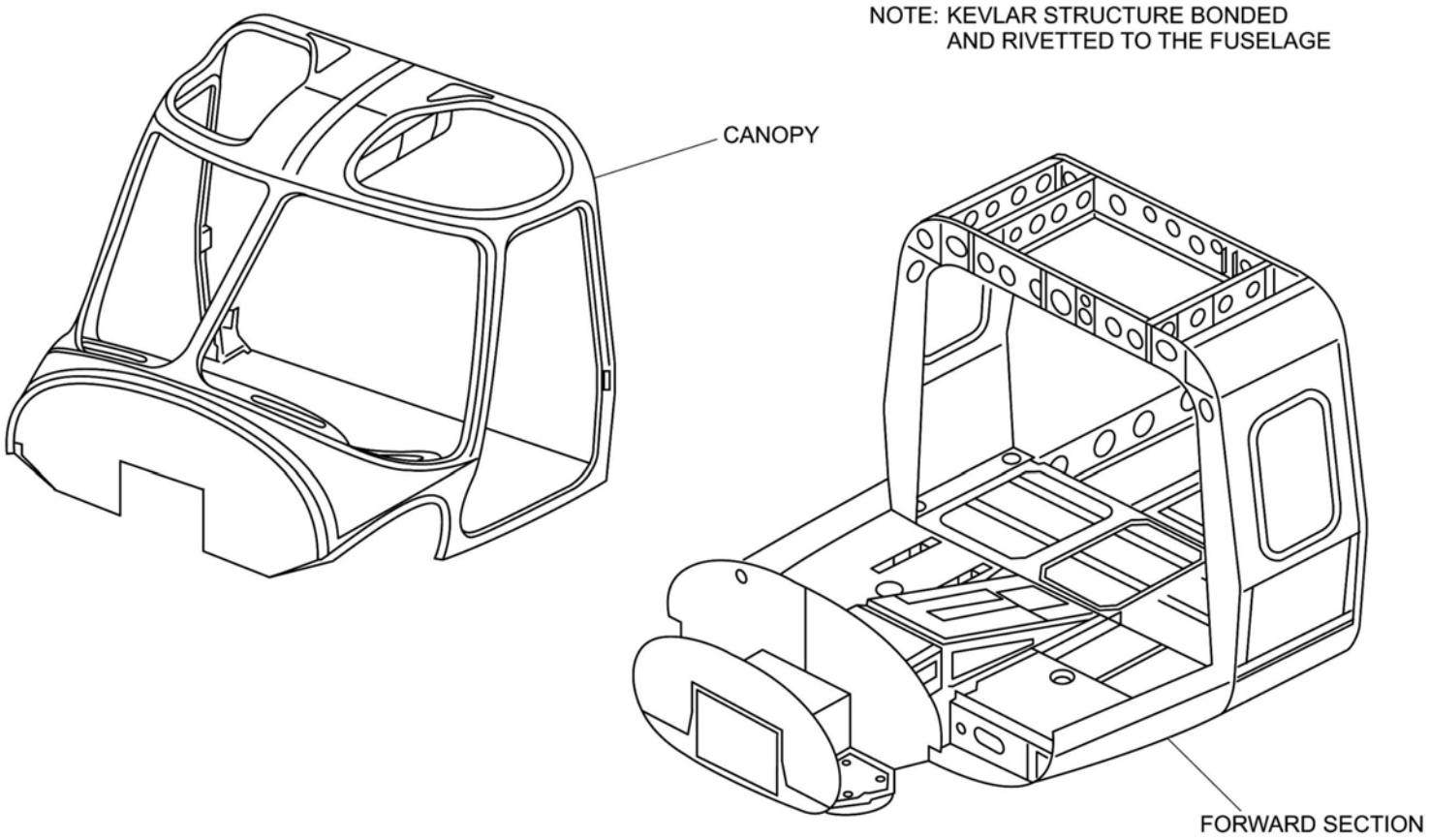
The forward section is designed to provide the required safety conditions defined by:

- Crash loads producing controlled deformations without jeopardize occupants escape.
- Natural frequencies out of main and tail rotor multiple frequencies.
- Avoid ground resonance.
- Sustain lighting effects.
- Impact damage tolerance.



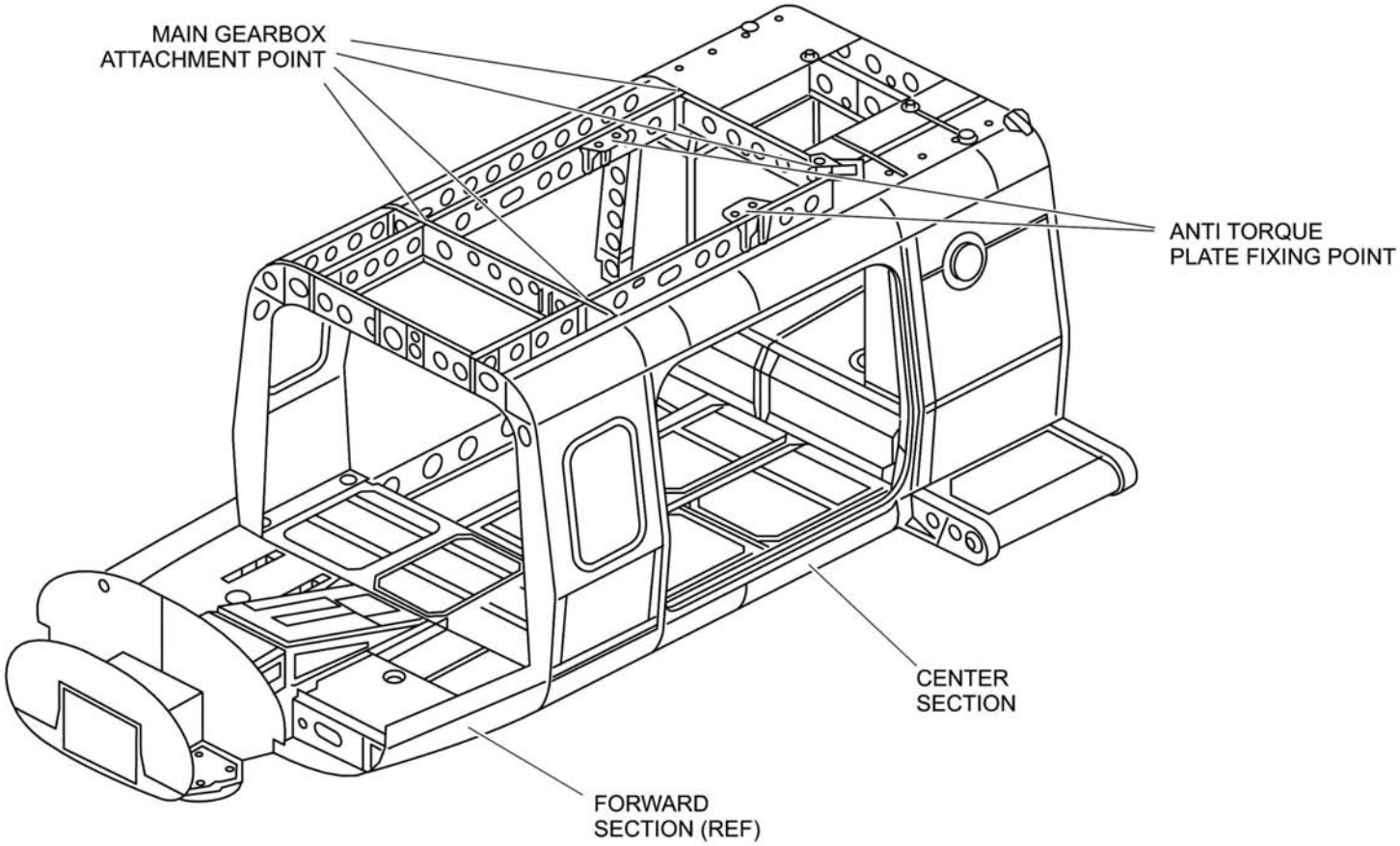
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FUSELAGE SECTION (SHEET 1 OF 5)



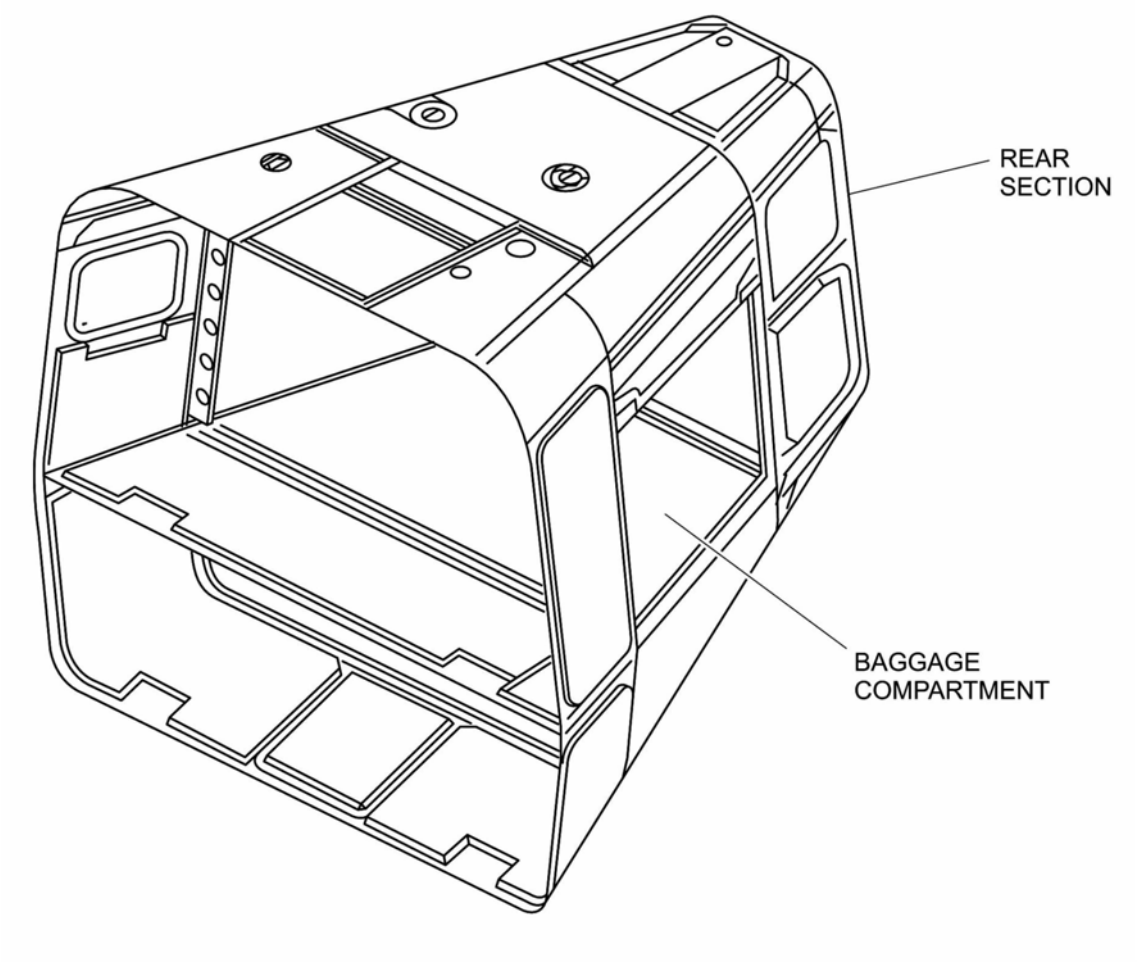
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FUSELAGE SECTION (SHEET 2 OF 5)



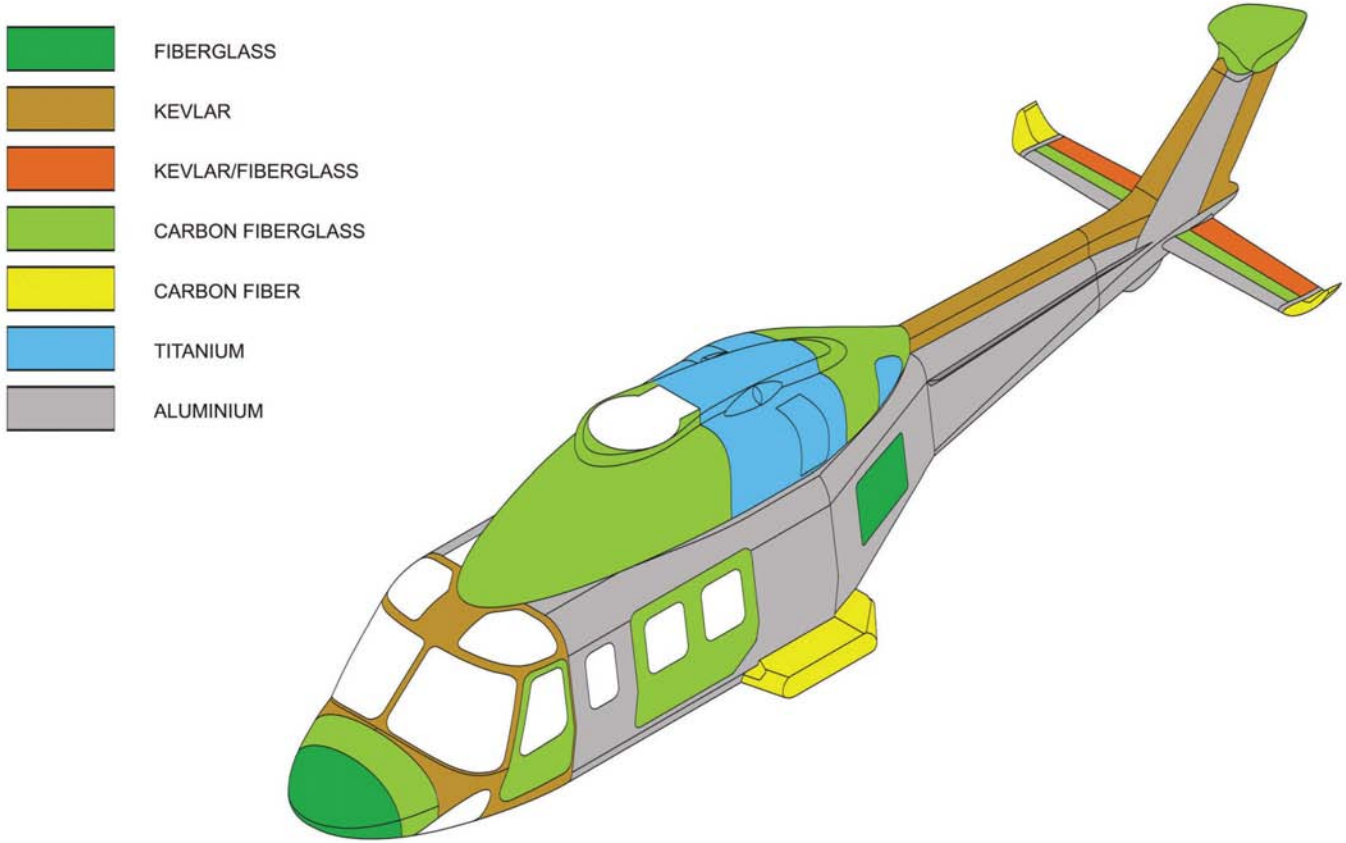
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FUSELAGE SECTION (SHEET 3 OF 5)



ICN-39-A-531000-G-A0126-00714-A-01-1

FUSELAGE SECTION (SHEET 4 OF 5)



ICN-39-A-531000-G-A0126-00715-A-01-1

FUSELAGE SECTION (SHEET 5 OF 5)

SECTION 40 - TAIL SECTION

DESCRIPTION

The Tail Section is a metallic semi-monocoque construction made with sheet, beams and frames; it mainly provides the Tail Rotor Drive, flight controls, Tail Rotor and stabilizers installation.
The tail unit is composed by a cone and a fin; subdivision is intended for convention only. The tail unit is completed with the tail rotor drive fairings and the fin leading and trailing edge manufactured in composite materials.
The tail unit is designed to provide the required safety conditions defined by:

- Crash loads producing controlled deformations.
- Natural frequencies out of main and tail rotors multiple frequencies.
- Avoid ground resonance.
- Sustain lighting effects.
- Impact damage tolerance.

MAINTENANCE INFORMATION

Before tail unit removal, it's required to remove:

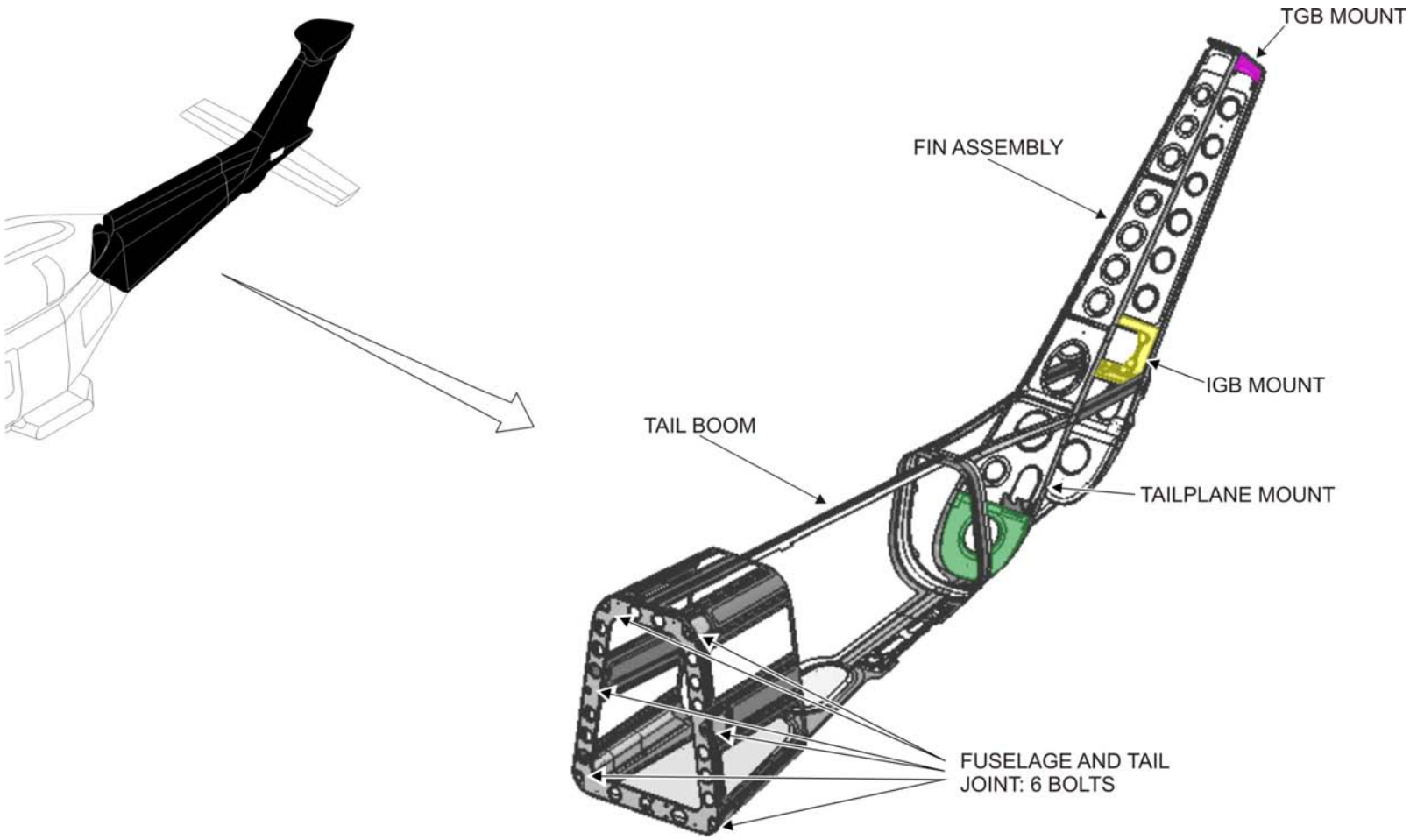
- Tail Rotor Drive Shaft 2.
- Tail Rotor.
- Yaw flight control rod Y7.
- PCM1 and PCM2 drained.

LEADING PARTICULARS

- Tail unit weight (only structure)..... 80.7 Kg (178 lb)

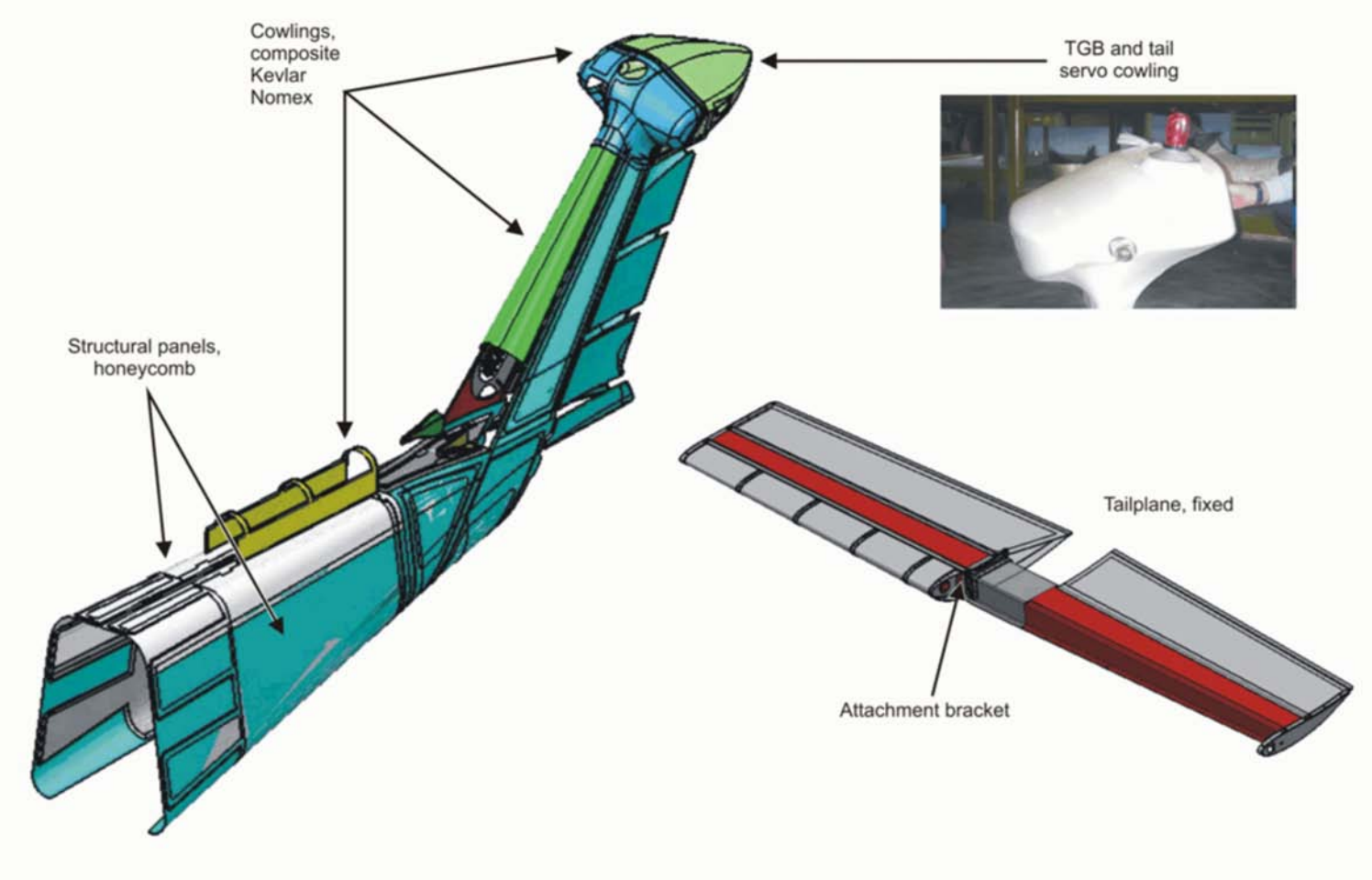
SAFETY PRECAUTION

Be careful when you remove / install the tail unit. The tail unit is heavy. An incorrect movement can cause an injury to persons and / or damage to the equipment. Get a second person to help you during removal /installation of the tail unit.



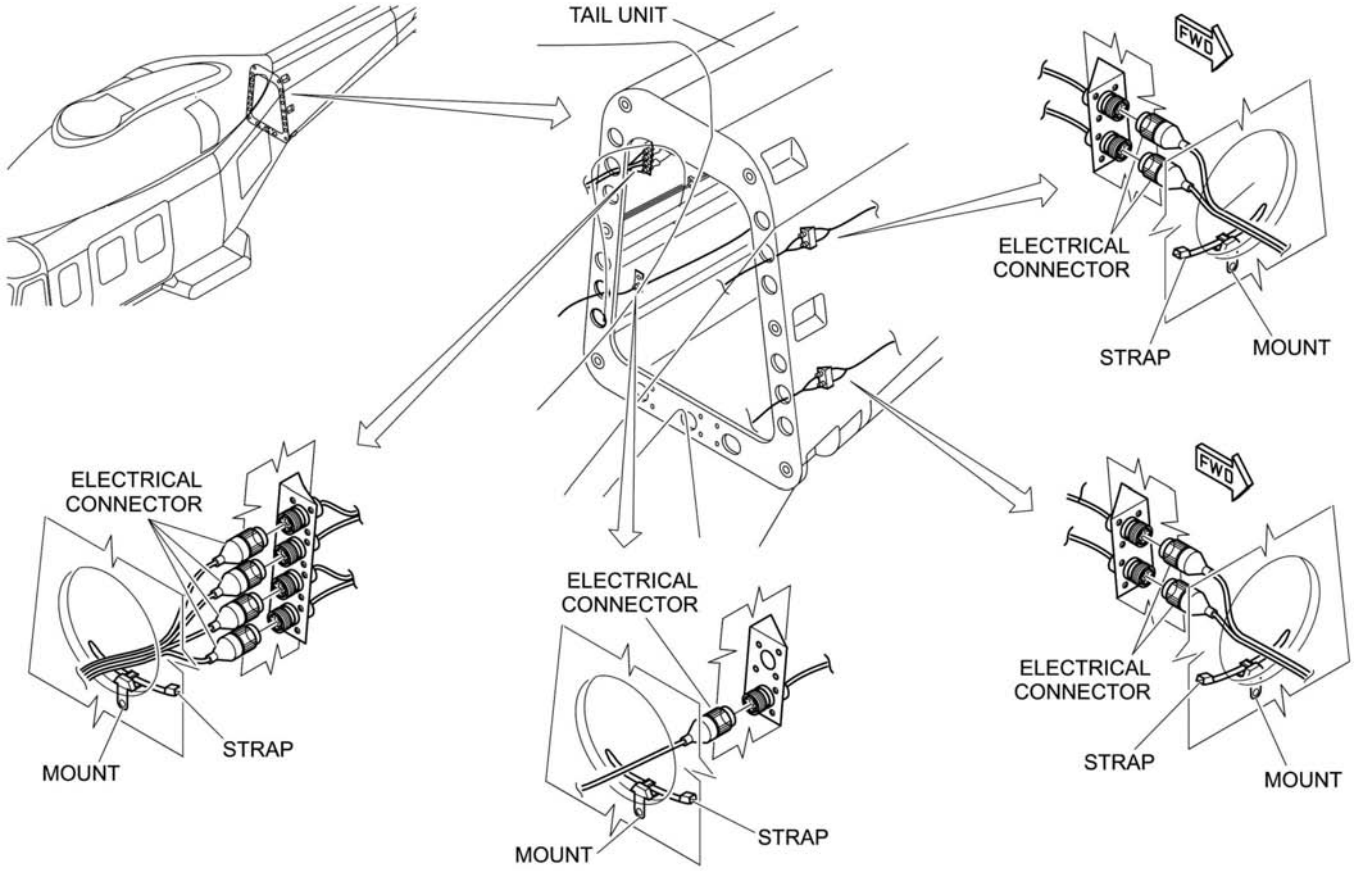
ICN-39-A-534000-G-A0126-00721-A-01-1

TAIL UNIT GENERAL LAYOUT



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TAIL SECTION

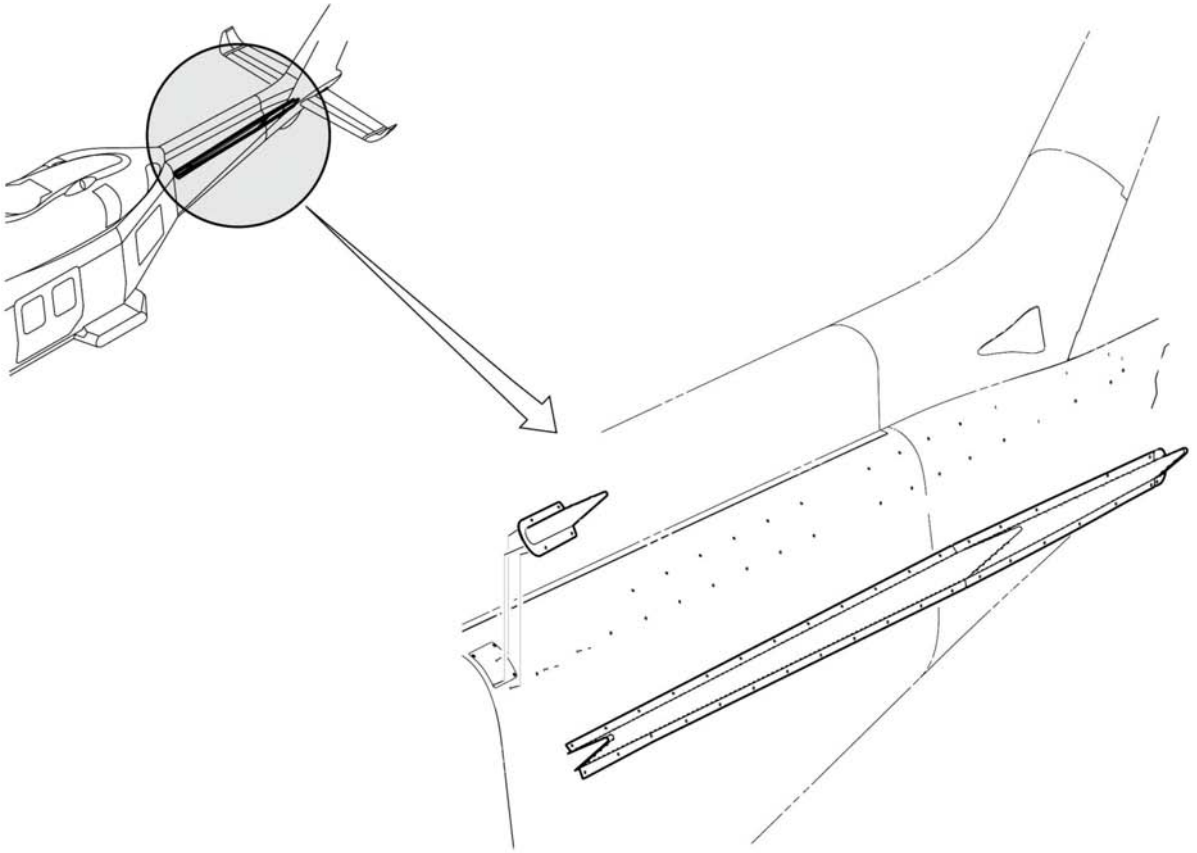


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TAIL UNIT REMOVAL/INSTALLATION

STRAKE

In hover helps the tail rotor in anti-torque. In low speed left flight reduces the left side suction.



ICN-39-A-534000-G-A0126-00726-A-01-1

STRAKE

CHAPTER

56

WINDOWS

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SECTION 00 - WINDOWS

WINDOWS – GENERAL

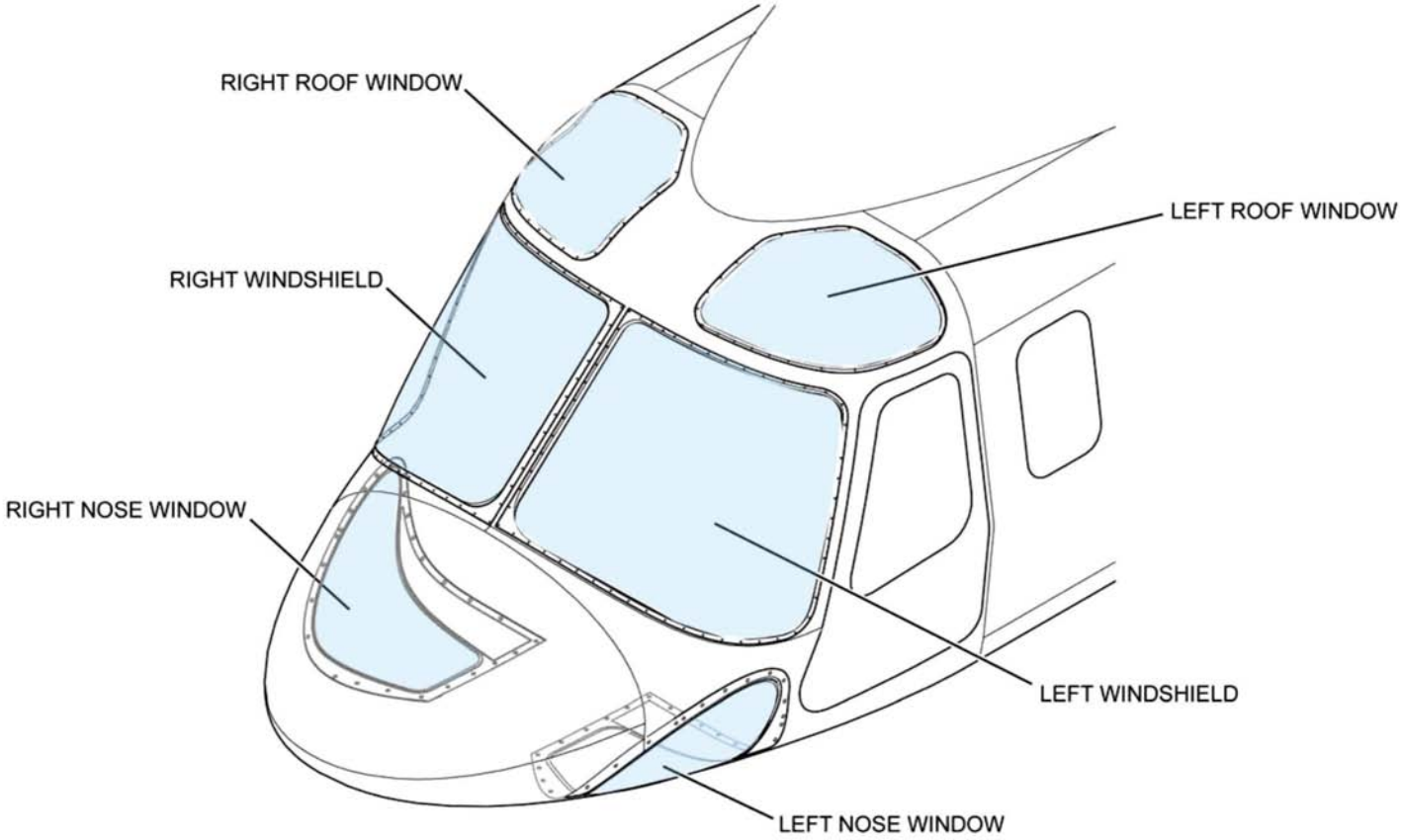
The helicopter is provided with windows in order to assure good visibility for crew and passengers. All windows are made of acrylic transparent material. Windows include:

- flight compartment windows
- fuselage compartment windows
- doors windows

FLIGHT COMPARTMENT WINDOWS

The windows of the flight compartment comprise:

- the left and the right windshield. The windshields are installed on the left (right) side of the cockpit and let the forward visibility to the crew.
- the left and the right roof window. The roof windows are installed on the top side of the cockpit and let a good visibility through the top side of the helicopter.
- the left and the right nose window. The nose windows are installed on the bottom side of the nose structure and let a good visibility through the bottom side of the helicopter.



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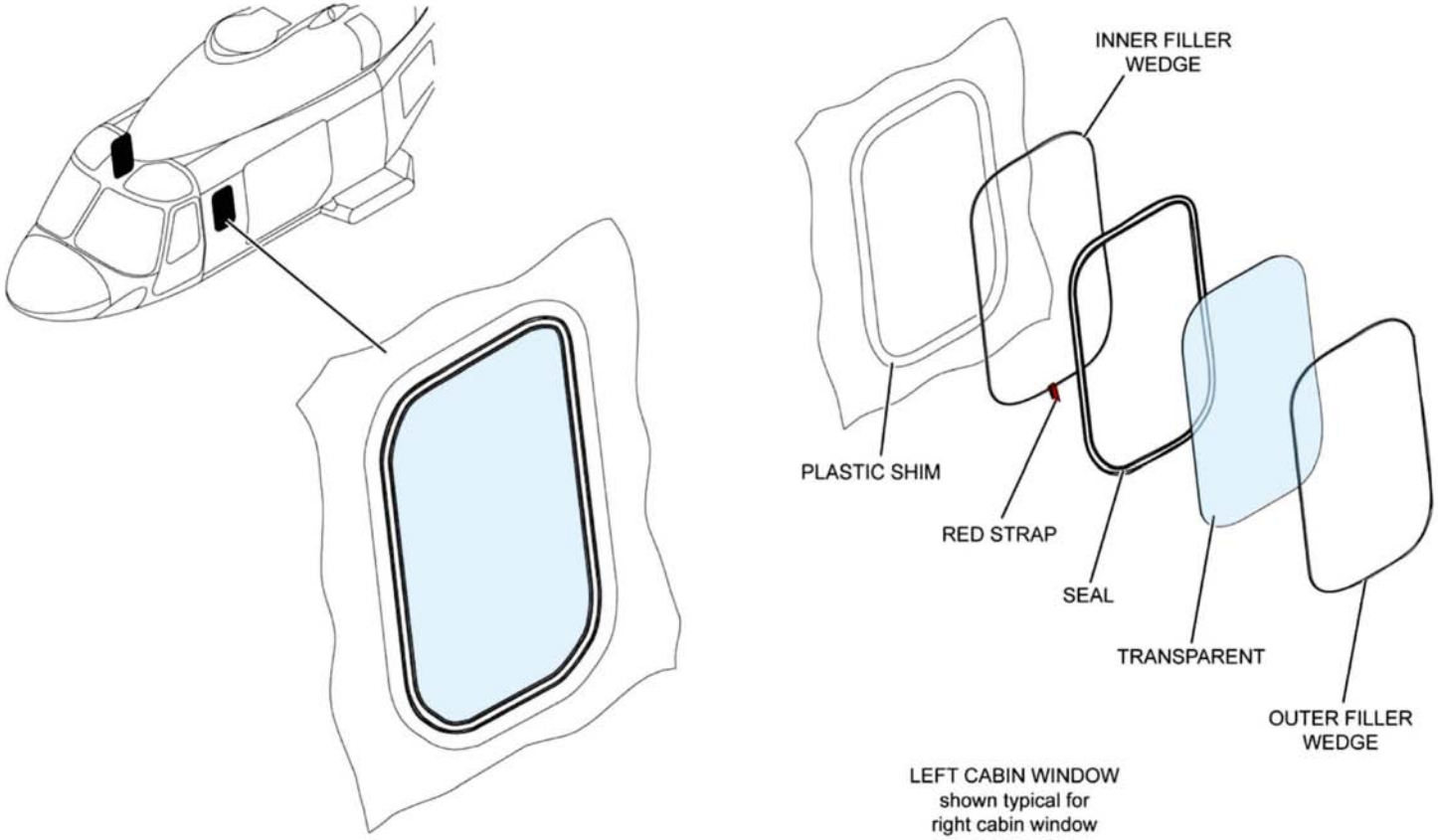
FLIGHT COMPARTMENT WINDOWS

FUSELAGE COMPARTMENT WINDOWS

the left and the right cabin windows.

- the left and the right cabin windows.

The cabin windows are installed on both sides of the cabin and are made of transparent material that lets a good visibility to passengers inside the aircraft.
A plastic shim, two filler wedges and a seal hold the transparent in its position.
A red strap is installed on the internal side of the inner filler wedge and is attached to a clinch stud.
The cabin windows can be removed in an emergency. To do this, it is necessary to remove the red strap from the clinch stud and pull it. This causes the removal of the inner filler wedge and the seal: consequently the window disengages from the frame and falls out when pushed out.



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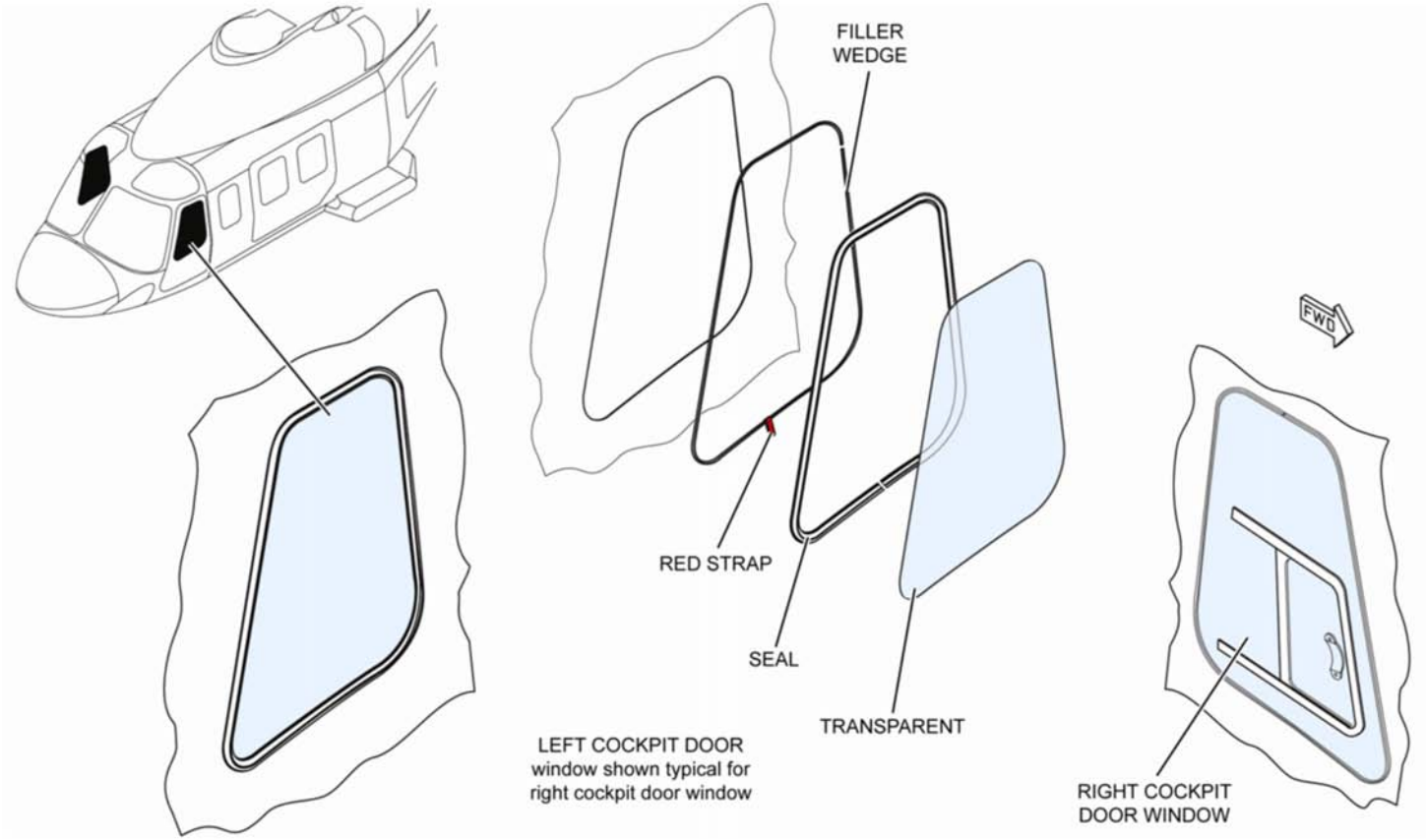
CABIN WINDOWS

DOORS WINDOWS

The door windows comprise:

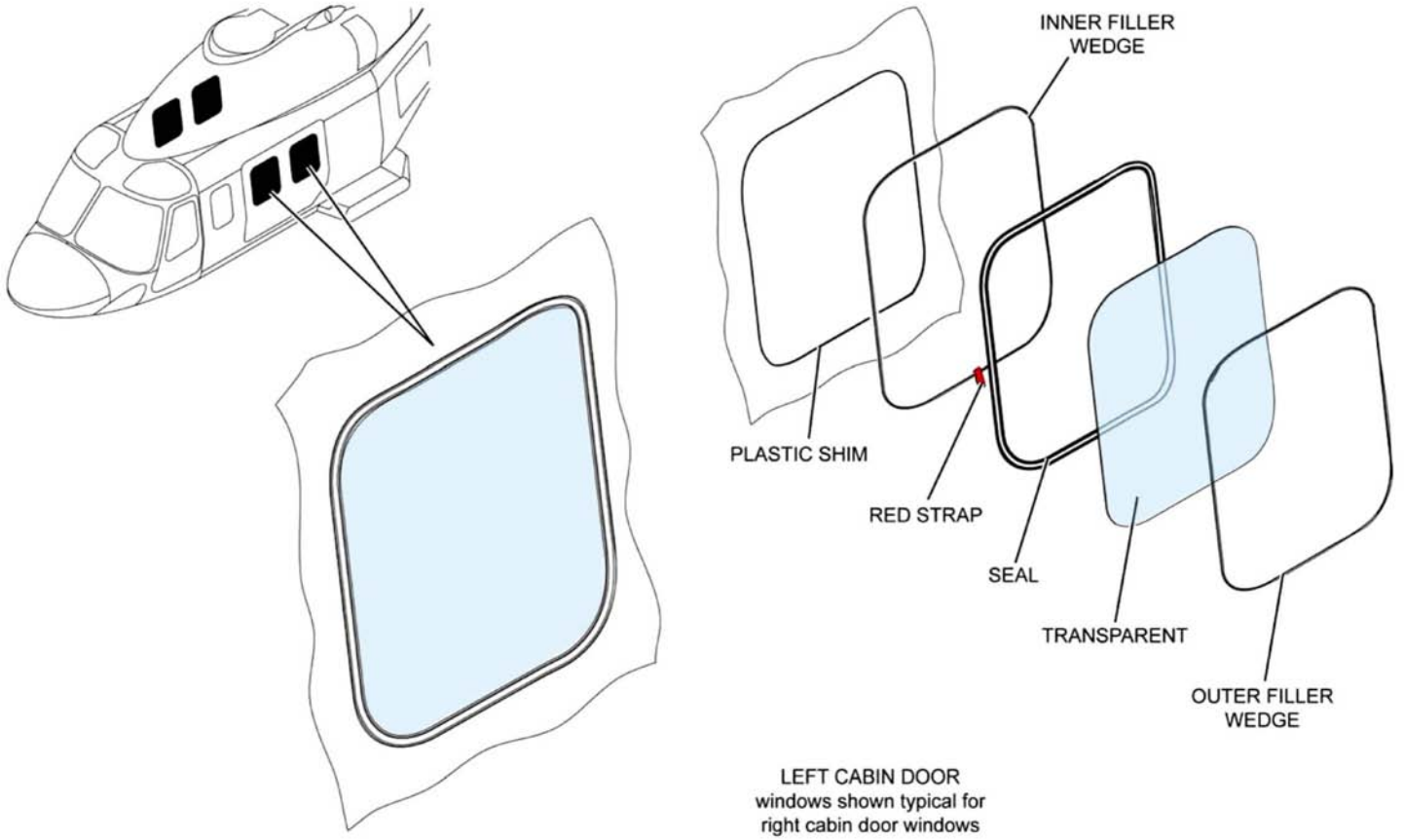
- the left and the right cockpit doors windows
- the left cabin doors forward and aft windows
- the right cabin door forward and aft windows

The left cockpit door window is installed on the left cockpit door. The window gives the visibility to the copilot and can be furnished with an additional sliding window (optional).
The left cockpit door window can be removed in an emergency. To do this it is necessary to remove the red strap and pull it: the window falls out when pushed.
The same installation is made for the right cockpit door window.
The left cabin door has two windows (one forward and one at the rear) installed on the left cabin door. The windows are made of transparent material that lets a good visibility to the passengers on the left side of the helicopter. As for the cockpit doors windows, the two cabin windows can be removed in an emergency in the way already described.
The same installation is made for the two right cabin door windows.



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COCKPIT DOORS WINDOWS



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CABIN DOORS WINDOWS

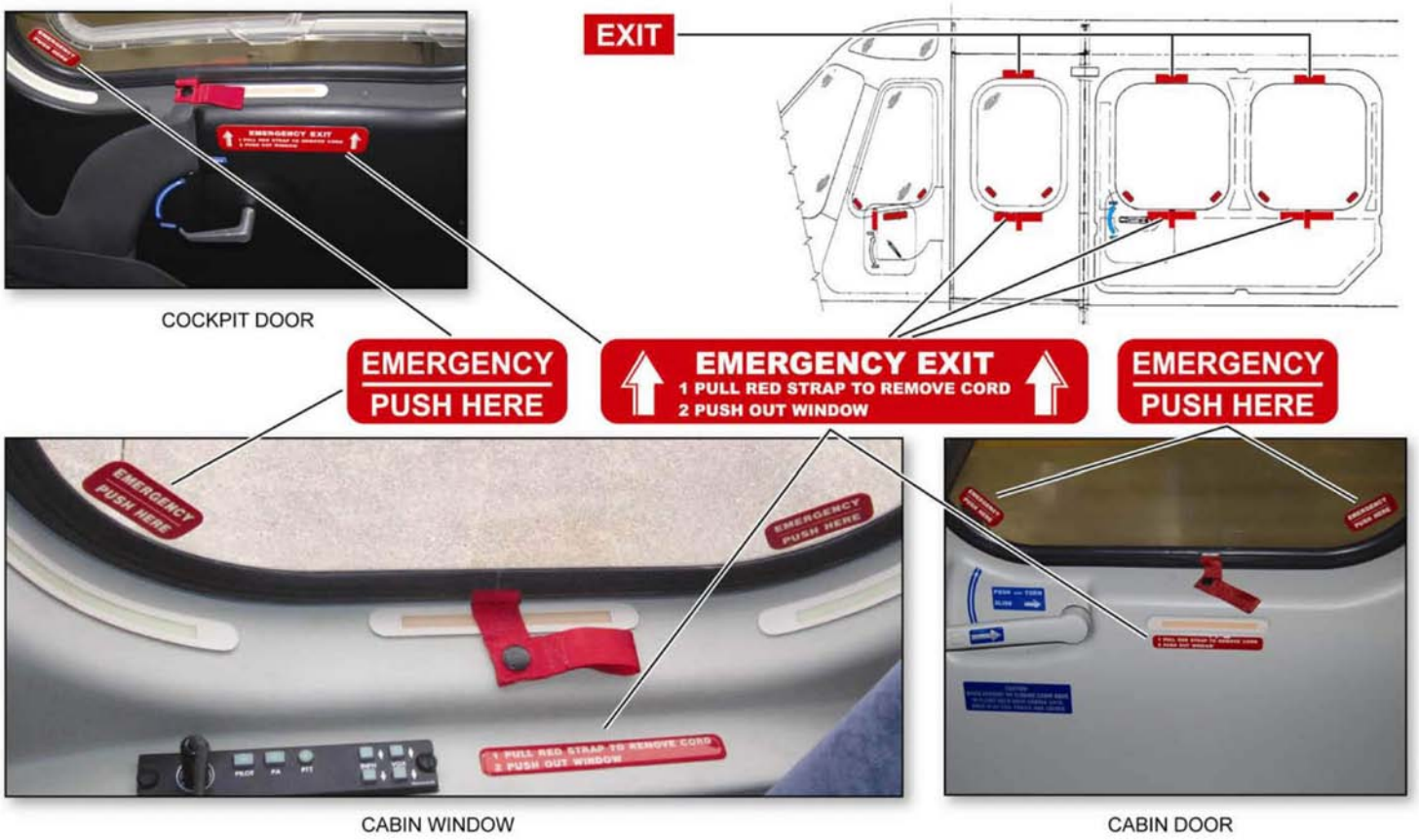
EMERGENCY EXIT

Emergency exits are located on cockpit and cabin windows.
The emergency exits of pilots and passengers are provided by the acrylic plastic windows. There are two windows in each large sliding door. The windows are jettisonable from the inside to the outside of the helicopter.
Each transparent is held in place with a silicone rubber seal. Applying hand pressure is enough to jettison the window.



ICN-39-A-510000-G-A0126-02071-A-01-1

EMERGENCY EXITS (SHEET 1 OF 2)



ICN-39-A-510000-G-A0126-02072-A-01-1

EMERGENCY EXITS (SHEET 2 OF 2)

CHAPTER
62
MAIN ROTORS

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SECTION 00 - MAIN ROTORS

MAIN ROTOR - GENERAL

The rotors provide for lift and thrust necessary for flight and include:

- the Main Rotor (MR);
- the Tail Rotor (TR).

The Main Rotor (MR) provides for aircraft lift and forward flight. The MR comprises:

- the main rotor head;
- the blades;
- the rotating controls and swashplate assembly.

The main rotor is a fully articulated type rotor with 5 blades.

MAIN ROTOR - MAIN COMPONENTS

MAIN ROTOR HEAD

The main rotor head is driven by the main rotor shaft and includes the hub, the tension links, the elastomeric bearings, the dampers, the control levers and the beanie.

HUB

The hub connects the blades to the MR shaft. It is made by titanium and splined internally to fit the MR shaft. The hub transmits the rotation to the swashplate via the scissors links.

TENSION LINKS

The tension links connect the blades to the hub and transmit centrifugal forces from the blades to the hub through elastomeric bearings.

ELASTOMERIC BEARINGS

The elastomeric bearings allow a fully articulation of the blades around the flap, pitch and lag axis. They are made by rubber and metallic disks.

DAMPERS

The hydraulic dampers react to lead lag motion and provide also the stops. A separated stop system is provided for flapping: it includes the upper stop and the lower stop. The first is provided with a support (for flight) and a limiter (for ground); the second is made by a bracket with a sliding ring at one end.
The pitch control levers provide the connection points between the MR head and the swashplate assembly. They transmit flight control input to the blades.

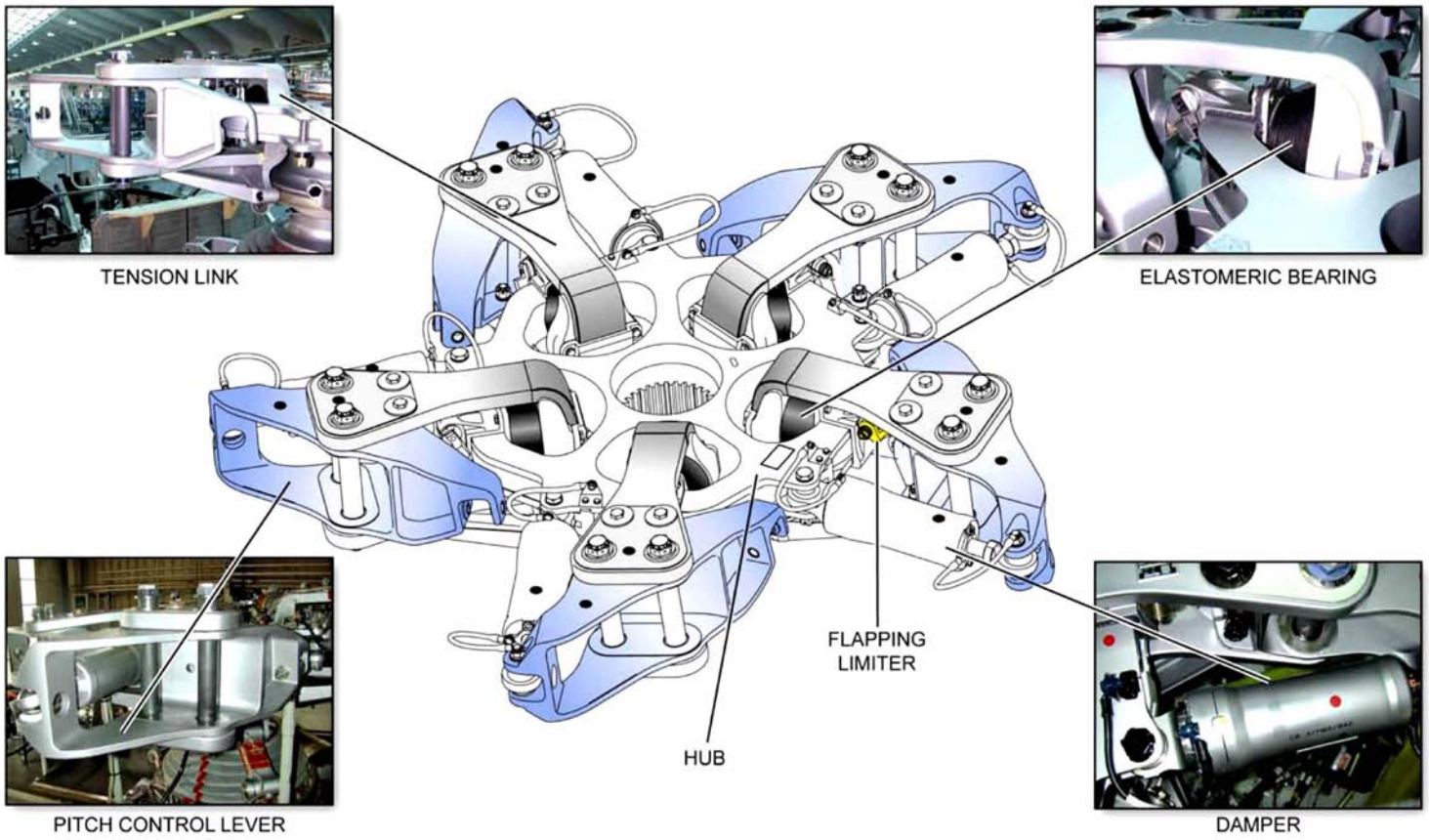
BEANIE

An aerodynamic beanie is bolded to the top of the MR head.



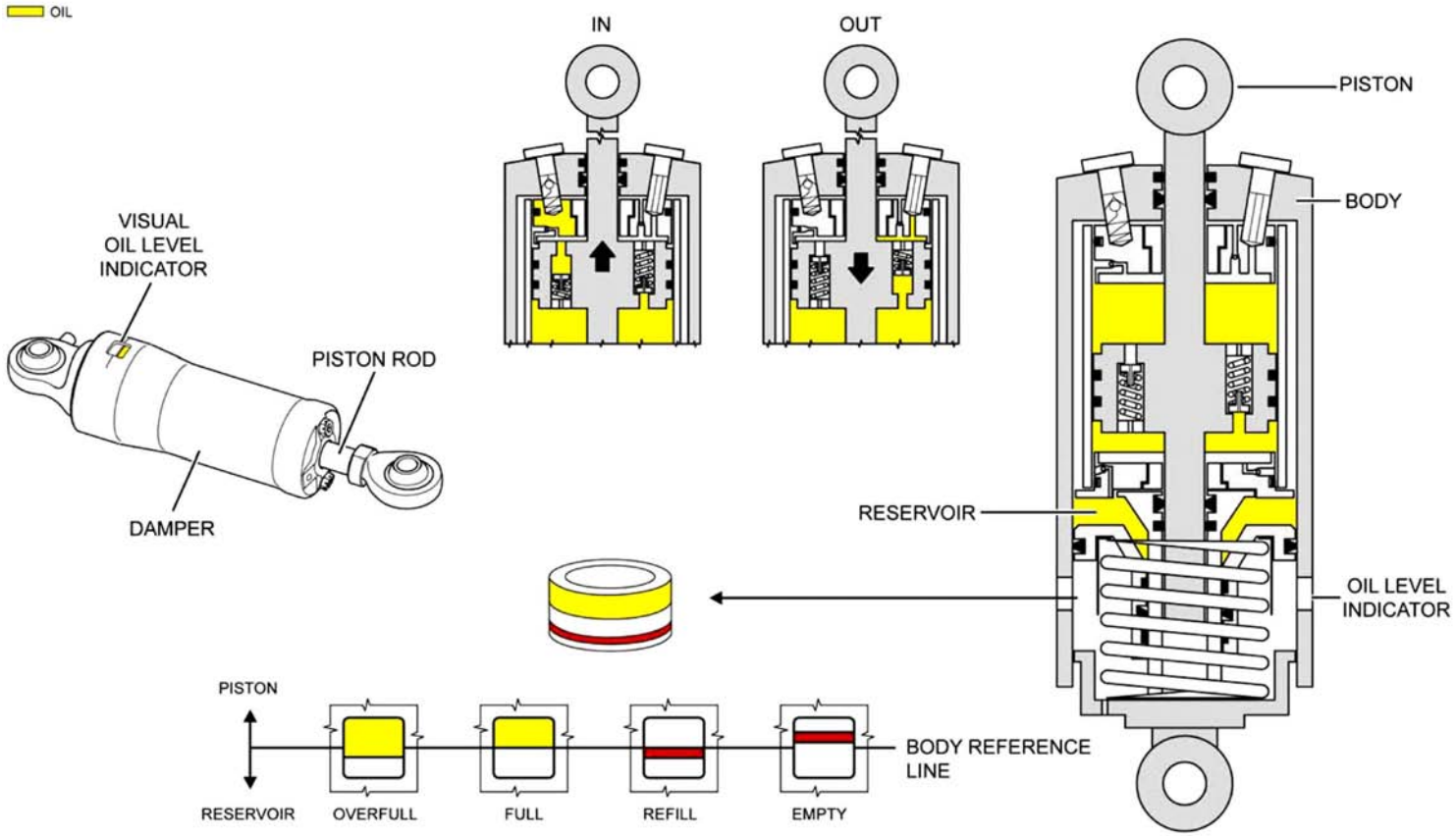
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MAIN ROTOR



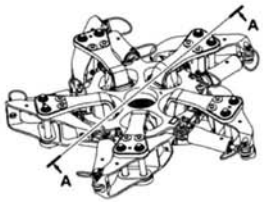
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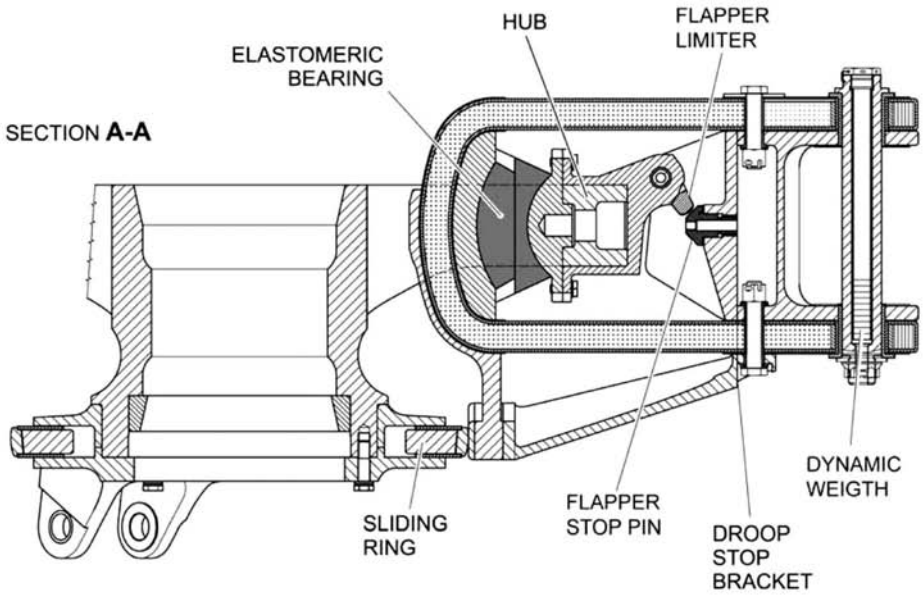
MAIN ROTOR COMPONENTS



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MAIN ROTOR DAMPER







FLAPPER LIMITER ENGAGED



FLAPPER LIMITER DISENGAGED

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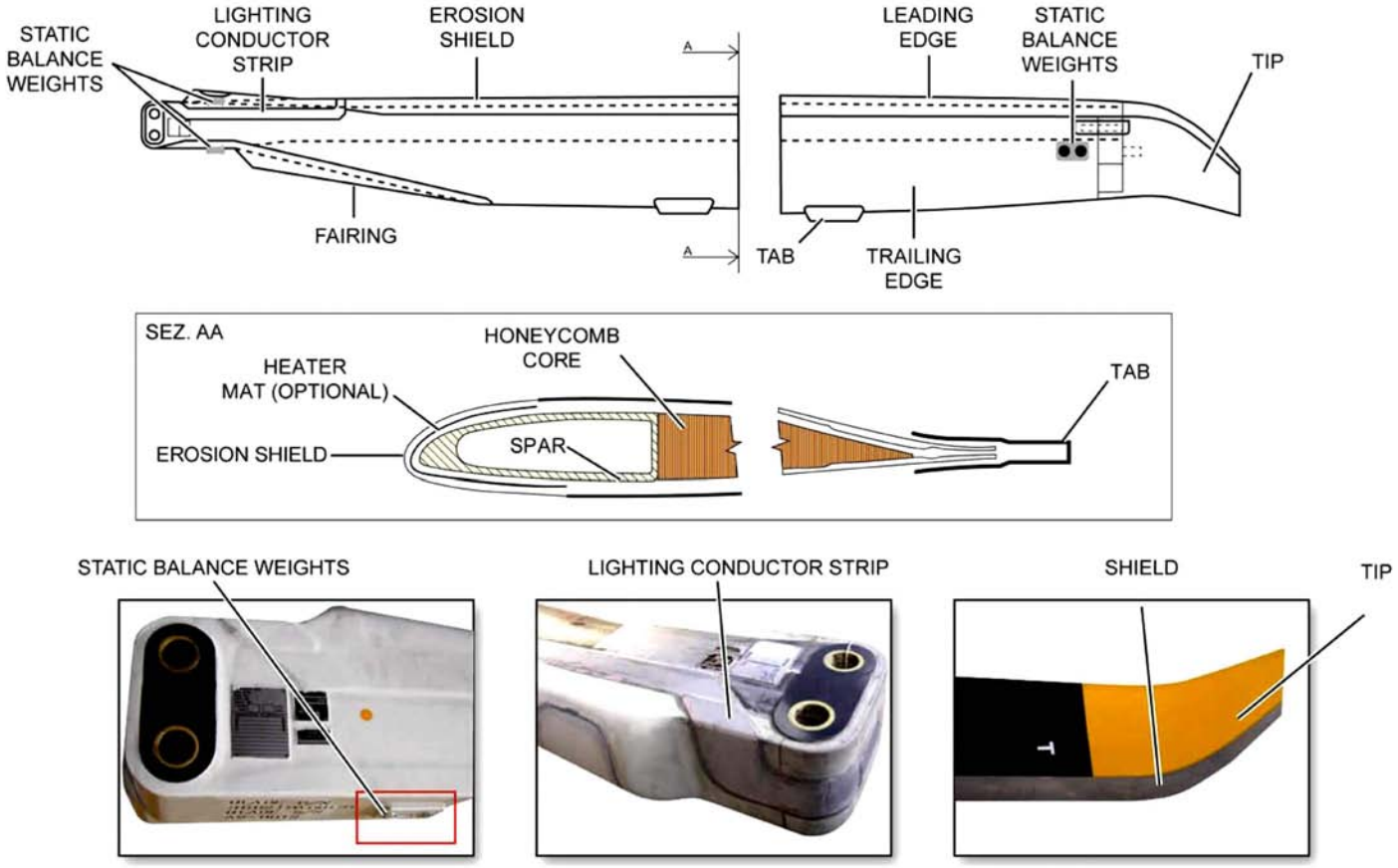
MAIN ROTOR SECTION

BLADES

Each rotor blade generates an aerodynamic lift which depends on the angle of attach of the blade.
The blades are made of composite material. A stainless steel erosion shield is attached to the leading edge of the blade. The leading edge and the trailing edge of the blade root have fairings.
Mass-balance weights are used to balance the blade statically and are inserted in two places: the first is located near the blade root and the second located near the blade tip cap.
Two metallic trim tabs are attached to the trailing edge and can be bent as necessary to do the tracking of the main rotor and to balance the blade dynamically.
A lightning conductor strip is attached to the top surface of each blade root.
The blade can be heated by means of an optional heater mat installed behind the leading edge of the blade.

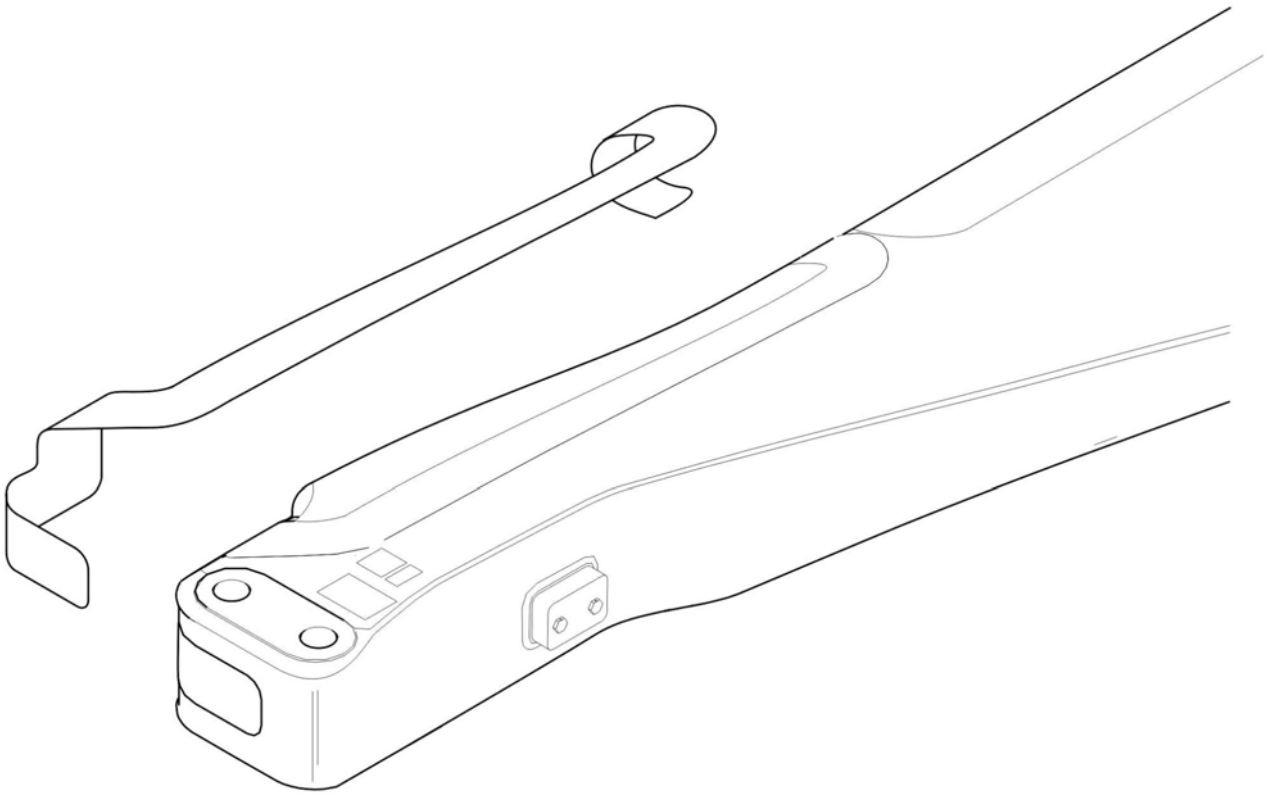
LIGHTNING CONDUCTOR JUMPERS

Five lightning conductor jumpers are installed on the main rotor head.
The position of each lightning conductor jumper is between a lag damper and the related damper bolt on the pitch-control lever assembly.
A bolt and a washer attach one of the terminal lug of the conductor jumper to the lag damper. The other terminal lug is attached to the damper bolt with a washer and a bolt.
The two bolts that attach the terminal lugs of the conductor jumper are safetied with wire.
Each lighting conductor jumper is the discharge path for the electricity if lightning hits the blade. It is necessary because the composite material of the blade is not an electrical conductor. Thus the conductor jumper connects electrically the pitch control lever of the blade and the lag damper.



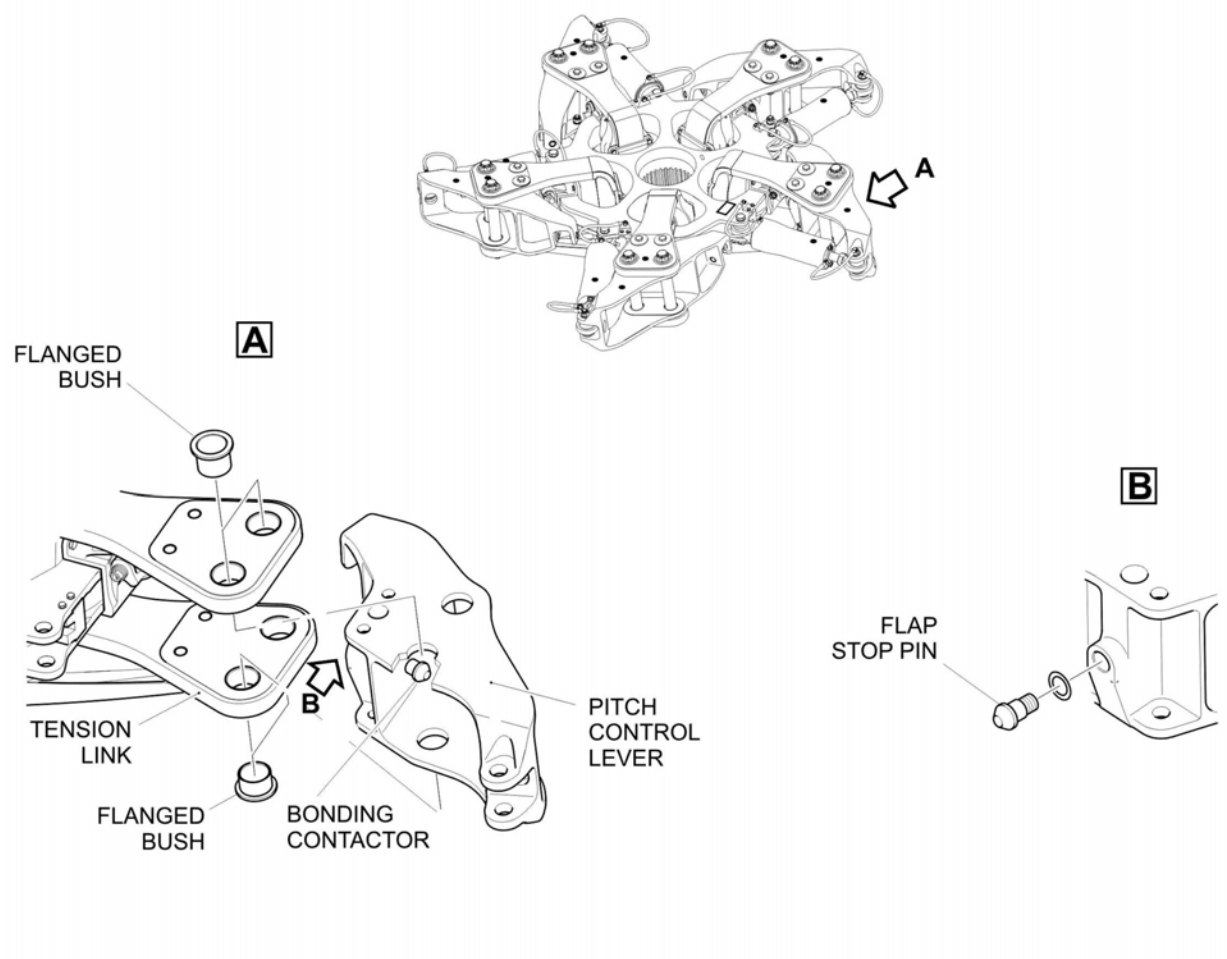
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BLADES



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MAIN ROTOR BLADE - LIGHTINING CONDUCTOR



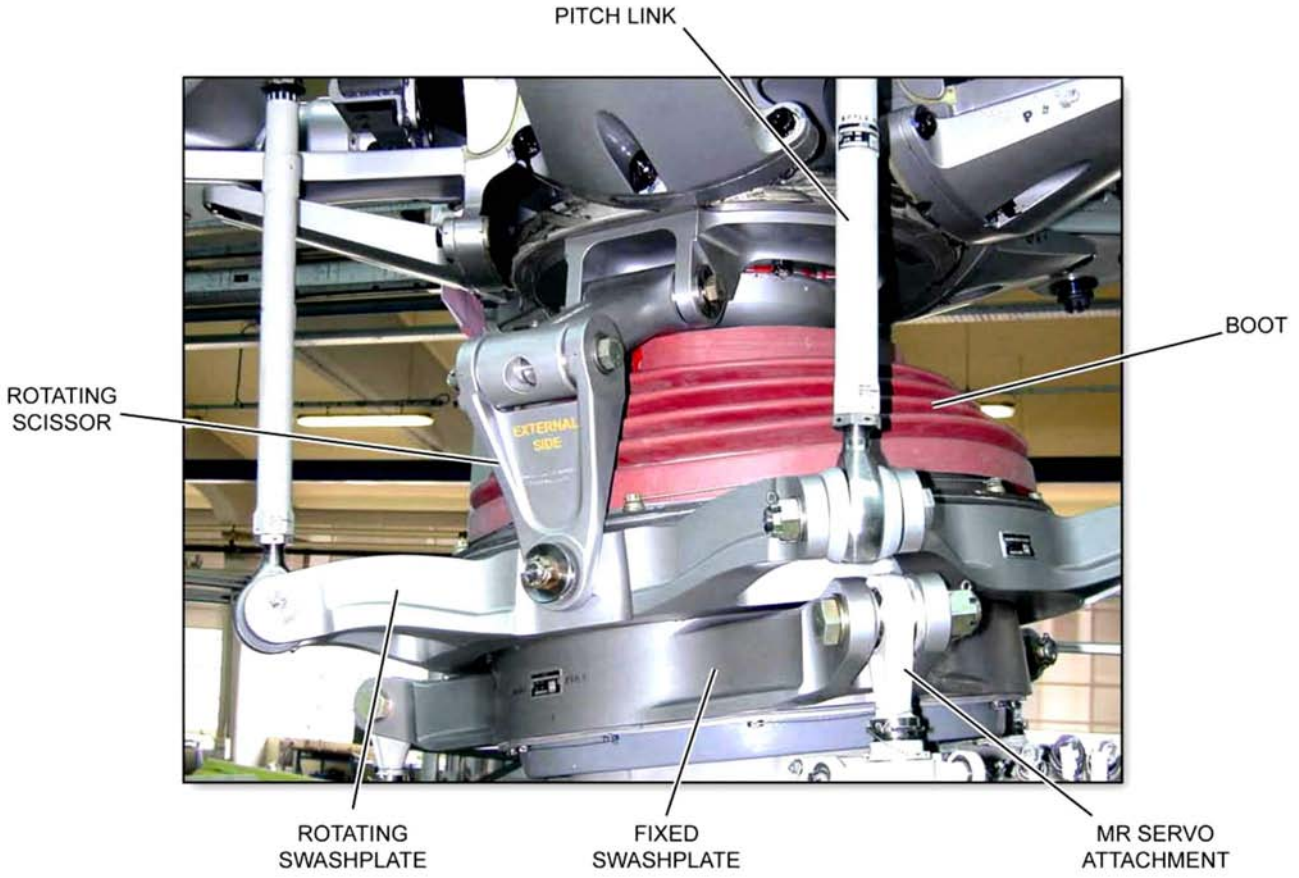
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BONDING CONTACTOR - FLAP STOP PIN

ROTATING CONTROLS AND SWASHPLATE ASSEMBLY

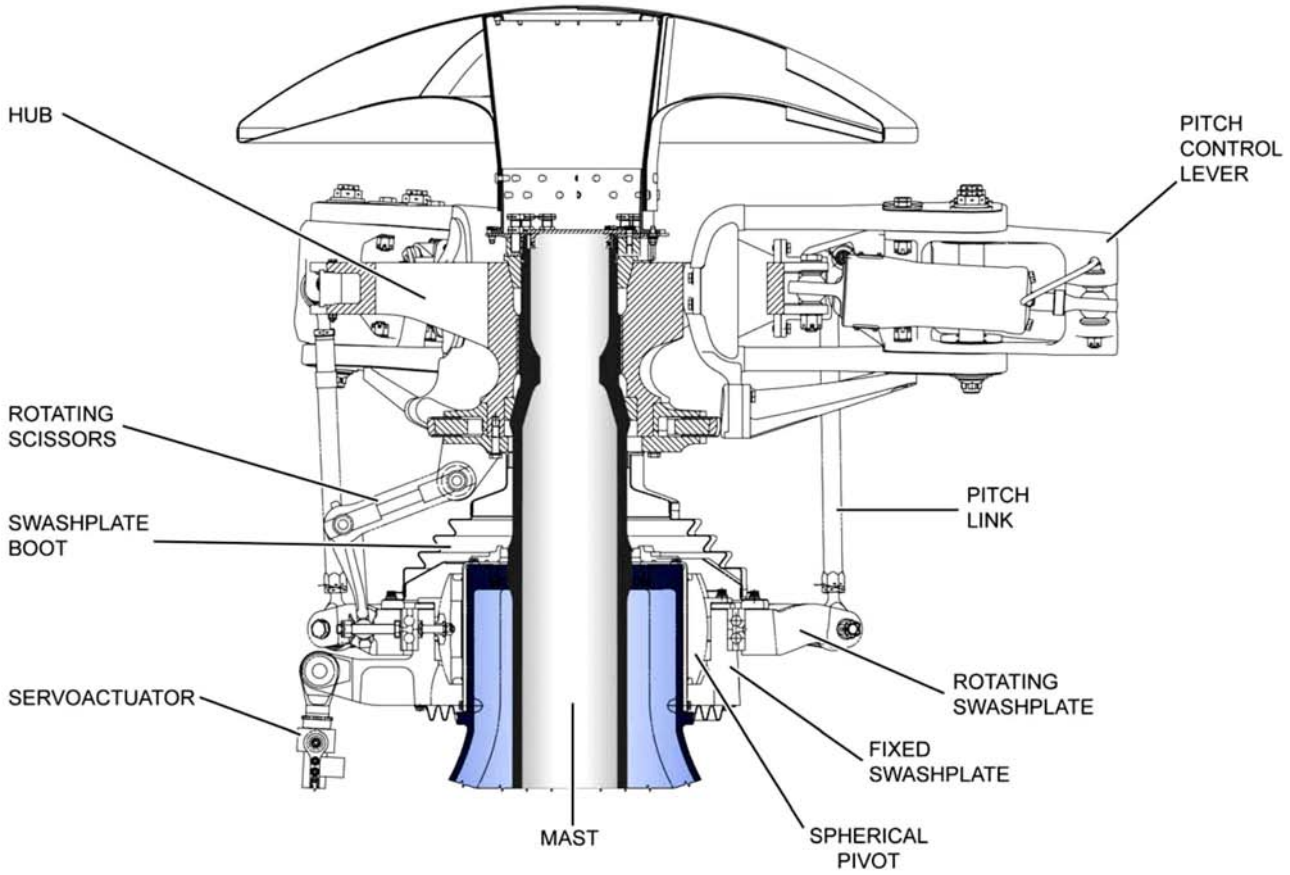
The main rotor interfaces the flight control system via the rotating controls and swashplate assembly. The rotating controls and swashplate assembly provides the attachment points for the three MR servo actuators and transmits the flight controls output coming to the blades.
The rotating controls include:

- five pitch links, one for each blade, connecting the pitch control levers of the MR blades to the swashplate;
- two rotating scissors assemblies installed between the MR hub and the rotating swashplate connecting the hub (scissors attachment flange) to the rotating swashplate and keeping it rotating. Each assembly is composed of an upper lever and a lower lever hinged to the swashplate through a ball bearing;
- two adapters installed between the MR head and the swashplate. A swashplate boot is attached to the top and bottom adapters with metallic clamps;
- the swashplate assembly installed on the top of the MGB between the MR head and the three MR servo-actuators and composed by rotating and stationary stars. The relative rotation is allowed by a double raw of ball bearing. The stationary star receives the control input from the MR servo-actuators (fixed controls) and transmits it to the rotating star. To do this the swashplate is tilted with respect to the axis of a central pivot element.



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ROTATING CONTROL AND SWASHPLATE ASSEMBLY (SHEET 1 OF 2)



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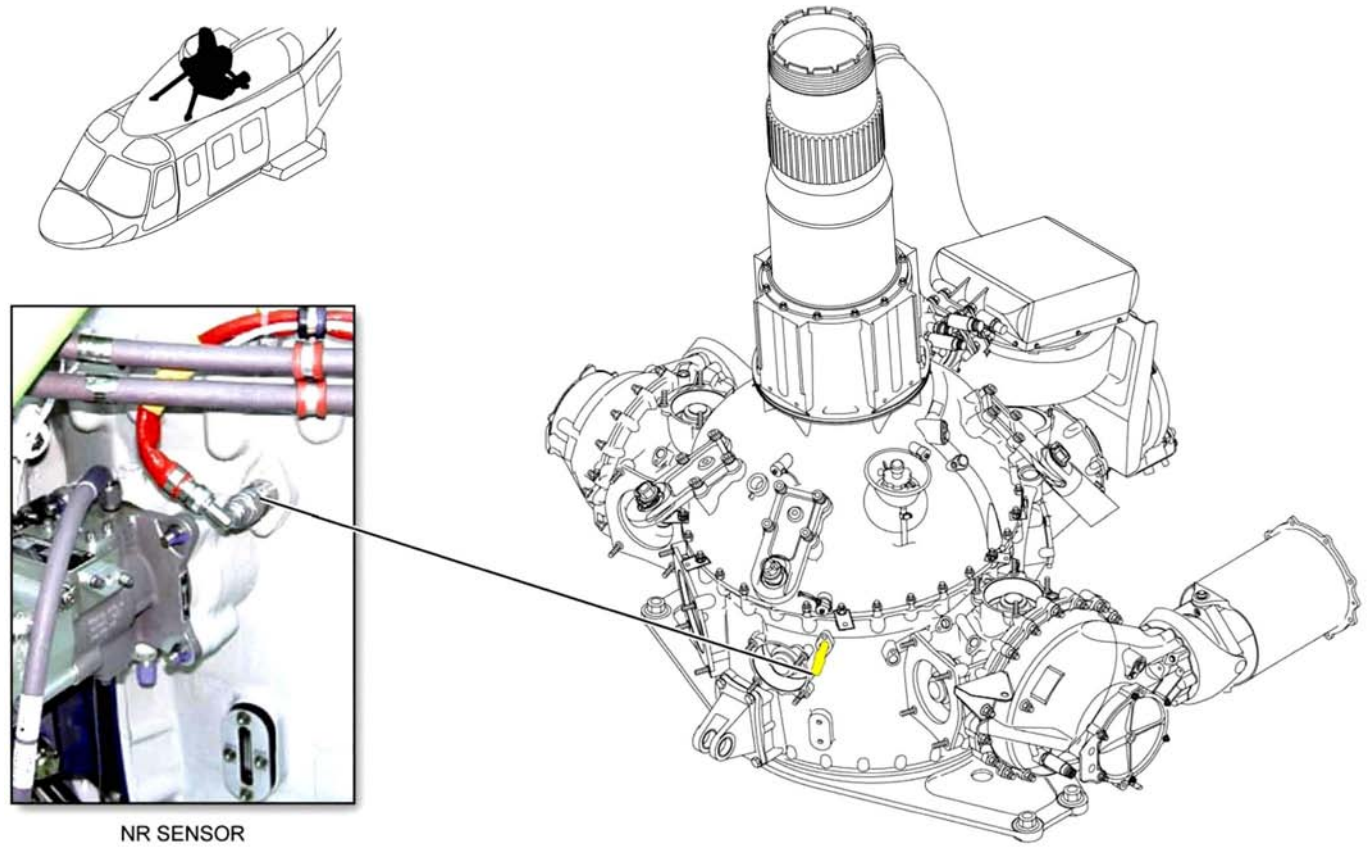
ROTATING CONTROL AND SWASHPLATE ASSEMBLY (SHEET 2 OF 2)

NR SENSOR

The main rotor speed (NR) is measured by a probe that sends three independent signals respectively to the EEC1 (Electronic Engine Control), the EEC 2 and to the MAUs (this last signal is an analogue signal used as a backup).
In addition each EEC receives relevant engine data and sends these data to the MAUs. The two EECs and the two MAUs are connected each other by the data bus ASCB-D to assure cross-talking of information.

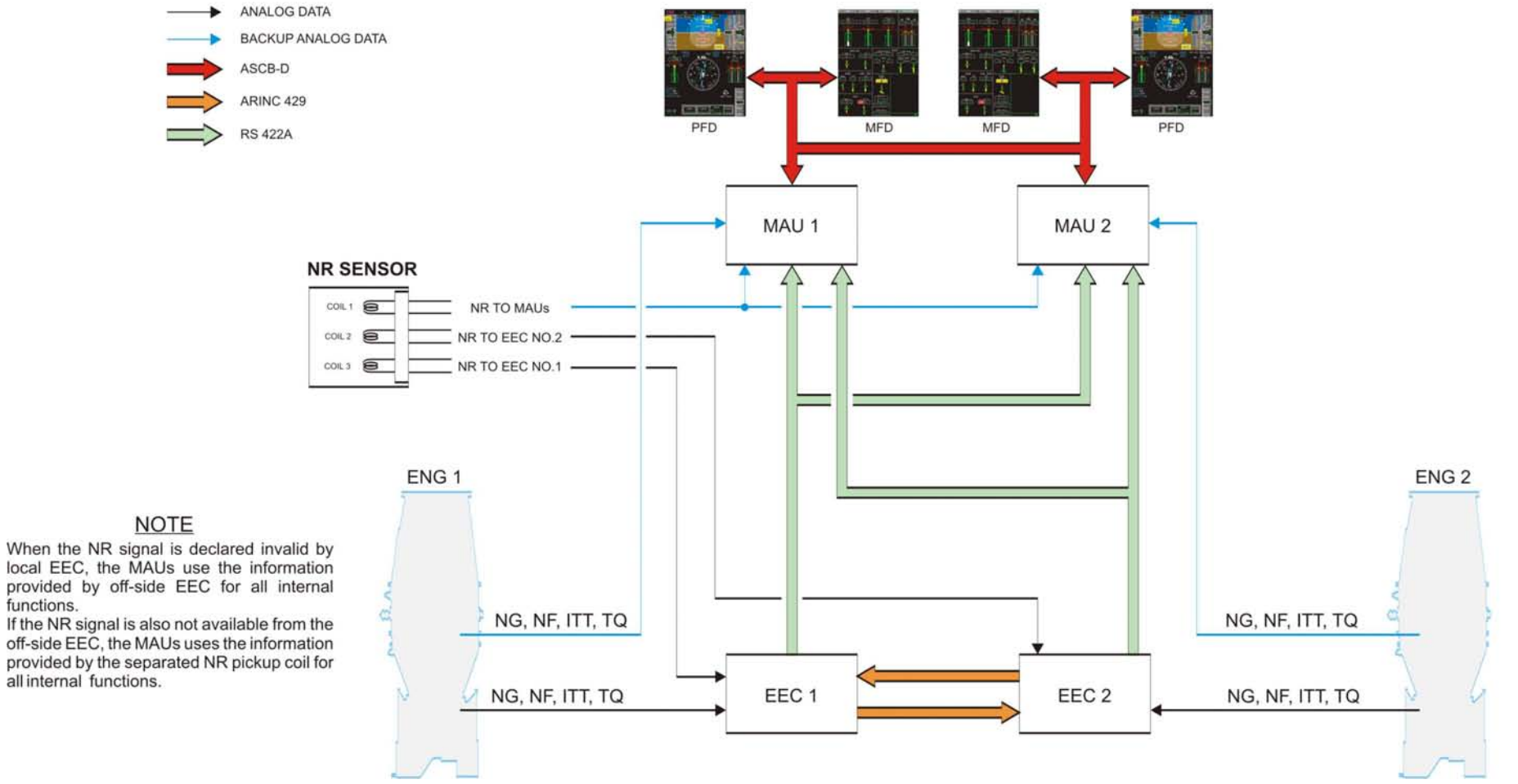
NOTE A

If the NR signal is declared invalid by the local EEC, the two MAUs use the information provided by the off-side EEC for all internal functions.
If the NR signal is also not available from the off-side EEC, than the MAUs use the information provided by the separated NR pickup coil for all internal functions.



ICN-39-A-620000-G-A0126-02080-A-01-1

NR SENSOR



NOTE

When the NR signal is declared invalid by local EEC, the MAUs use the information provided by off-side EEC for all internal functions.
If the NR signal is also not available from the off-side EEC, the MAUs uses the information provided by the separated NR pickup coil for all internal functions.

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NR SENSOR OPERATION

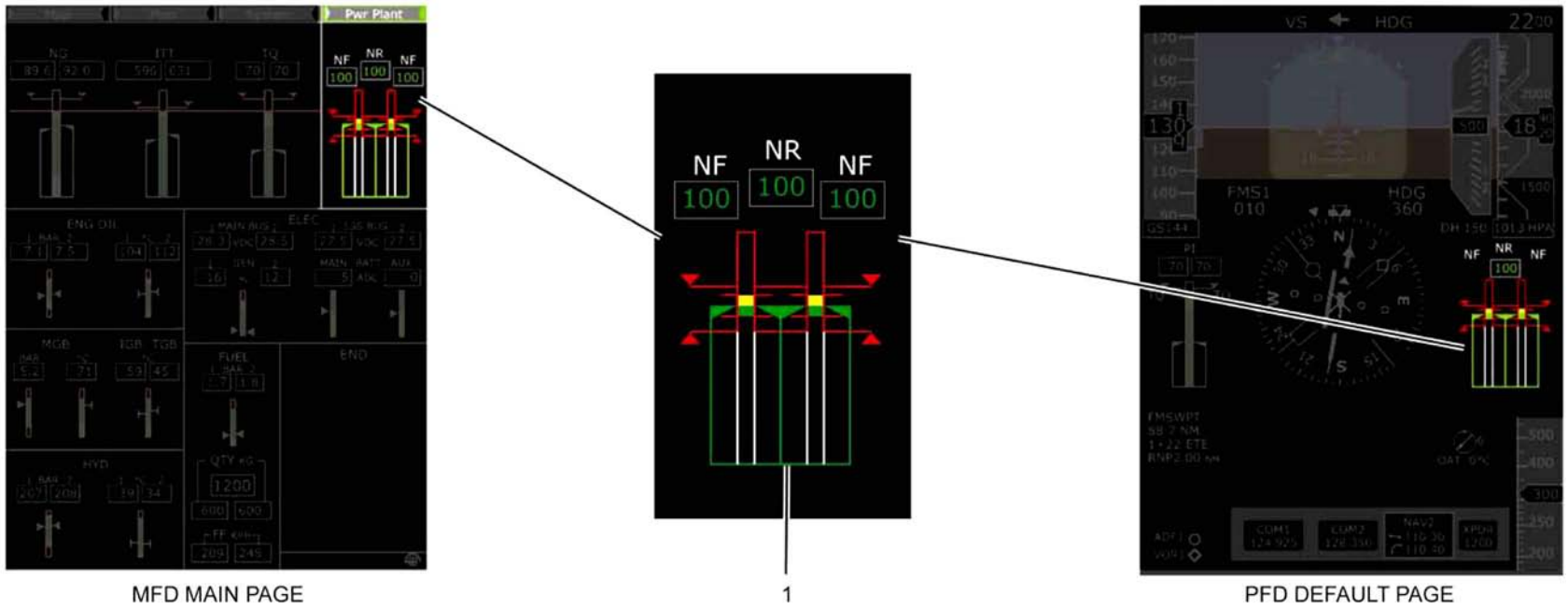
MAIN ROTOR – INDICATIONS

The PFD default page and the MFD MAIN page display the values of Nf or NF (engine free turbine speed) and NR grouped in the TRIPLE TACHOMETER scale.

- NF/NR (Power ON)..... The values of NF in the engine no.1 and no.2 are represented by digital readouts under the two NF labels on the left and on the right respectively. Graphically these values are represented on a vertical scale by means of two pointers (triangles) that match the color of the area on the scale.
  The digital readouts and the pointers are displayed red when the pointer is in the red zone (warning), amber when the pointer is in the amber zone (caution) and green in normal conditions.
  A red horizontal line is displayed corresponding to the MIN TRANSIENT NR 95% and the MAX TRANSIENT NR/NF 106%.

When the EECs and the Modular Avionic Units (MAUs) detect an invalid NR/NF input signal, the associated pointer is removed from the display and amber dashes replace the numerical readouts.
If the NF sensor fails, the relative pointer is removed and replaced by the amber legend FAIL in reverse video.
The baseline of the triple tachometer is green at all times, even when the pointers are amber or red.

When a parameter being monitored exceeds the normal range of operation (green band), the colour of associated pointer matches the colour of applicable range marking in order to highlight the particular critical condition.



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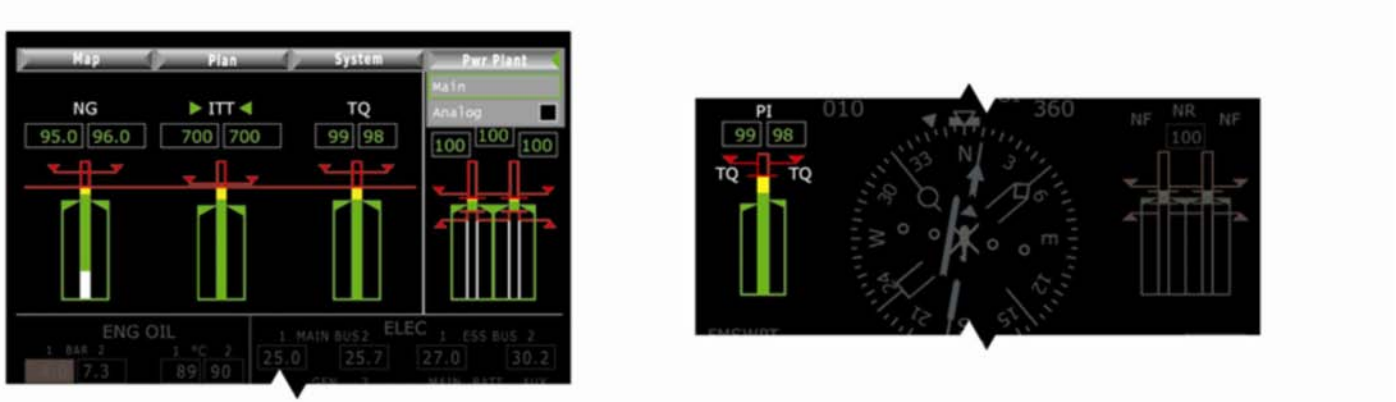
MAIN ROTOR INDICATION

PRIMARY DATA / BACKUP DATA INDICATING

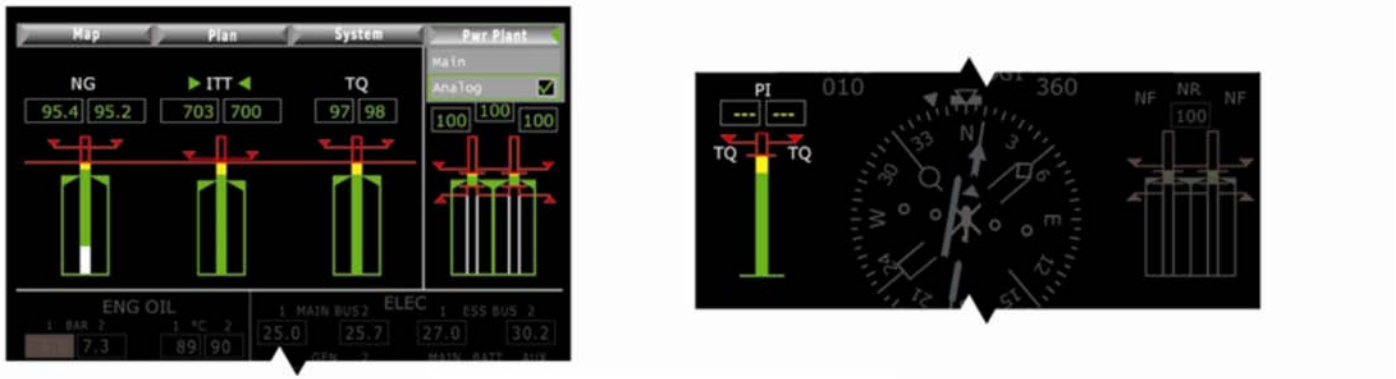
In case any of the digital primary engine data is not available from the EEC, the corresponding analog input from the NR sensor to the MAU is automatically selected for display (backup data source). If the primary engine data digital bus communication is lost, the 1(2) EEC DATA caution message is displayed in the CAS.
The backup data can also be manually selected for display by checking the Analog check box in the Pwr Plant page menu.
When backup data source is selected for display, PI indication is not computed: PI pointer is removed and digital readout shows amber dashes.

MAUS COMPARISON FUNCTION

In case of miscompare between analogue data and digital data, the caution message will be generated in the CAS window. (Ref. to pag. 25)
In case of failure of an analogue parameter, the caution message ENG ANALOG FAILURE will be generated in the CAS window.
When ANALOG is selected the displayed data (backup data) comes directly from the relative sensors.
In case of primary date is declared invalid by the EEC, the MAUs use the back-up sensor for display, monitor warning, etc., purpose.



ENGINE PRIMARY DATA SELECTED



BACKUP ENGINE DATA SELECTED (ANALOG)

ICN-39-A-620000-G-A0126-03355-A-01-1

PRIMARY DATA / BACKUP DATA INDICATING

MAIN ROTOR – CONTROLS AND INDICATORS

1 100% / 102% RPM selector switch on the pilot collective grip

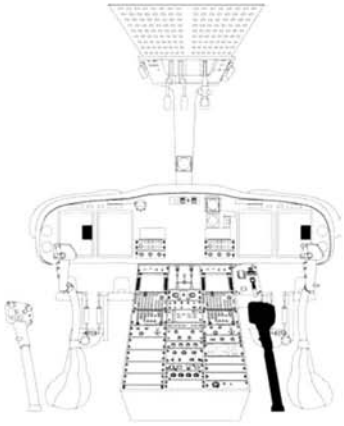
- 100% ..... set the NR / NF to 100%
- 102% ..... set the NR / NF to 102%

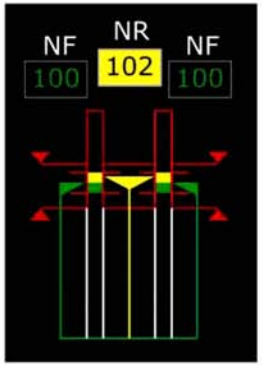
NOTE B

With RPM selected at 102% VNE is 90 KTS
NR reach yellow range
There are NO CAS messages during this type of selection



PILOT COLLETIVE ONLY





ICN-39-A-620000-G-A0126-02084-A-01-1

MAIN ROTOR – CONTROLS AND INDICATORS

CAS WARNING MESSAGES	
CAS CAPTION	**FAILURE DESCRIPTION**
ROTOR LOW	Rotor RPM below limit • when the NR < 98 % in power-on condition • when the NR < 95 % in power-off condition • when the NR < 90 % in power-on condition (OEI) 3 tones + voice warning: ROTOR LOW – ROTOR LOW This sequence is continuously repeated until the failure condition is corrected or the reset input activates
ROTOR HIGH	Rotor RPM above limit • when the NR < 104 % in power-on condition • when the NR < 110 % in power-off condition • when the NR < 104 % in power-on condition (OEI) 3 tones + voice warning: ROTOR HIGH – ROTOR HIGH This sequence is continuously repeated until the failure condition is corrected or the reset input activates

CAS CAUTION MESSAGES	
CAS CAPTION	**FAILURE DESCRIPTION**
NR MISCOMPARE	Discrepancy between EEC and analogue value of NR NOTE C The caution is generated when comparison with backup parameter exceeds NR 3% This caution shall be displayed only if NR_{EEC} > 20% or NR_{EEC} < 20%
ENG ANALOG FAILURE	Failure of an analogue parameter
RPM SELECT	RPM selector switch failed

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CHAPTER
63
MAIN ROTOR DRIVES

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SECTION 00 - MAIN ROTOR DRIVES

TRANSMISSION - GENERAL

The transmission supplies the drive from the engines to the Main Rotor (MR), the Tail Rotor (TR) and the accessories (pumps, ECS compressor, and alternators - if installed).
The transmission is composed by:

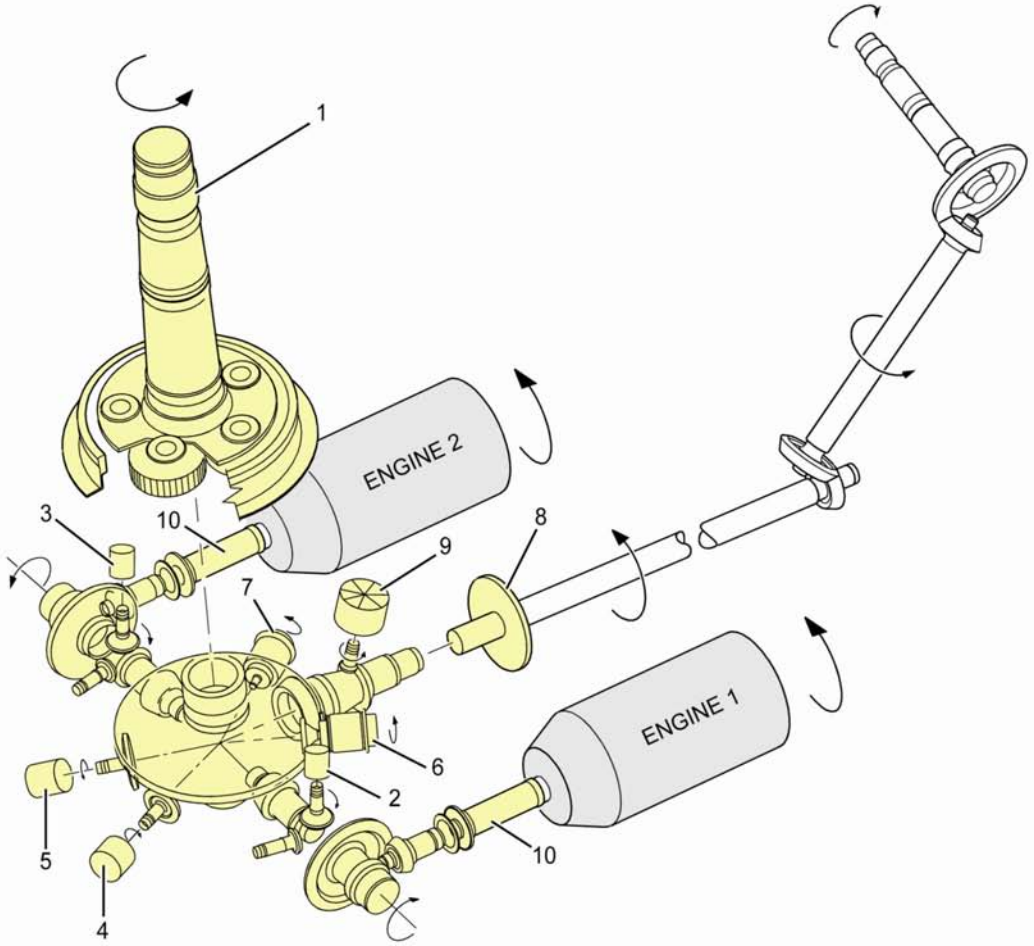
- the Main Rotor Drive System (MRDS)
- the Tail Rotor Drive System (TRDS)

as visualized in the schematic that follows. This chapter describes the Main Rotor Drive System.

☐ Main Rotor Drive System

1. Main Rotor mast
2. Pump 1
3. Pump 2
4. Pump 4
5. ECS Compressor - optional
6. MGB lubricating pump (left)
7. MGB lubricating pump (right)
8. Rotor Brake (RB) disk
9. MGB oil cooler fan
10. Drive shaft

☐ Tail Rotor Drive System
Refer to Ch. 65



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TRANSMISSION - GENERAL

MAIN ROTOR DRIVE SYSTEM – GENERAL

The Main Rotor Drive System transmits and decreases the rotational drive from the engines to the Main Rotor. The MRDS comprises:

- the Main Gear Box (MGB);
- two engine gearbox couplings;
- two input modules;
- the engine couplings;
- the freewheels.

The MGB reduce the rotational speed between the input and output drive. The MGB has a self-contained splash lubrication and condition-monitoring oil system.

MRDS – MAIN COMPONENTS

MAIN GEARBOX

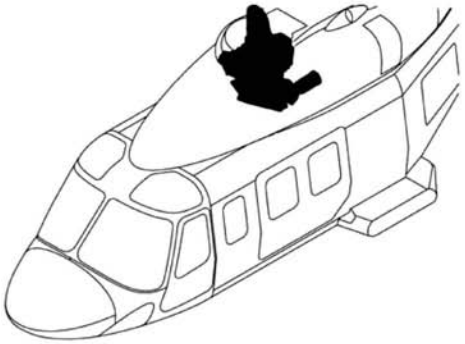
The MGB is the most important component of the MRDS. The MGB is made of aluminium alloy and comprises two (engine) input shaft modules cases, the main case, the upper module case and the bottom case.

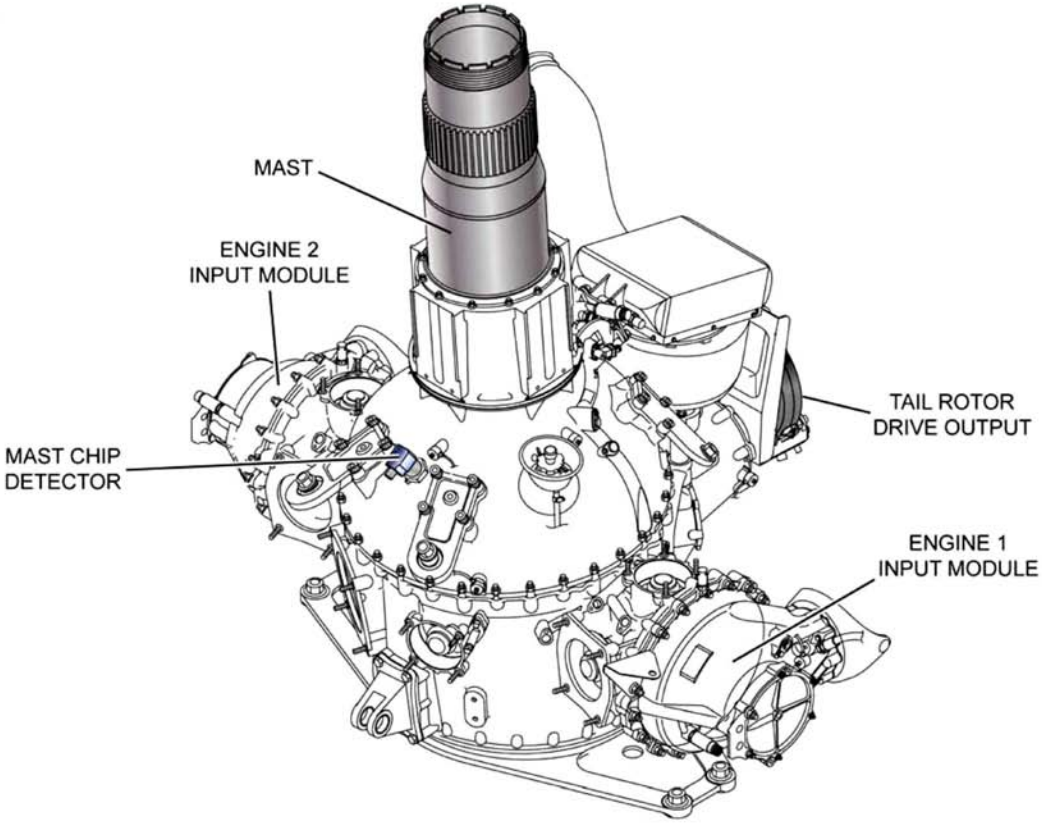
The MGB changes the horizontal drive from the engines to a vertical drive of the main rotor (MR) shaft. The MR shaft is secured to the MGB and drives the MR head on which are attached the blades.

The MGB has three reduction stages that reduce the 21000 rpm of the engines input speed to the 296 rpm of the MR speed (at 100% Nr).

The MGB supplies drive through the Tail Rotor Drive System (TRDS) to the Tail Rotor (TR) and supplies drive at different rotational speeds to the following accessories:

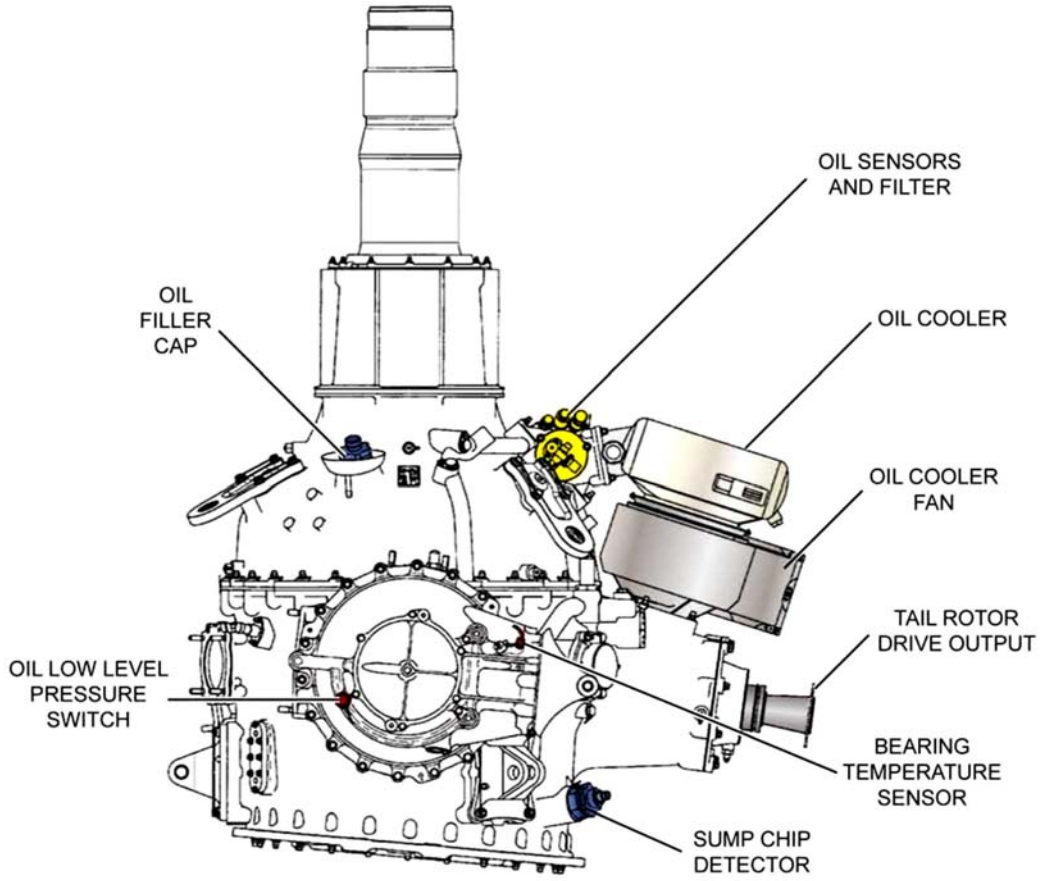
- one MGB oil cooler fan;
- two MGB lubrication pumps;
- three hydraulic power pumps (pump 1, pump 2, pump 4);
- one ECS compressor (optional).





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MAIN GEARBOX



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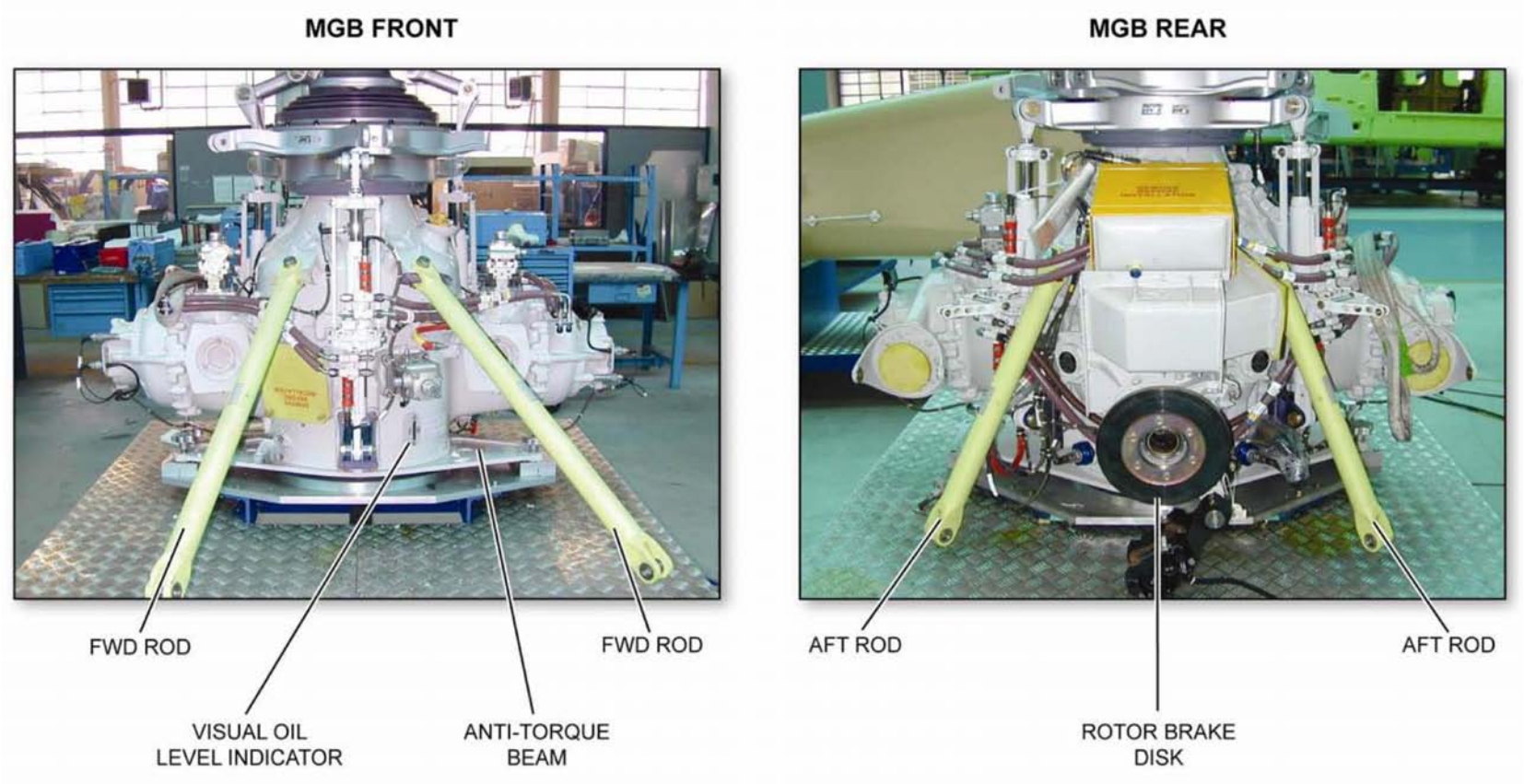
MAIN GEARBOX – SIDE VIEW

MOUNTS AND ATTACHMENTS

Mounts and attachments support and restrain the MGB to the helicopter fuselage. The MGB holds the attachments of

- three MR servo actuators (FW, LH, RH)
- four rods (two FWD and two AFT)
- one anti-torque beam bolted to the bottom of the MGB
- the rotor brake system

The four rods and the anti-torque beam connect the MGB to the helicopter upper deck and allow to transfer torque and dynamical loads from the main rotor mast to the fuselage.
The MGB is tilted 5° forward respect to the helicopter upper deck.



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MAIN GEAR BOX MOUNTS AND ATTACHMENTS

ENGINE GEARBOX COUPLINGS

Each one of the two engine gearbox coupling transmits the drive from the relevant engine shaft to the MGB input module (left and right input module).

INPUT MODULES (LEFT AND RIGHT)

Each one of the two input module (left and right) realises the connection between the engine shaft and the MGB.
The input module changes the direction of the axis of rotation and acts as first reduction stage for the MGB. Each input module includes a centrifugal freewheeling unit that allows to override the engine in case of failure or engine shutdown so that the rotor can be driven by the remaining operative engine.

GIMBAL ASSEMBLIES

There are two gimbal assemblies; each one gives forward mounting-support to the relevant engine.

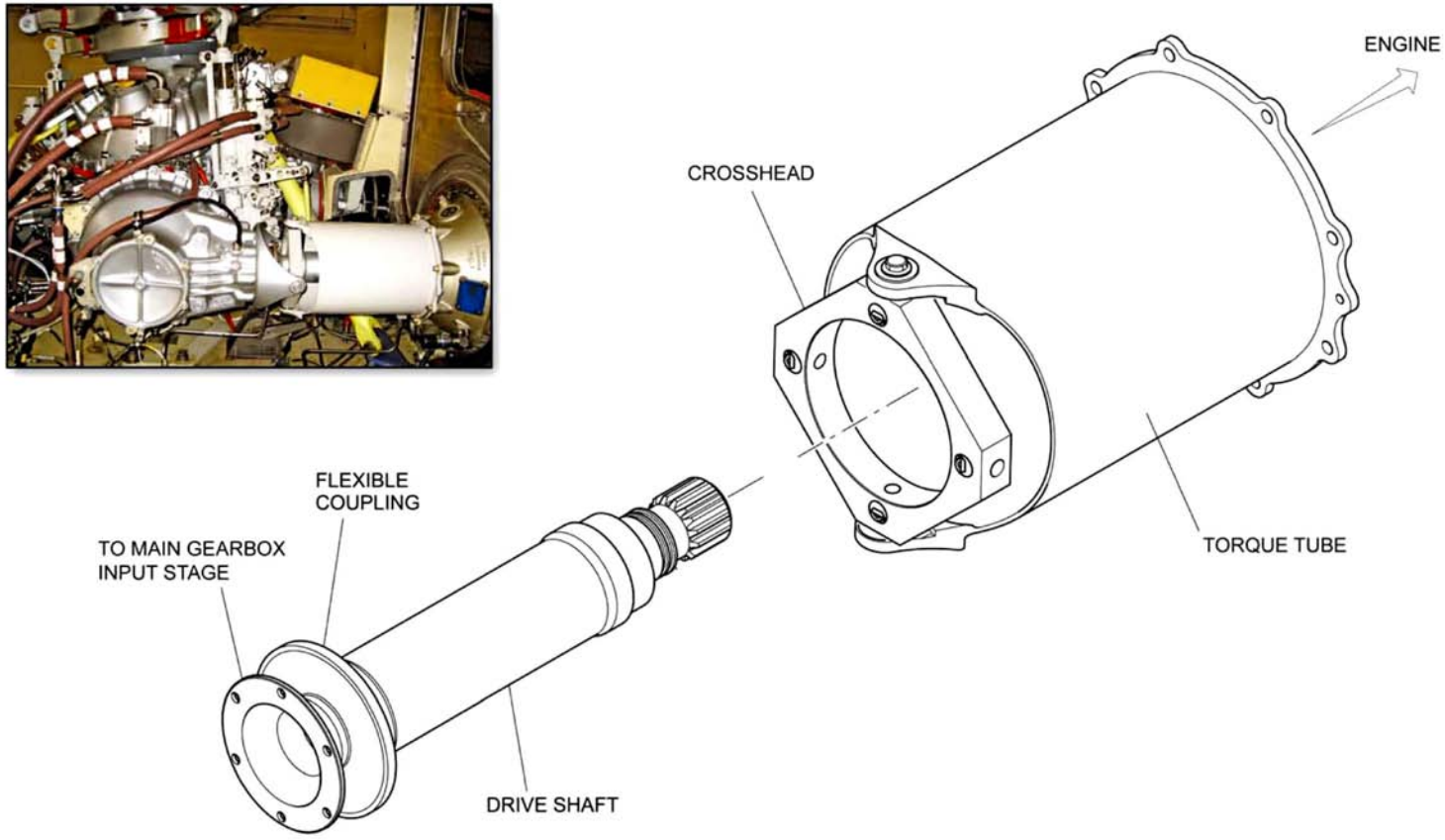
FREEWHEELS

Freewheels enable the movement to be transmitted only in the engine to rotor direction. There are two freewheels, one for each input module. There are two possible conditions:

- Freewheel engaged: rollers are caught between the input shaft (engine shaft) and the MGB output shaft that remains driven;
- Freewheel disengaged (autorotation or one engine failed): the output shaft which is driven by the rotor (or by the second engine) frees the rollers: the freewheel acts as a bearing and the (failed) engine is not driven.

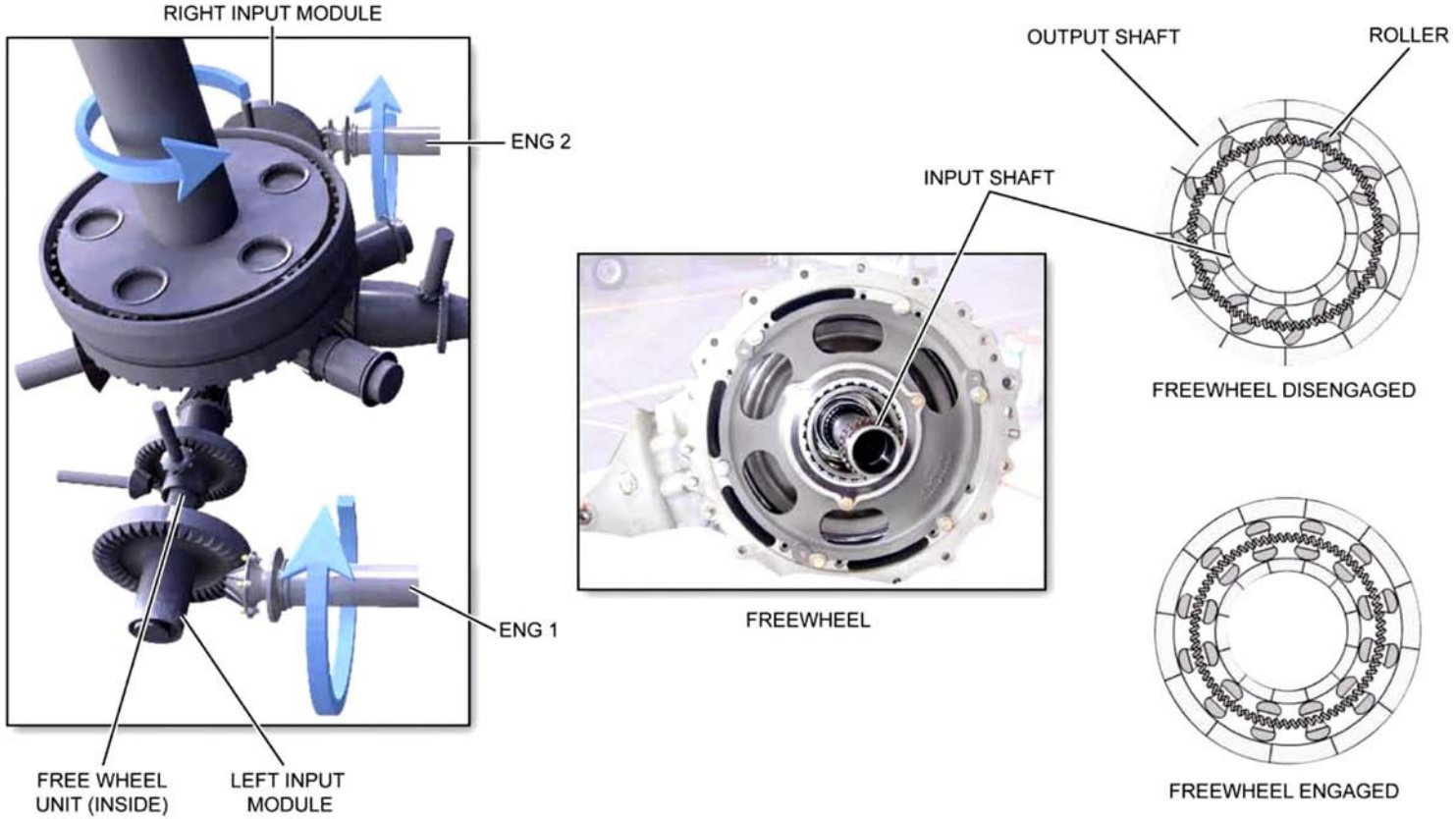
When both engines are operating, the LH and the RH freewheels are engaged and the main rotor and tail rotor (and the accessories) are driven.

In autorotation both freewheel are disengaged: the tail rotor and the accessories are driven by the main rotor.



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ENGINE - MGB COUPLING



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FREEWHEELS UNIT

MAIN ROTOR DRIVE – INDICATIONS

The PWR PLANT PAGE displays the values of pressure and temperature in the MGB area.

1 PRESSURE
The values of pressure in the MGB are represented by digital readouts under the label BAR. Graphically these values are represented on a vertical scale by means of a pointer (triangle) that matches the color of the area on the scale.
The digital readouts and the pointers are displayed red when the pointer is in the red zone (warning), amber when the pointer is in the amber zone (caution) and green in normal conditions.

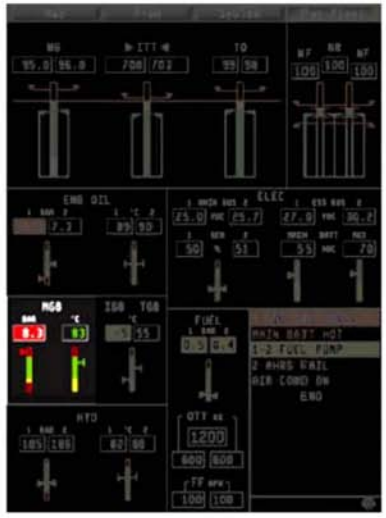
2 TEMPERATURE
The oil temperature values in the MGB are represented by digital readouts under the label °C. Graphically the values are represented on a vertical scale by means of a pointer (T symbol) that matches the color of the area on the scale.
Green band of the analogue vertical scale represents a normal condition for the hydraulic oil temperature and so the associated digital readout values in Celsius degrees.
The amber band represents a caution condition while the red band is associated to a warning condition.

The COMPOSITE FORMAT displays the values of pressure.

3 PRESSURE
The values of pressure in the MGB are represented by digital readouts aside the label MGB.



MFD CRUISE PAGE



MFD MAIN PAGE



COMPOSITE FORMAT





ICN-39-A-630000-G-A0126-02091-A-01-1

MAIN ROTOR DRIVE – INDICATIONS

MAIN ROTOR DRIVE – CONTROLS AND INDICATORS

1 CHIP BURNER push-button

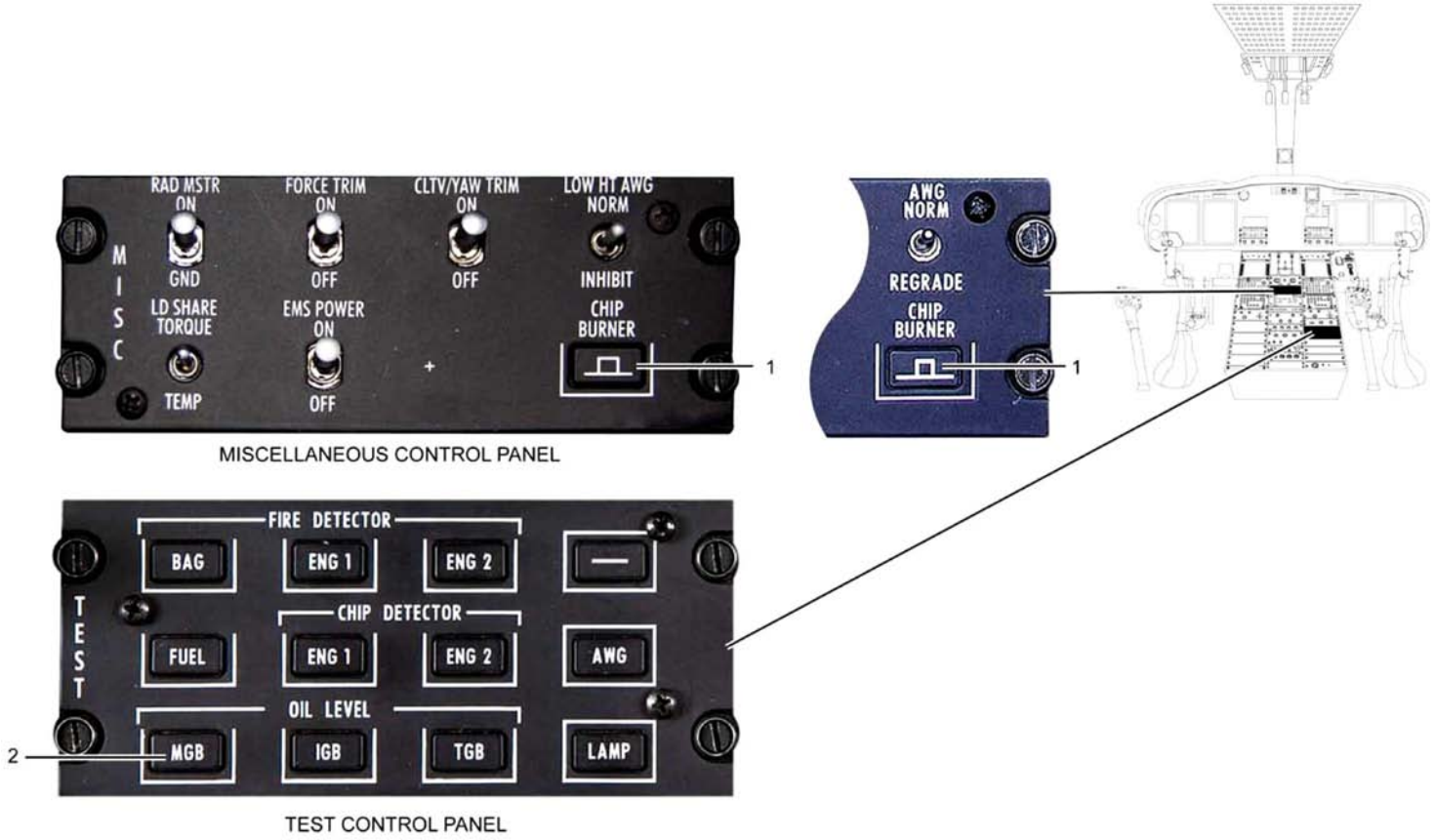
- PRESSED ..... a chip burning attempt is performed

2 OIL LEVEL MGB push-button (on the TEST panel)

- PRESSED ..... the caution MGB OIL LOW comes on in inverse video in the CAS window
  The test is possible only when the helicopter is on the ground and the NR is less than 2%

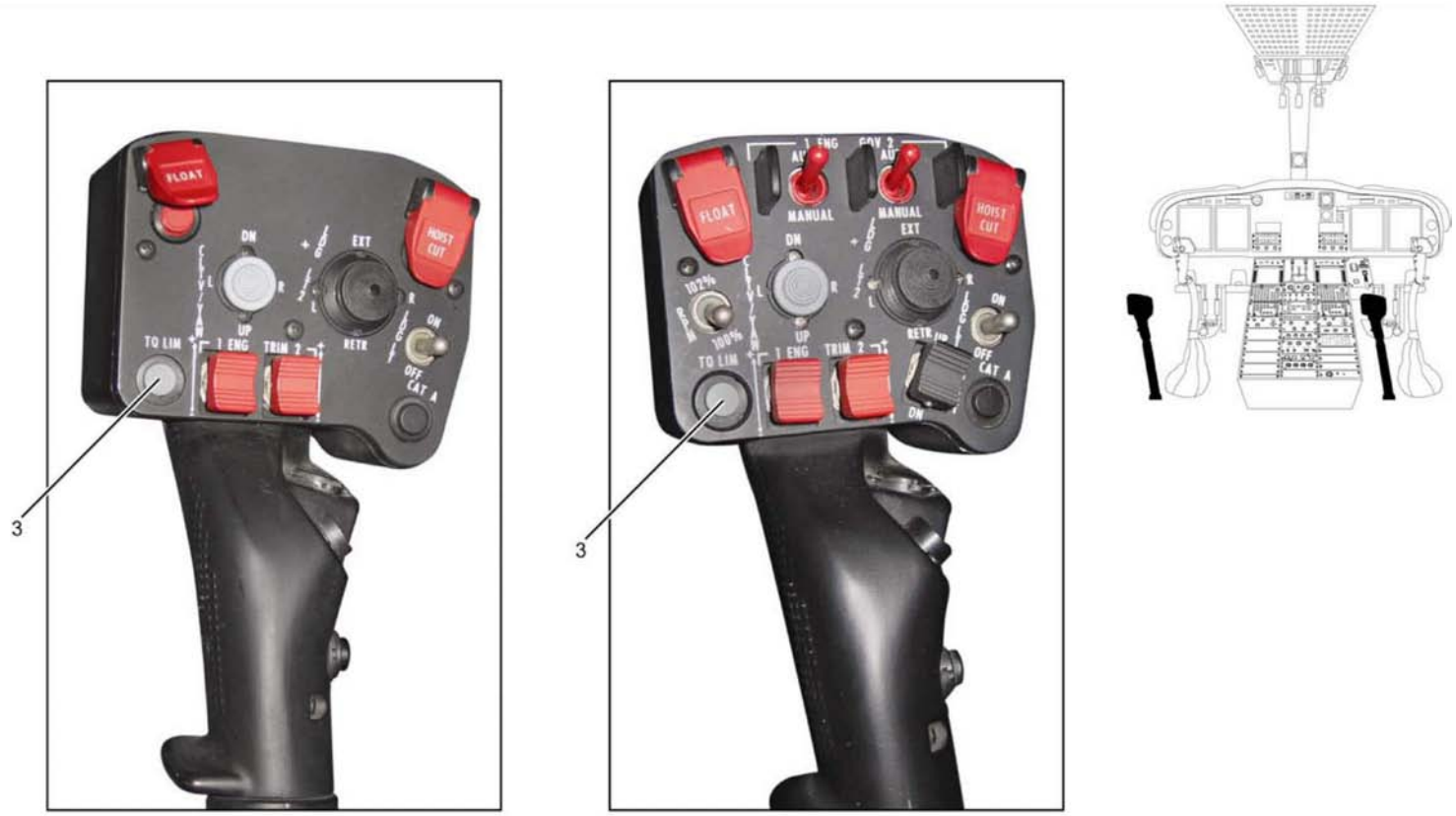
3 TQ LIM toggle switch

- PRESSED ..... engages / disengages the torque limiter



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MAIN ROTOR DRIVE – CONTROLS AND INDICATORS



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MAIN ROTOR DRIVE – CONTROLS AND INDICATORS

TORQUE LIMITER

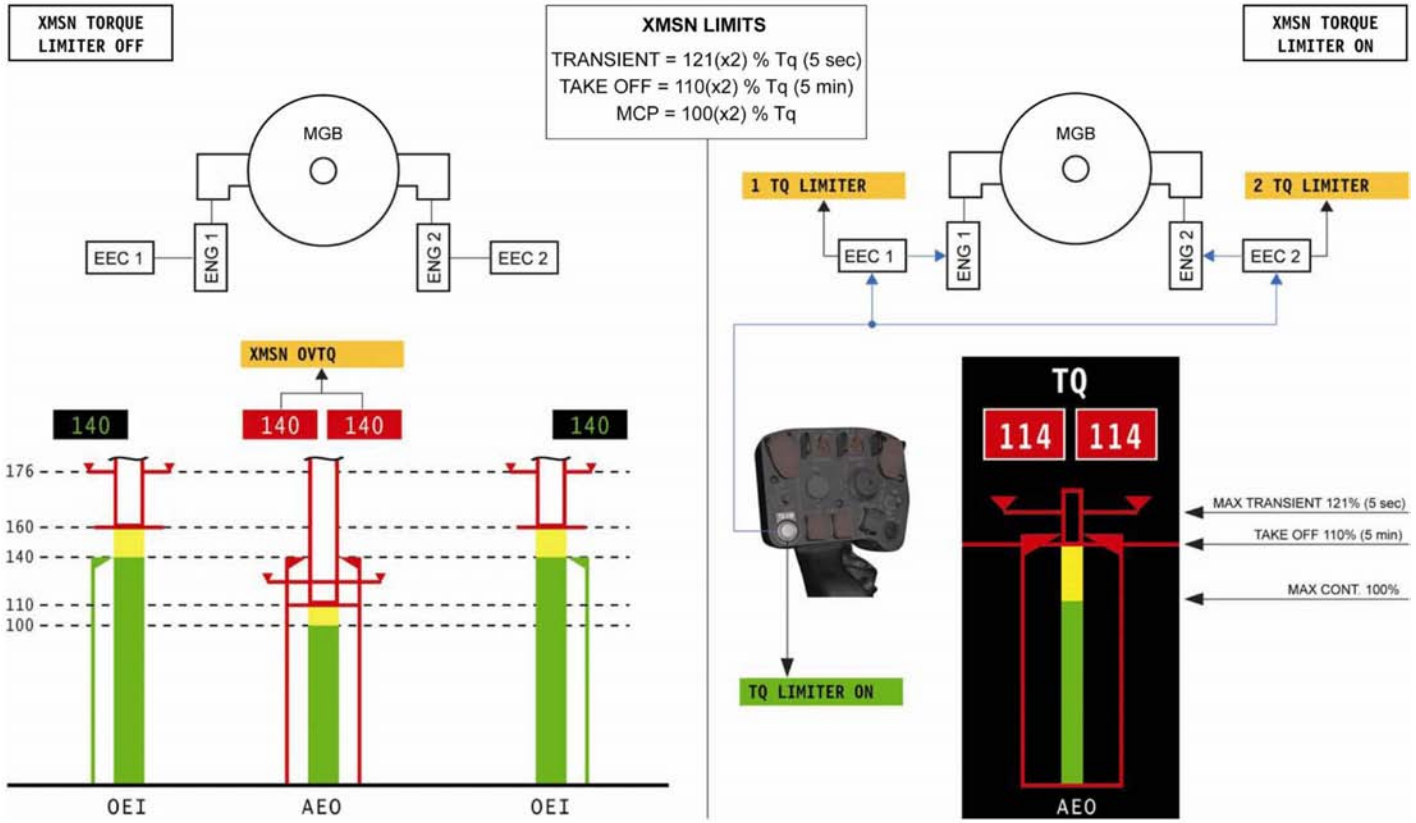
On the AW139 helicopter, the torque limiter function is set to OFF as default.

If the pilot wants to engage the torque limiter function, the TQ LIM push-button on the collective grip, must be pressed. After selection, the green advisory message TQ LIMITER ON will be displayed in the CAS window.

When the torque limiter function is set to ON, the AEO total torque will be limited to be combined torque value of TQ 228%. The OEI engine torque will not be affected.

When pressed a second time, the torque limiter function is deactivated and full power is available from engines.

To perform the OEI TRAINING, the TQ LIM function must be set to ON.



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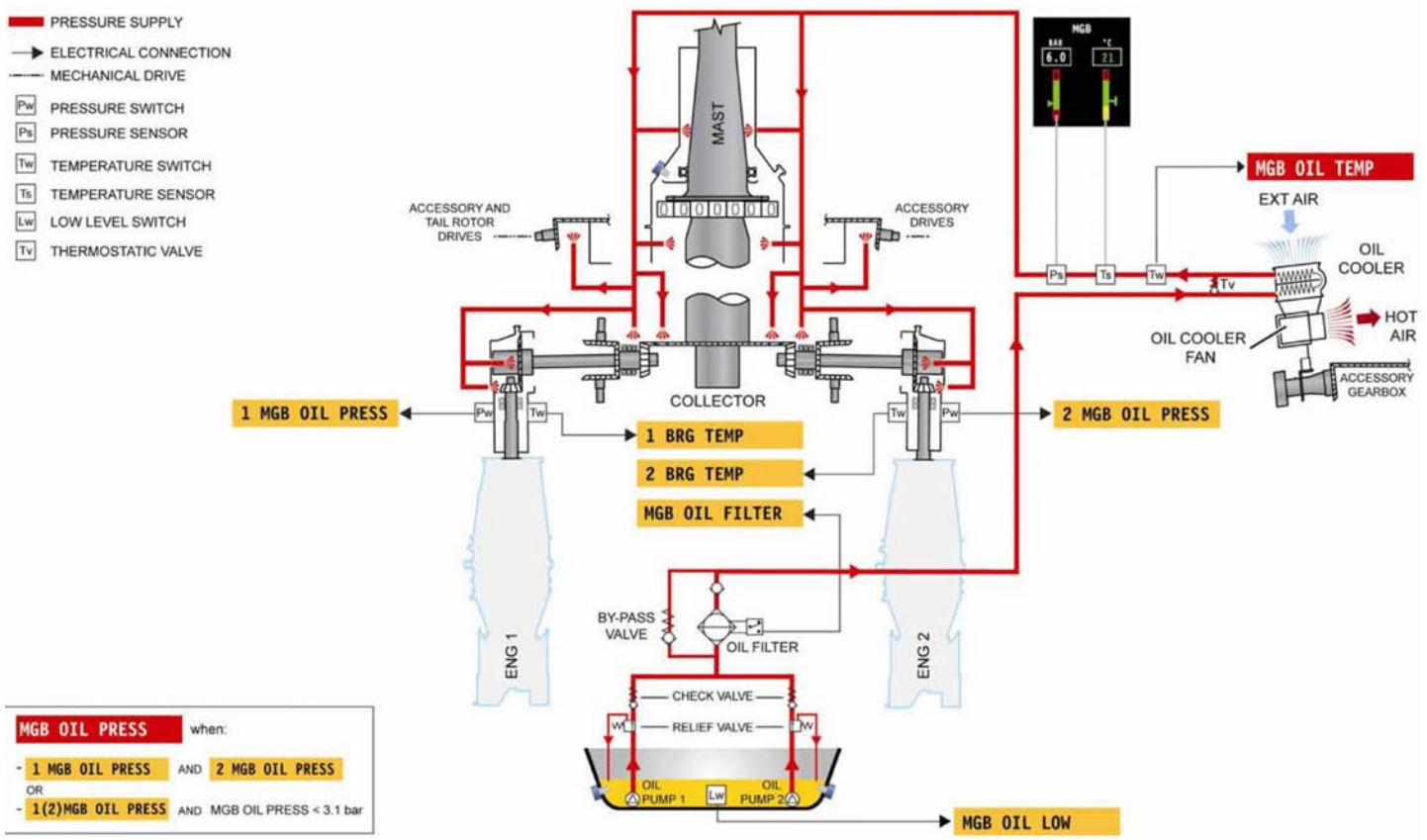
TRANSMISSION TORQUE LIMITS

MGB LUBRICATION

The MGB is lubricated by an oil system where all oil ducts are contained inside the MGB casing to avoid the possibility of oil leaks.
The lubricating system comprises two hydraulic pumps, check valves, a filter, an oil cooler and a fan which sends fresh air through the oil cooler. The lower part of the MGB case acts as a tank for the lubricating oil: in fact the lubrication pumps suck the oil from the sump and deliver it to the distribution system. The distribution of the lubricating oil is achieved by a series of calibrated nozzles (jets).
The outlet pressure of each pump is limited by a pressure relief valves that return the excess of oil to the oil sump. After the pressure relief valves the oil passes through a check valve and then to the oil filter.
The filter includes a by-pass and an impending bypass switch. The filter element will be bypassed if the pressure drop across the filter reaches a preset value.
An oil level sight glass is located in front of the MGB.
An oil low level sensor permits to have a caution message related to the oil level condition.
No indications are displayed in case of pump failure.

OIL LOW LEVEL SENSOR

The oil low level sensor is an optical sensor type which gives a signal to MAU 1 and MAU 2 when the oil level in the sump is less the minimum level.



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MAIN GEAR BOX LUBRICATING SYSTEM

DEBRIS WARNINGS SYSTEM (DWS)

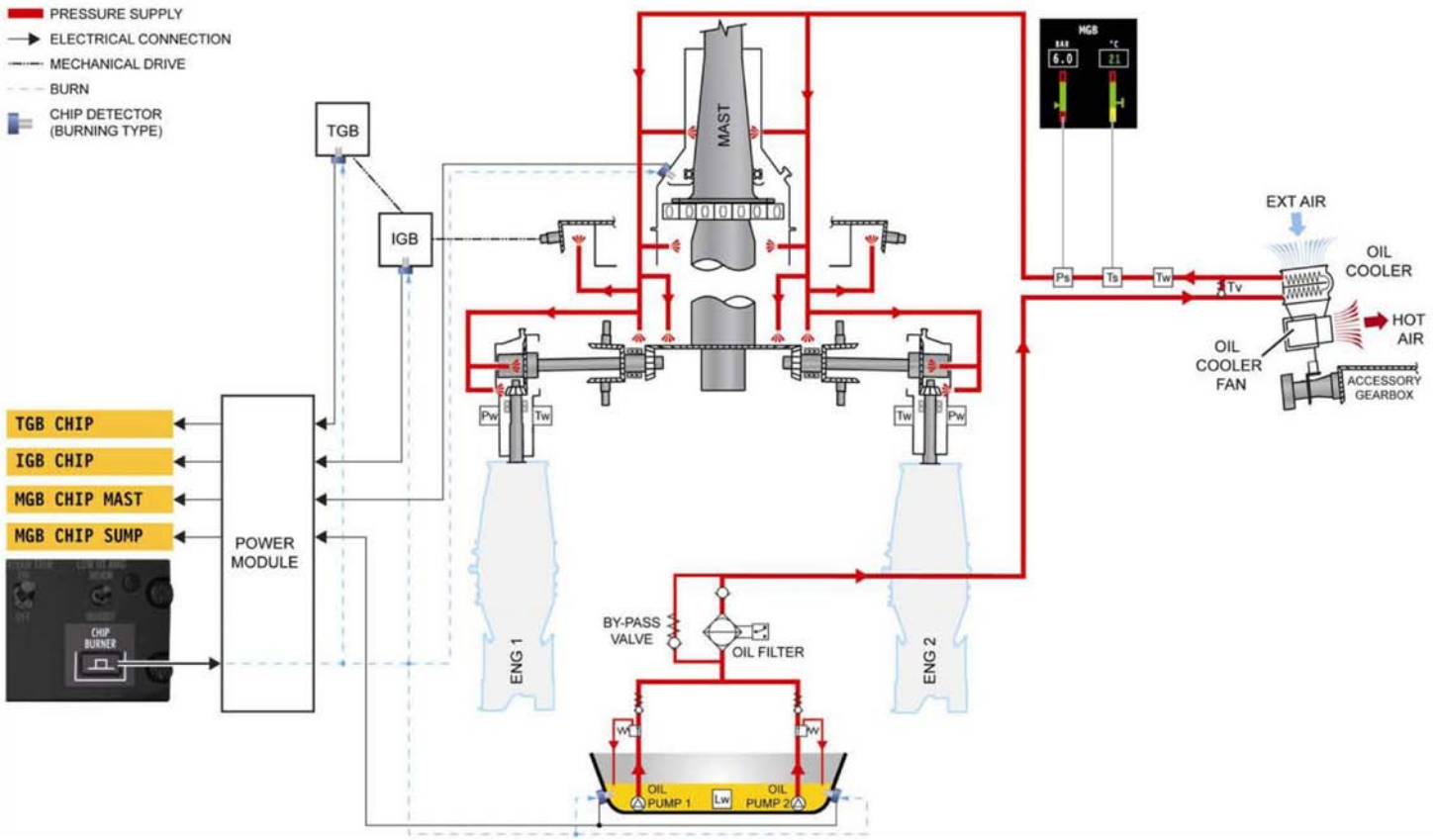
Five chip detectors and one Chip Detector Power Unit constitute the Debris Warnings System (DWS).
The purposes of the DWS are the followings:

- detect ferrous particles introduced into the lubricating oil by wear or failure of gears, bearing and other wetted parts
- burn the small ferrous particles detected

The DWS includes:

- a power module interfaced with the chip detectors, the two MAUs, the burn command on the MISC control panel;
- five chip detectors (three chip are located on the MGB, one chip on the Intermediate Gear Box IGB, and one chip on the Tail Gear Box TGB).

When the MGB CHIP MAST or MGB CHIP SUMP caution messages are shown, it is possible to try to burn the ferrous particles detected in the oil. To do this the pilot has to push the CHIP BURNER switch on the MISC panel.
If the caution extinguishes the burning is successful and the particle is classified as “small”. In this case the particle has been removed from the oil circuit.
In the opposite case the particle is considered “large” and cannot be burned.
If the caution message goes out (the particles are burned) the data that follow are written to the Non Volatile Memory (NVM) of the Central Maintenance System (CMS).



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DEBRIS WARNING SYSTEM

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SECTION 40 - INDICATING

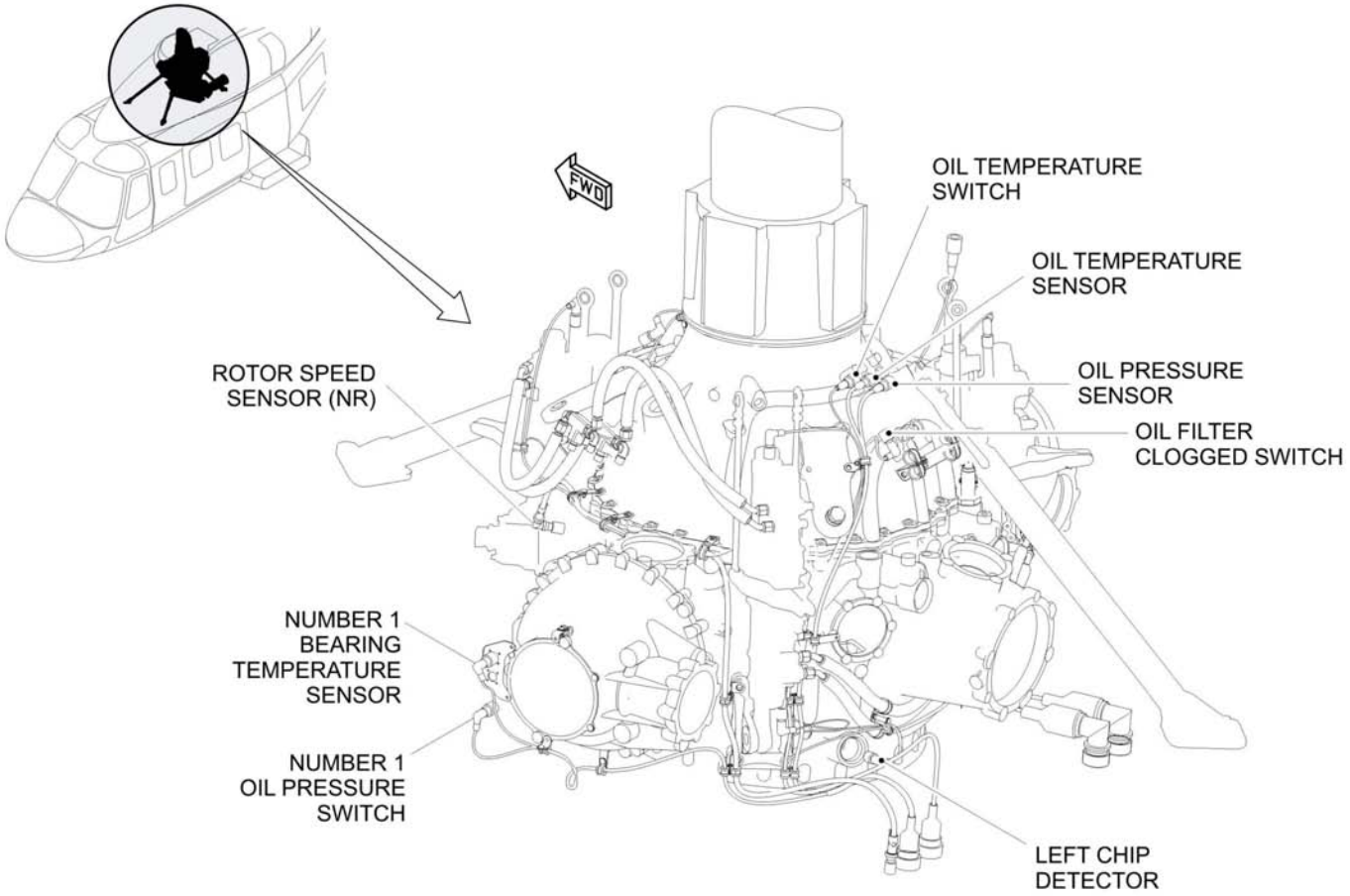
DESCRIPTION

The main gearbox (MGB) assembly is driven by two engines and drives the main rotor shaft at about 296 RPM and the tail rotor.
At each MGB input, a spring type freewheeling unit is installed allowing override of each engine in case of engine shutdown or failure, so that the rotor is continuously driven by the remaining engine.
The MGB is monitored by the following sensors:

- 3 chip detectors with a Chip Detector Power Unit
- (DEBRIS WARNINGS SYSTEM)
- 1 oil level sensor
- 1 oil pressure transducer
- 1 oil temperature transducer
- 1 oil filter switch
- 1 oil temperature switch
- 2 oil pressure switches
- 2 thermocouple sensors (Bearing)
- 1 magnetic speed sensor (NR).

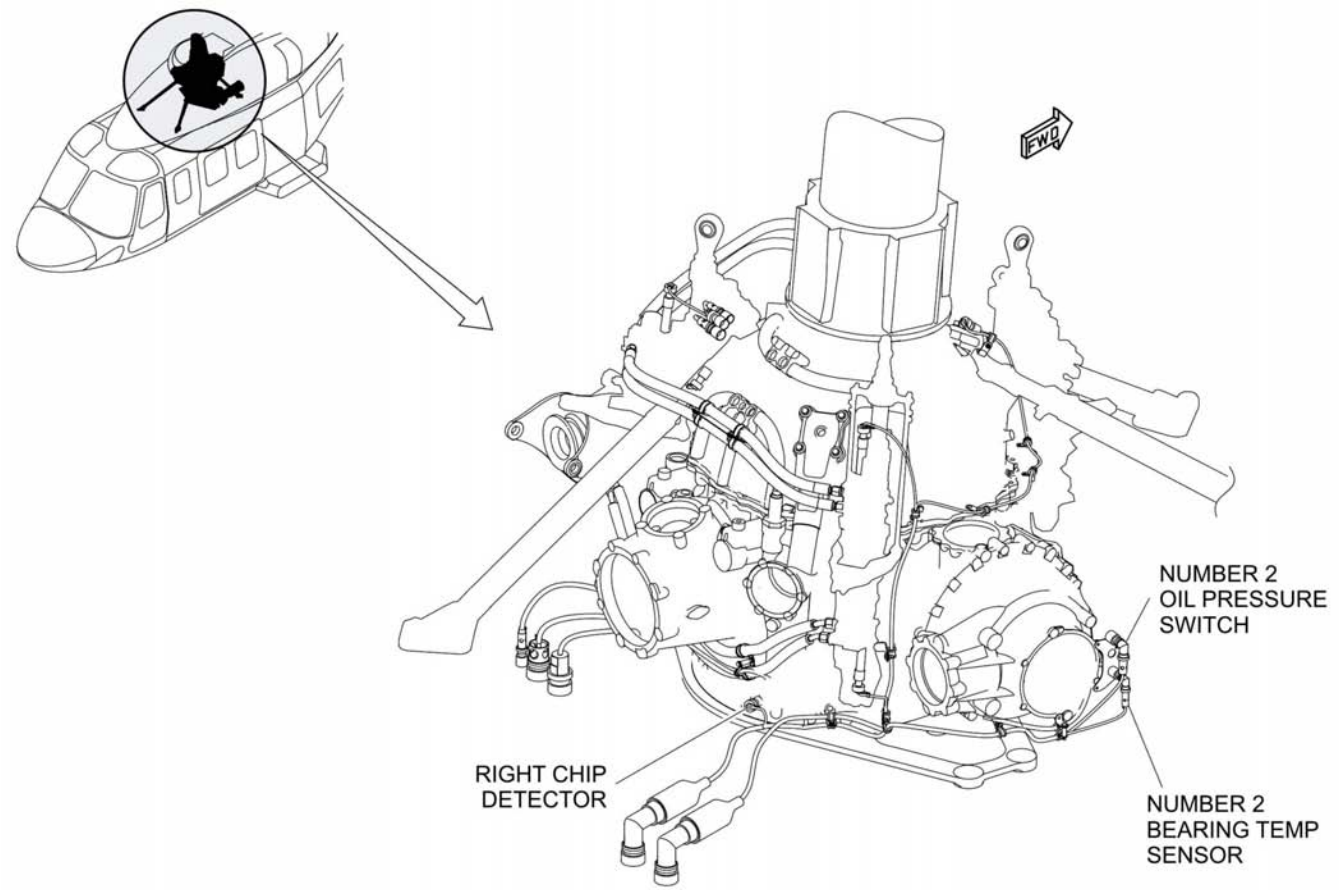
The DWS installed on the AW139 is formed by the 5 chip detectors, the power unit chip detector box, a data bus ARINC 429 to MAUs and a control panel for the chip burner.
The purpose of the power unit chip detector box is:

- to sense the detected ferrous debris introduced into the lubricating oil of the transmission system by wear or failure of gears, bearings and other oil wetted parts.
- to monitor the status of the five channels to determine if a short condition (CHIP) is being detected, if there is an open condition (FAULT) or if it is a normal condition and reports this to the MAU over the ARINC 429 interface.
- to burn the small particles. The burning pulse shall be delivered in sequence to the sensors having the chip.
- to self test the internal circuitry and chip sensors automatically.
- to test the chip sensors manually through an external switch on the box. The test consists of a continuity test resistor of 30 kΩ that is provided inside each sensor across pins A and B as close as possible to the detector gap.
  This resistor allows the testing and trouble shooting to check the continuity of the cabling, connector and wire receptacle of each individual sensor without need of accessing it.



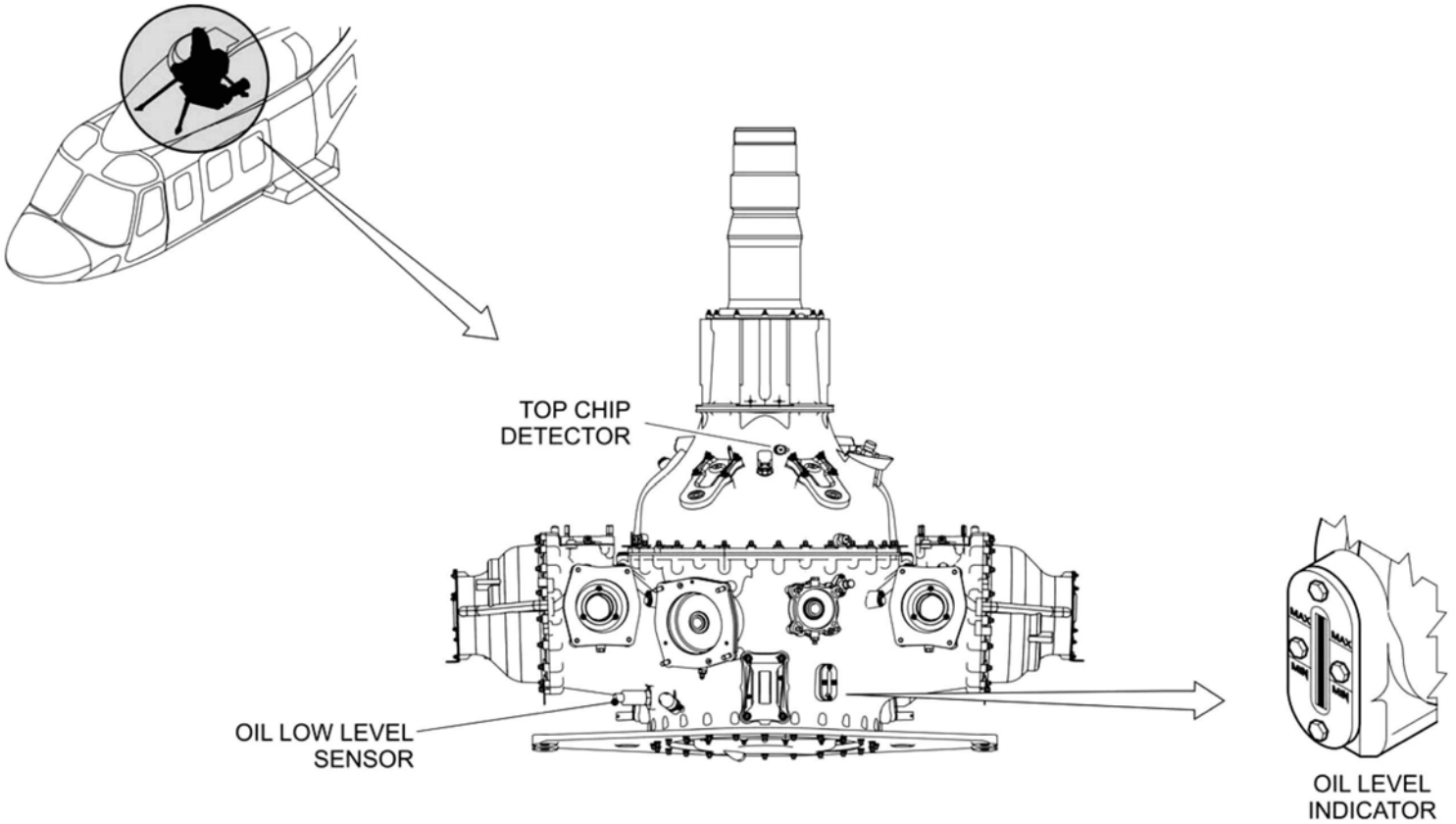
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MAIN GEARBOX INDICATING COMPONENT LOCATION (SHEET 1 OF 5)



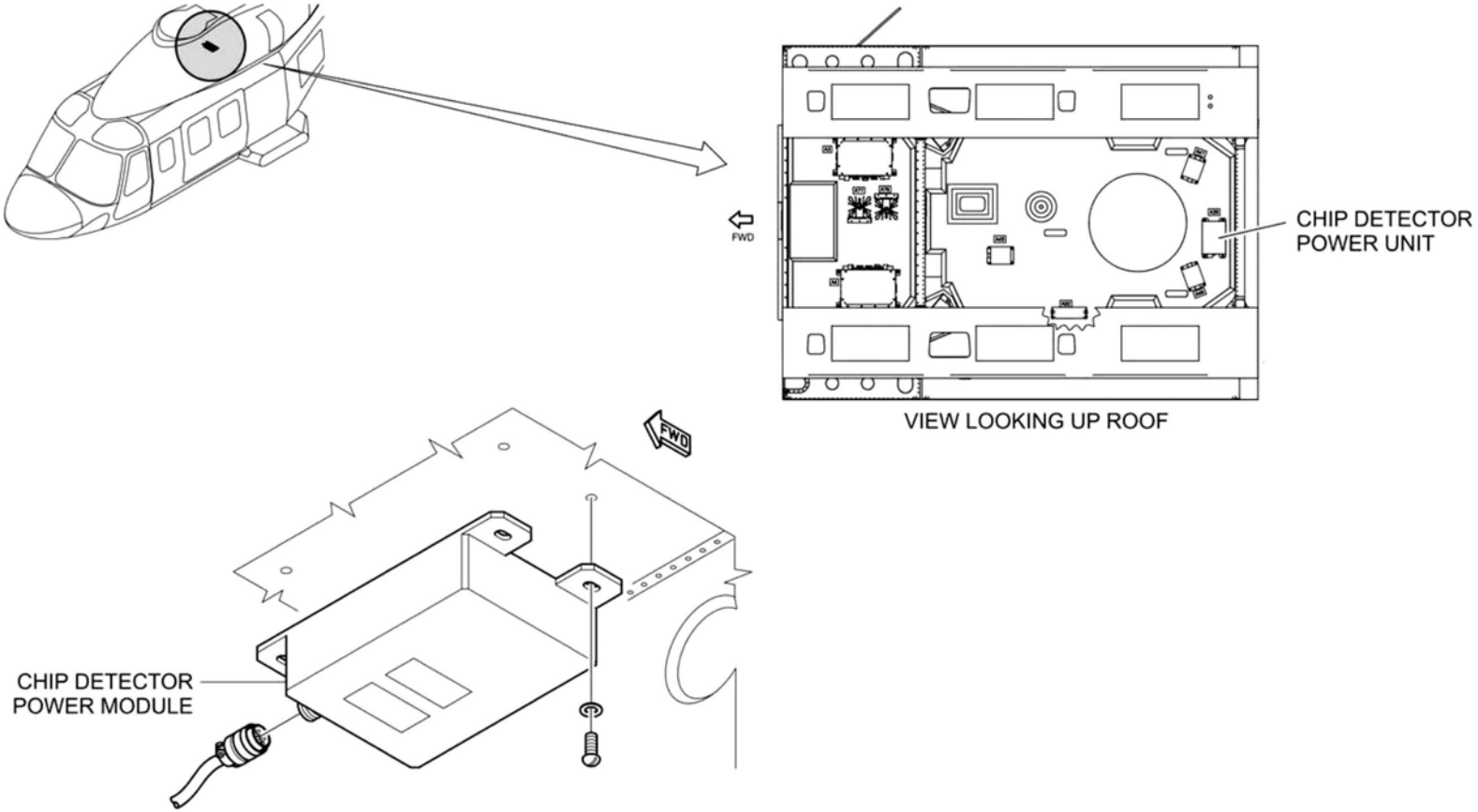
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MAIN GEARBOX INDICATING COMPONENT LOCATION (SHEET 2 OF 5)



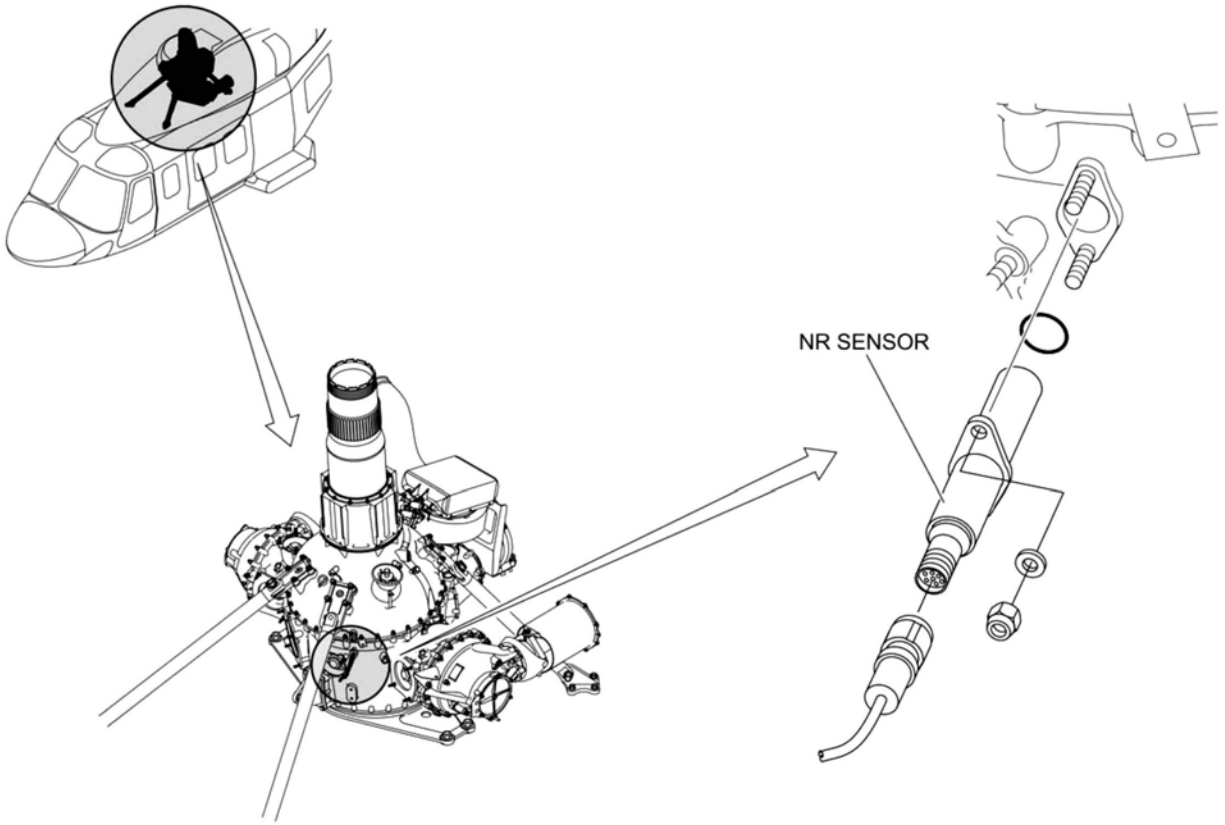
ICN-39-A-634000-G-A0126-00883-A-01-1

MAIN GEARBOX INDICATING COMPONENT LOCATION (SHEET 3 OF 5)



ICN-39-A-634000-G-A0126-00884-A-01-1

MAIN GEARBOX INDICATING COMPONENT LOCATION (SHEET 4 OF 5)

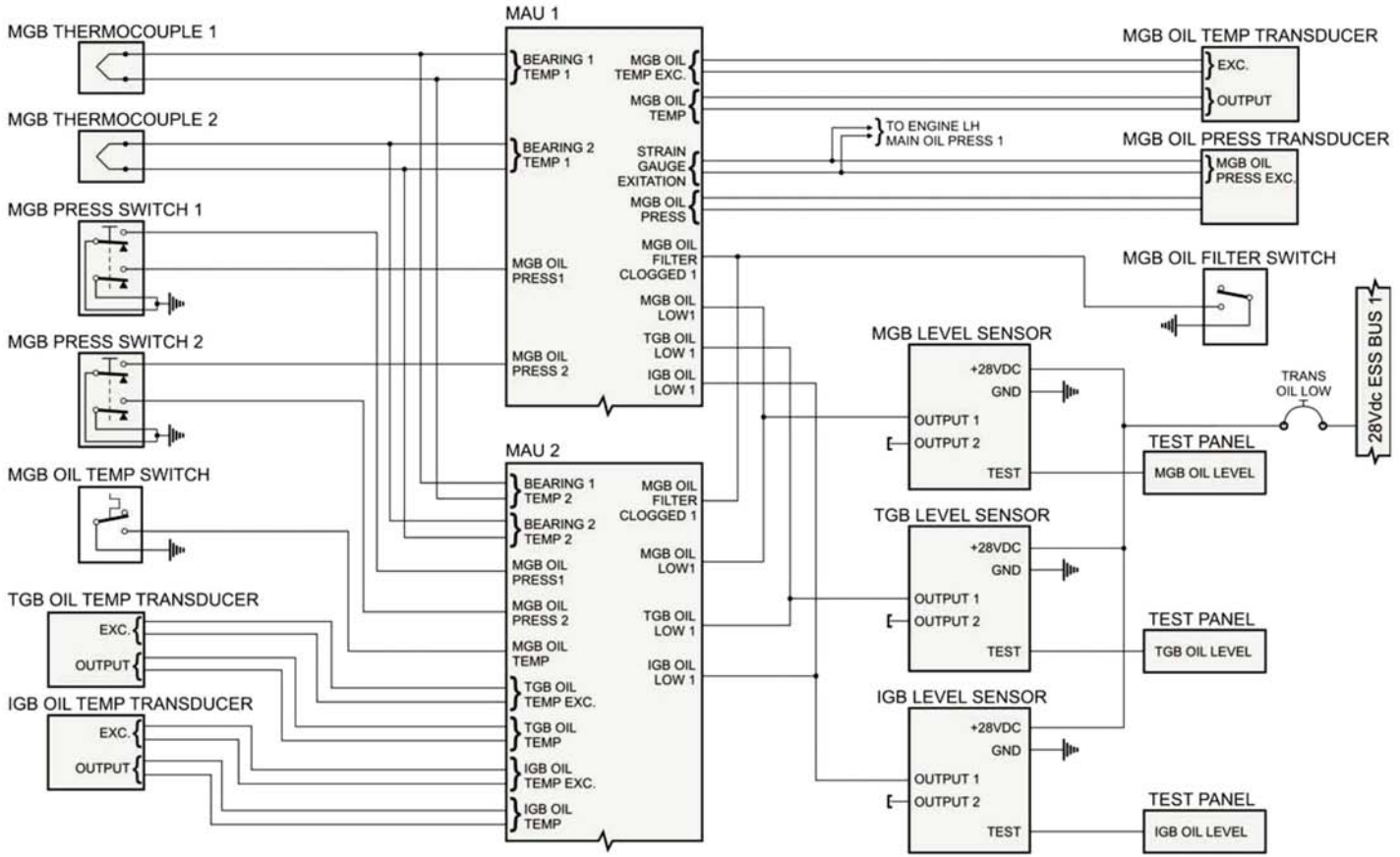


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MAIN GEARBOX INDICATING COMPONENT LOCATION (SHEET 5 OF 5)

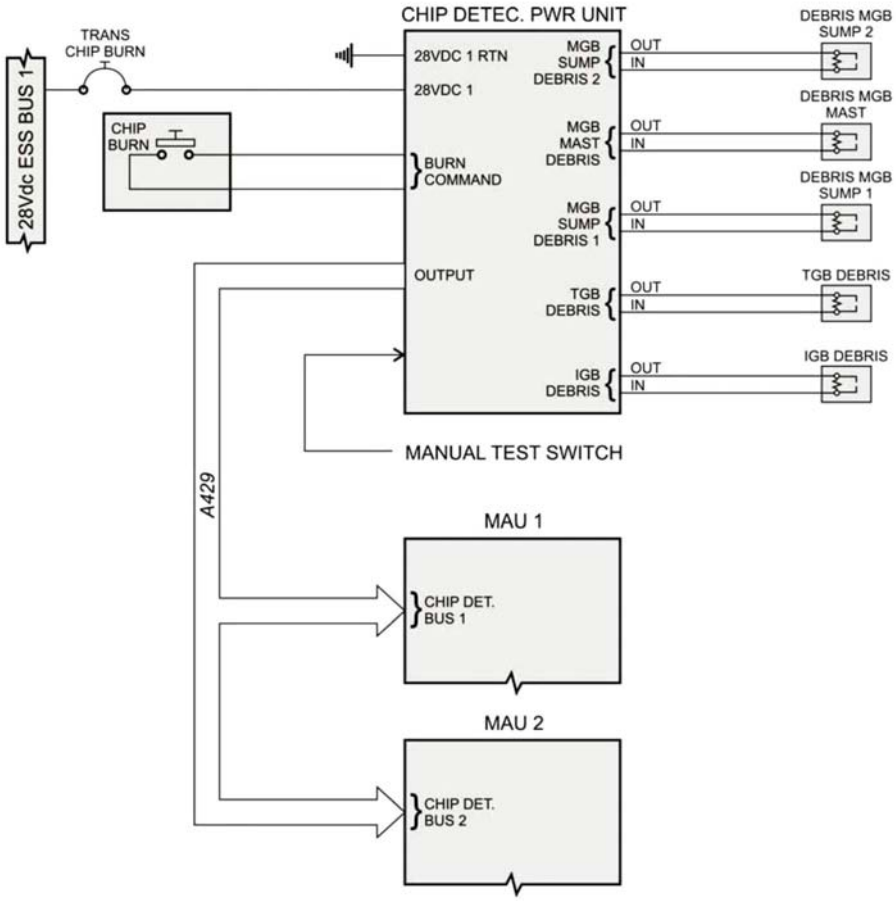
OPERATION

On the MGB three chip detectors are fitted: two in the sump and one close to the mast bearing.
When a particle is detected by any of the sensors, the relevant caution alarm is generated.
For small particles a burning attempt can be performed by the crew through the manual use of a push-button fitted on the miscellaneous control panel.
If the signal is extinguished (burning successful), the relevant information of small particle captured together with the date and time, sensor name, number of burning attempts are stored in the NVM.
The debris warning system can be tested before flight by the operator through a switch located on the Power Unit Chip Detector box.
The oil level sensor is an optical sensor type which provides an electrical signal when the oil level in the reservoir and sump is below a predetermined minimum level.
The MAUs generate the caution alarm on the display.
During the pre-flight check, the flight crew uses the test panel to perform the MGB oil level test.
The oil pressure transducer is a strain-gauge type that provides an analog input to the MAU NO.1.
When the pressure drops below a predetermined value, the warning / caution alarm will be represented on the MFD.
The oil temperature transducer is a thermo-resistance type that provides an analog input to the MAU N.1.
When the temperature rises above a predetermined value, the warning alarm will be represented on the MFD.
An oil filter with a differential pressure sensor, microswitch type, senses the differential pressure between the filter input and output and the MAUs generate this caution alarm that is represented on the MFD.
The oil temperature switch senses the temperature of the MGB oil circuit and when it reaches a predetermined value, the MAU NO.2 will generate a warning alarm on the MFD.
The two oil pressure switches are installed to sense the low pressure of the MGB and when one of the oil sensor pressure switch pressure switch senses the minimum value, the MAUs generate the caution alarm on the MFD.
When the pressure value reaches the minimum value and the two oil pressure switches sense this minimum value, the warning alarm will be represented on the MFD.
There are two thermocouple sensors installed on each MGB drive bearing 1 and 2.
The sensing probes are metal sheathed type K thermocouples and are held nearly in contact with the measured part by means of a helical spring.
When the operating temperature range increases to a predetermined value the MAUs generate the caution alarm on the MFD.
The N_R speed sensor is a triple coil magnetic speed sensor and provides 3 independent frequency outputs directly related to the speed of the rotor gear to the MAUs and EECs.
The N_R value is represented on the triple tachometer on the MFD and the PFDs.



ICN-39-A-634000-G-A0126-00886-A-01-1

MGB-IGB-TGB/INDICATING ELECTRICAL SCHEMATIC (SHEET 1 OF 2)



ICN-39-A-634000-G-A0126-00887-A-01-1

MGB-IGB-TGB/INDICATING ELECTRICAL SCHEMATIC (SHEET 2 OF 2)

CONTROLS AND INDICATORS

The main gearbox (MGB) controls are located on the Miscellaneous Panel and on the Test Control Panel both installed in the cockpit console.
The MFD (Main and Cruise Page) displays the MGB OIL TEMP/PRESS parameters.
The MFD also provides the malfunction condition in the CAS message window #1.
The Miscellaneous Panel provides the following function:

1 CHIP BURNER: ..... when pressed a burning attempt for the MGB CHIP MAST or MGB CHIP SUMP is performed.

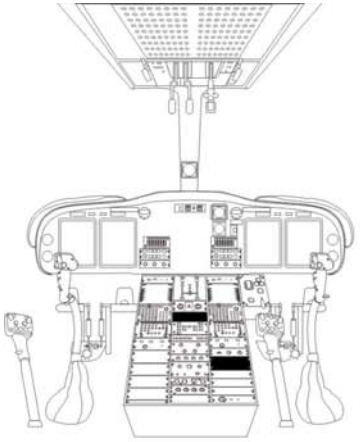
The Test Control Panel provides the following function:

2 MGB OIL LEVEL

- Push button test switch: ….. when pressed the caution MGB OIL LOW in inverse video is displayed in the CAS window.

NOTE A

This test is operative with the A/C on ground and NR < 2%.



NOTE: ACTIVE ONLY WITH ENGINE SHUT DOWN AND NR BELOW 2% (ON GROUND ONLY)



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MAIN GEARBOX PANEL

CAPTIONS

The warning provided in the CAS is as follows:

1 MGB OIL PRESS: ..... when the MGB oil pressure in the lubricating system is below 3.1 bar; resets above 3.4 bar.

2 MGB OIL TEMP: ..... when the MGB oil temperature in the lubricating system is above 109°C; resets below 97°C.

NOTE B

When one of this warning is detected, besides the MWL activation, the AWG shall provide the following:

- No tone + WARNING – WARNING (aural message)

This sequence is transmitted once only. This message has priority number 5.

The caution provided in the CAS message window is as follows:

3 1(2) MGB OIL PRESS: ..... when the oil pressure in the MGB – Engine 1(2) input stage is below 3.1 bar; resets above 3.4 bar.

NOTE C

With the A/C both on ground and flight the warning MGB OIL PRESS will suppress the caution 1-2 MGB OIL PRESS.

4 MGB CHIP MAST: ..... when inside the MGB oil lubricating system a chip is detected on the MAST bearing

5 MGB CHIP SUMP: ..... when inside the MGB oil lubricating system a chip is detected in the SUMP.

6 CHIP MAST FAIL: ..... when the channel relative to the CHIP MAST detector system is sensed faulty.

7 CHIP SUMP FAIL: ..... when the channel relative to the CHIP SUMP detector system is sensed faulty.

8 MGB OIL FILTER: ..... when the differential pressure between filter input and output exceeds 1.25 bar.

9 MGB OIL LOW: ..... when the MGB oil level is at minimum or during the MGB OIL LEVEL TEST.

10 1(2) BRG TEMP: ..... when the MGB-Engine 1(2) input stage bearing temperature is above 140°C; resets below 135°C.

11 AVIONIC FAULT: ..... when the loss of communications to a single MAU is detected for the CHIP DETECTOR POWER UNIT. The maintenance message in the CMC is A429/RS 422 BUS.

NOTE D

In case of double malfunction, caution message 1-2 BRG TEMP will replace the 1 BRG TEMP and 2 BRG TEMP.

12 XMNS OVTQ

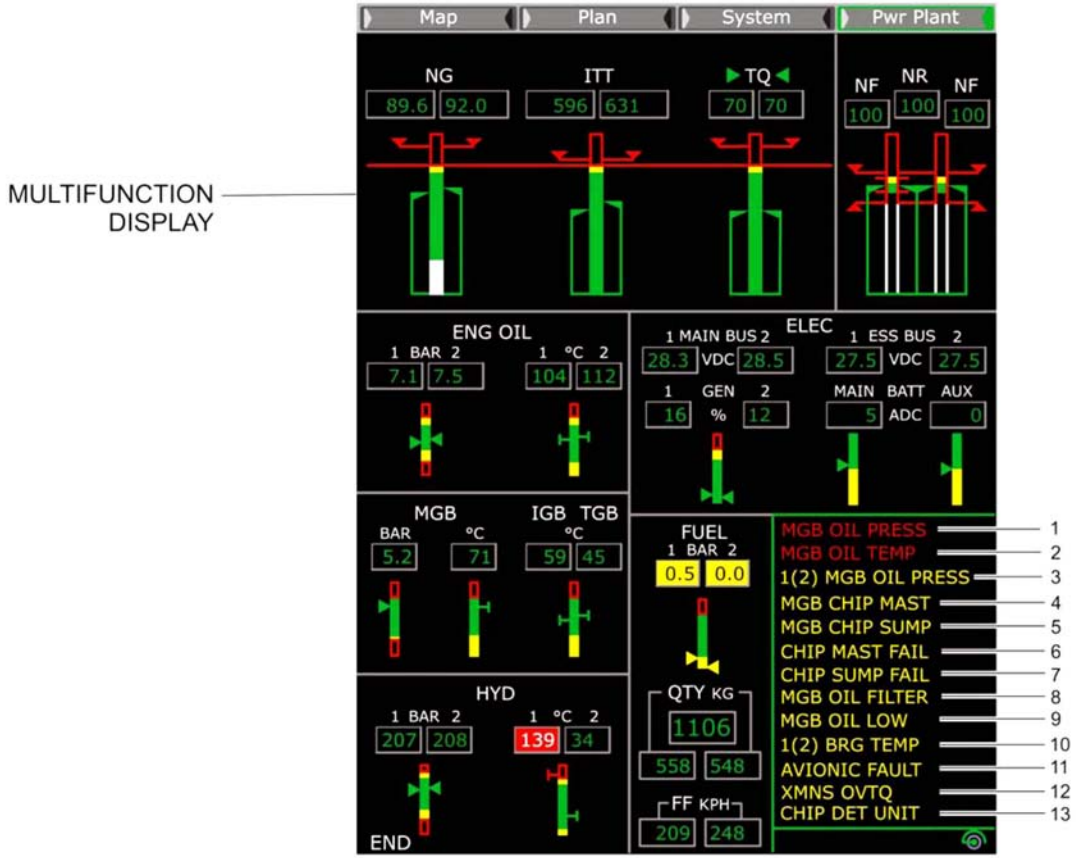
- AEO: ..... when in AEO condition, the TQ1> 110% or TQ2 > 110%.

- OEI: ..... when in OEI condition, the TQ1> 160% or TQ2 > 160%.

NOTE E

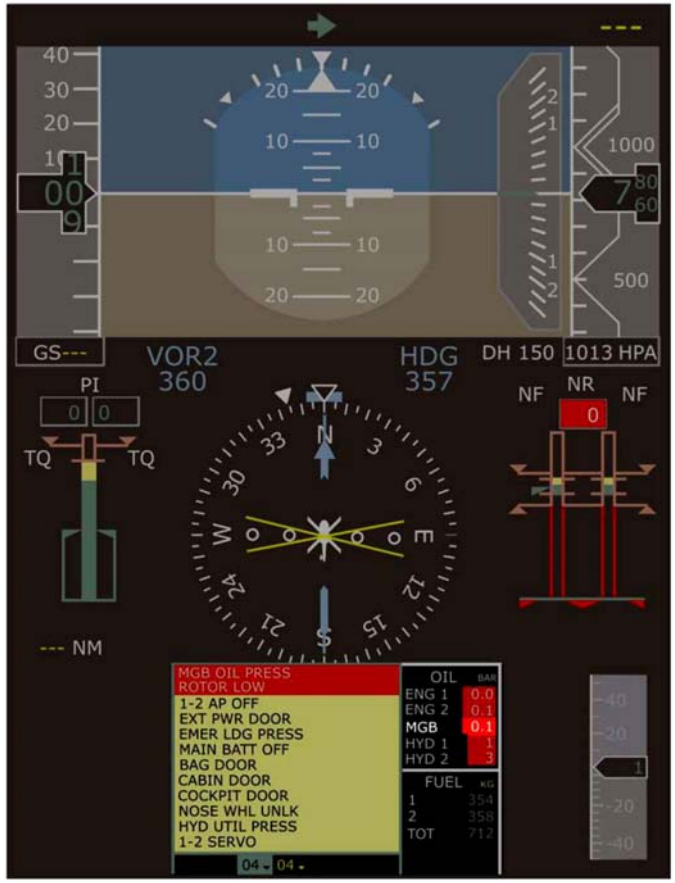
This condition is set after 1 sec from the detection; is reset after 300 msec from when the malfunction is removed.

13 CHIP DET UNIT: ..... when the Chip Detector Power Unit does not respond to the power up built in test or continuous built in test or responds to the manual test when the push-button switch on the CDPU unit is pressed.



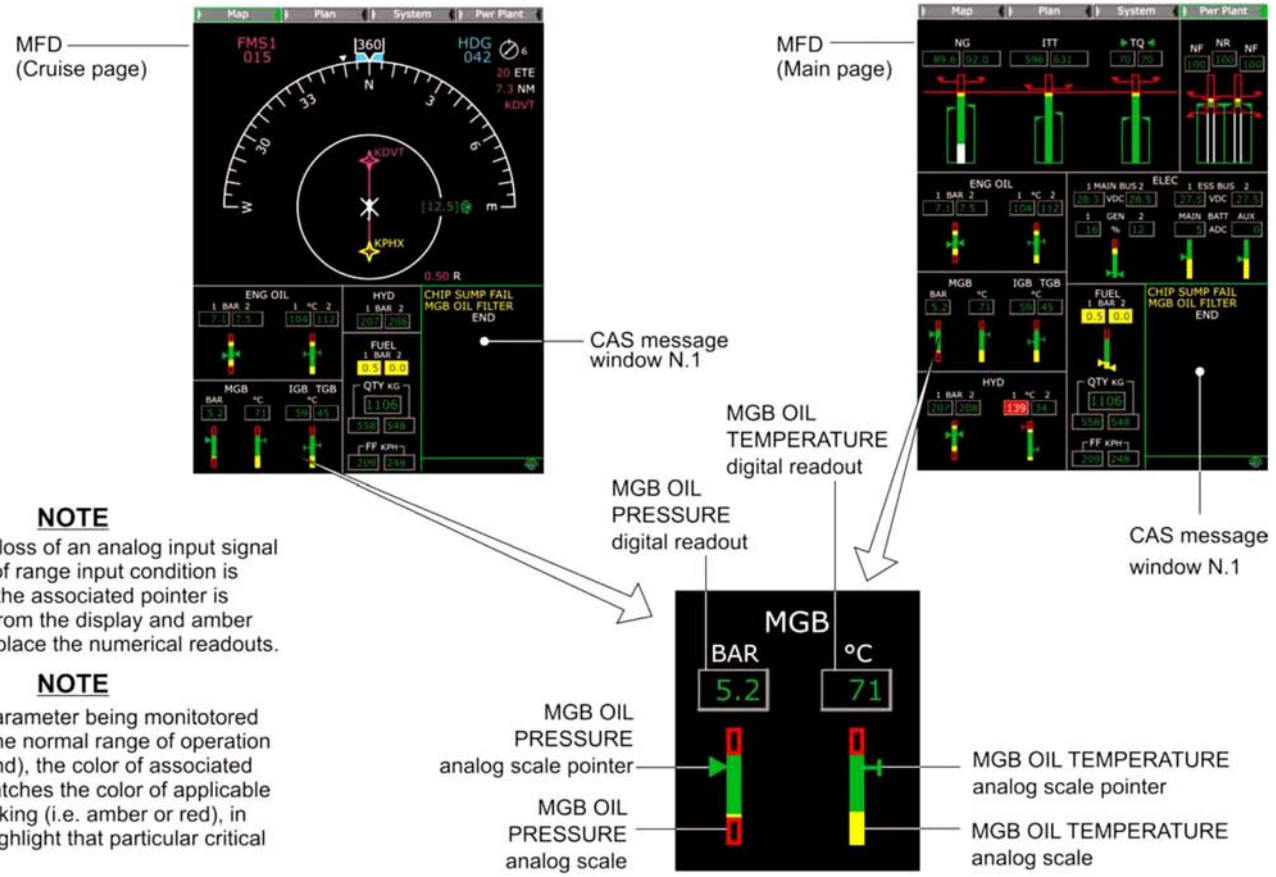
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MAIN GEAR BOX INDICATING (SHEET 1 OF 3)



ICN-39-A-634000-G-A0126-03045-A-01-1

MAIN GEARBOX INDICATING (SHEET 2 OF 3)



NOTE

When the loss of an analog input signal or an out of range input condition is detected, the associated pointer is removed from the display and amber dashes replace the numerical readouts.

NOTE

When a parameter being monitotored exceeds the normal range of operation (green band), the color of associated pointer matches the color of applicable range marking (i.e. amber or red), in order to highlight that particular critical condition.

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MAIN GEARBOX INDICATING (SHEET 3 OF 3)

FUNCTION	RANGE	DIGITAL READOUT COLOR MARKINGS	ANALOG SCALE COLOR MARKINGS
MGB OIL PRESS digital readout	0 to 10.0 bar	see Note	
MGB OIL TEMP digital readout	-50 to 150°C	see Note	
MGB OIL TEMP analog scale	110 to 150°C		Red band
	110°C		Red line
	1 to 109°C		Green band
	-50 to 0°C		Amber band
MGB OIL PRESS analog scale	6.1 to 10.0 bar		Red band
	6.1 bar		Red line
	3.1 to 6.0 bar		Green band
	2.3 to 3.0 bar		Amber band
	2.2 bar		Red line
	0 to 2.2 bar		Red band

NOTE F

In normal condition the background is black and the numerals green; the digital readout features an inverse video technique when an exceedance occurs.
When the parameter is in amber or red band, the digital display background turns amber or red respectively and the numerals are black or white respectively.

CAS WARNING MESSAGES	
CAS CAPTION	**FAILURE DESCRIPTION**
MGB OIL PRESS	Oil pressure below limits at one or both engine MGB inputs and in MGB oil system. (less than 3.1 bar) (resets above 3.4 bar) *– voice warning: WARNING-WARNING repeated once*
MGB OIL TEMP	MGB oil temperature above limit (greater than 109°C) (resets below 97°C) *– voice warning: WARNING-WARNING repeated once*

CAS CAUTION MESSAGES	
CAS CAPTION	**FAILURE DESCRIPTION**
XMSN OVTQ	Transmission TQ limit exceeded, Take Off rating 5 min limit (either engine above 110%) or 2.5 min OEI limit (160%)
MGB CHIP MAST MGB CHIP SUMP	Activated CHIP burner (on MISC control panel) It is permitted to activate the CHIP BURNER up to 3 times to clear a chip in one flight. On the 4th CHIP caution Land As Soon As Practicable
MGB OIL FILTER	MGB oil filter blockage and in bypass (differential pressure over filter exceeds 1.25 bar)
MGB OIL LOW	MGB oil level low (caution only active with engines shut down and NR below 2%)
1(2) MGB OIL PRESS	oil pressure at MGB input 1 or 2 low (less than 3.1 bar) possible blockage in oil duct to engine NOTE G If both 1 MGB OIL PRESS and 2 MGB OIL PRESS are displayed, then the red warning MGB OIL PRESS will come on and suppresses the two previous cautions

CAS CAUTION MESSAGES	
CAS CAPTION	**FAILURE DESCRIPTION**
CHIP DET UNIT CHIP DET TEST	Chip detect unit malfunction
CHIP MAST FAIL CHIP SUMP FAIL	Associated gearbox chip sensor failed
1(2) BRG TEMP	Associated MGB engine input bearing over temperature
1(2) TQ LIMITER	Associated engine torque limiter system not functioning

CAS ADVISORY MESSAGES	
CAS CAPTION	**MESSAGES DESCRIPTION**
TQ LIMITER ON	Torque limiter activated

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SECTION 50 - ROTOR BRAKE

ROTOR BRAKE – GENERAL

The Rotor Brake (RB) is described in SECTION 5 – OPTIONAL EQUIPMENT SUPPLEMENTS of the AW139–RFM–4D.
The Rotor Brake allows:

- a rapid deceleration of the rotors after engines shut down
- to maintain the rotors stopped up to 8 hours
- the manual restore of the parking pressure

ROTOR BARKE – MAIN COMPONENTS

ROTOR BRAKE CONTROL LEVER (RBCL)

The pilot manually operates the Rotor Brake Control Lever for the selection of one of the following positions:

- OFF (RB release)
- PUMPING from forward detent and backward stop (pumping limit)
- BRAKE

The RBCL transmits mechanically the control to the Rotor Brake Control Module (RBCM). The trigger with the spring lets the pilots to unlock the lever and move it between the OFF and BRAKE position.

ROTOR BRAKE CONTROL MODULE (RBCM)

The Rotor Brake Control Module (RBCM) transforms the action of the pilot in hydraulic power of the RB circuit. The RBCM includes:

- a dual-stage hydraulic pump that supplies the necessary pressure and fluid displacement;
- an outlet valve which discharges the pressure in the hydraulic reservoir;
- an hydraulic accumulator which provides the hydraulic fluid displacement during the brake application;
- a shut-off valve which manages the RB circuit;
- an interlock pressure switch which detects the pressure in the circuit;
- a pressure transducer which provides monitoring of pressure values and gives a visual indication on the RBPI.

ROTOR BRAKE PRESSURE INDICATOR (RBPI)

The Rotor Brake Pressure Indicator (RBPI) is installed inside the cockpit and gives:

- rotor brake pressure indications (on the display),
- the position of the caliper.

ROTOR BRAKE ASSEMBLY (RBA)

The Rotor Brake Assembly (RBA) comprises the caliper, the pads and the pistons.
The RBA provides for the braking action when pistons are hydraulically actuated to clamp the rotating disc bolted to the tail output drive shaft flange.

ROTOR BRAKE ACTUATION ASSEMBLY (RBAA)

The Rotor Brake Actuation Assembly (RBAA) provides for to move and to maintain the RBA caliper on its operative UP and DOWN positions.

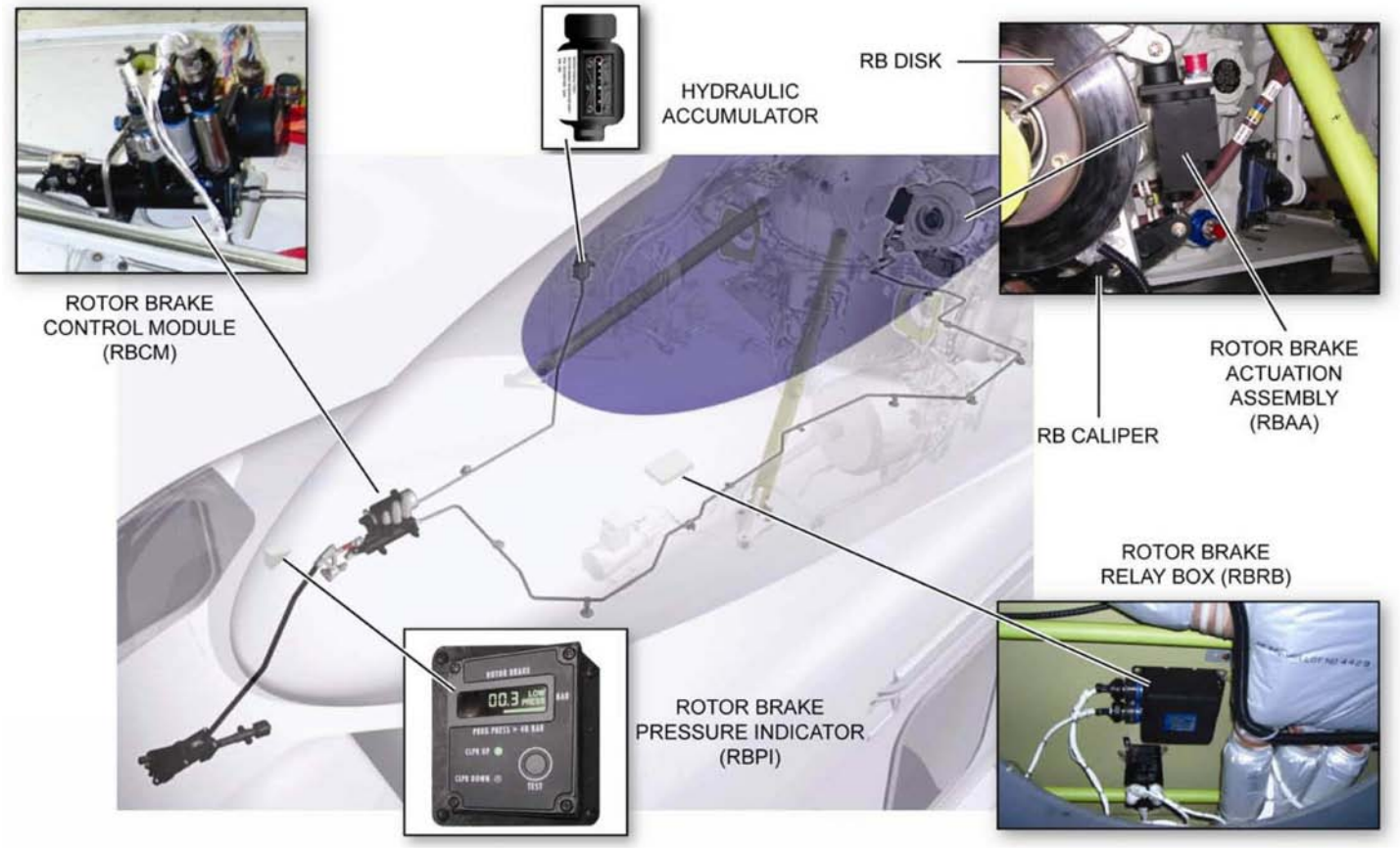
ROTOR BRAKE RELAY BOX (RBRB)

The Rotor Brake Relay Box (RBRB) controls the UP and DOWN movements of the caliper according to this logic:

- during flight the caliper is maintained in DOWN position far from the rotor brake disc;
- on the ground with the engines OFF the caliper is moved in UP position.

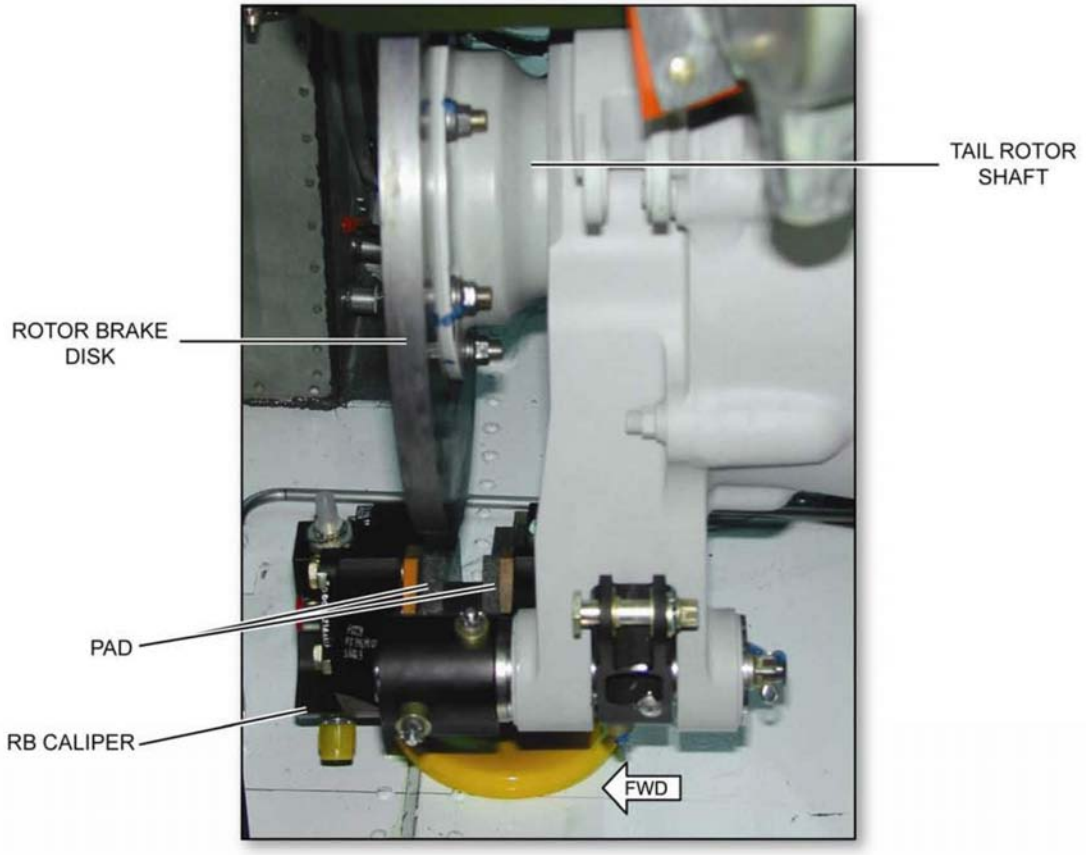
The electrical signals used for the braking action are provided by:

- the Rotor Brake Control Lever position;
- the Rotor Brake Actuation Assembly UP/DOWN position;
- the engine mode selector position;
- the Weight–On–Wheels (WOW) microswitch.



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ROTOR BRAKE – MAIN COMPONENTS (SHEET 1 OF 2)



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ROTOR BRAKE – MAIN COMPONENTS (SHEET 2 OF 2)

ROTOR BRAKE – CONTROLS AND INDICATORS

1 ROTOR BRAKE CONTROL LEVER (RBCL) at the right side of the overhead panel

- OFF ..... the rotor brake is released
- BRAKE .....
  - NORMAL BRAKING
    Allows to pressurize the system at 26-28 bar for dynamic braking by filling the accumulator
  - PARKING
    to perform a pumping action by moving the lever between PUMPING and BRAKING positions, increases the pressure in the circuit and in the accumulator up to a maximum parking pressure of 46.5 bar
- PUMPING action ..... (forward detent and backward step) permit to increase the pressure inside the system

NOTE A

The minimum parking brake pressure is 26 bar and may be achieved after 8 hrs. Below this pressure is possible to restore the correct one in the circuit, pumping again

2 ROTOR BRAKE PRESSURE INDICATOR (RBPI) at the right side of the overhead panel

3 CLPR DOWN ..... green led. Lit when the caliper is down; flashes during the transition

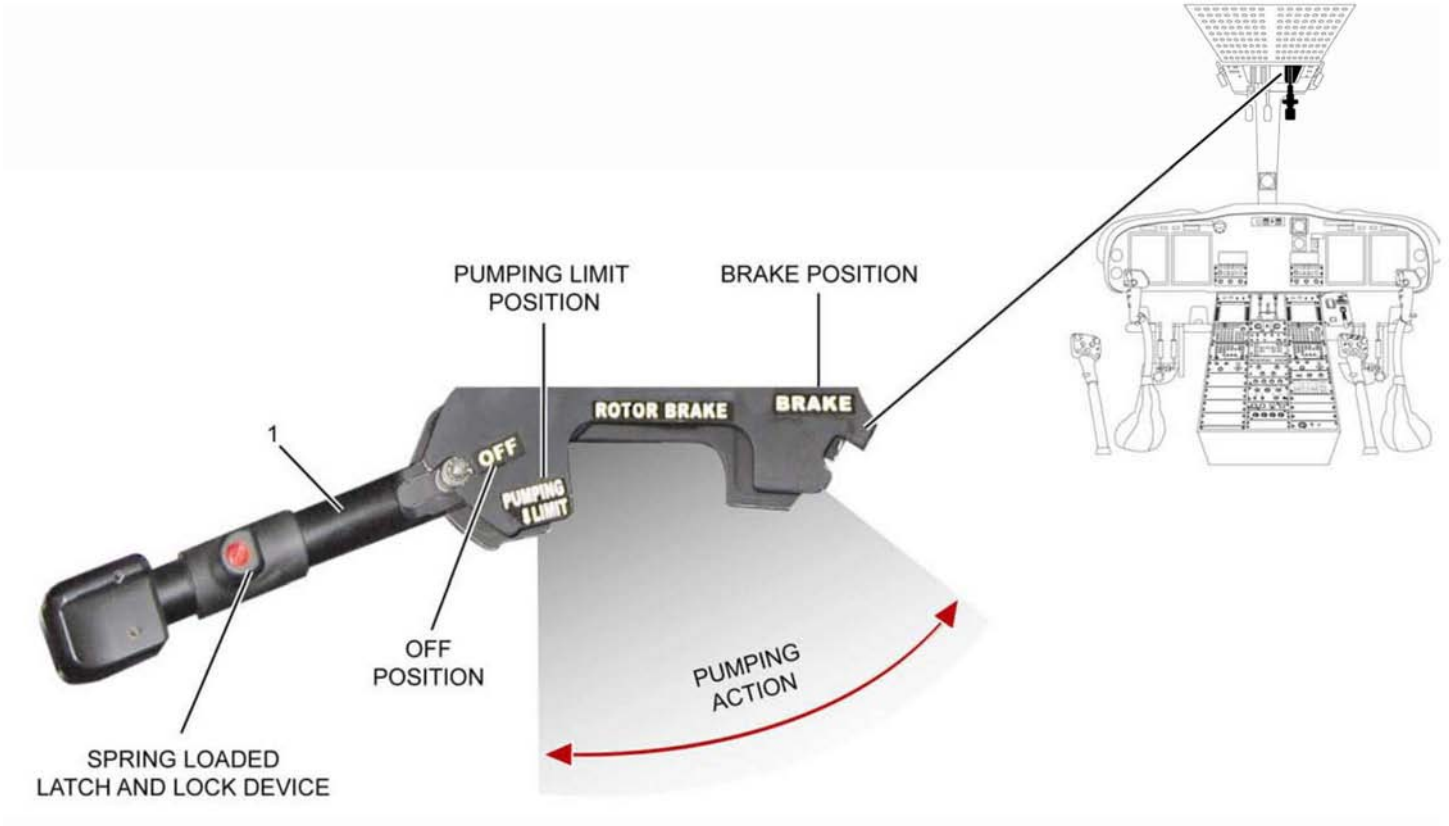
4 CLPR UP green led ..... green led. Lit when the caliper is up; flashes during the transition

5 HIGH PRESS ..... red underlined. Lit when the pressure is equal or above 50 bar

6 LOW PRESS ..... green underlined. Lit when the pressure is equal or less than 20 bar

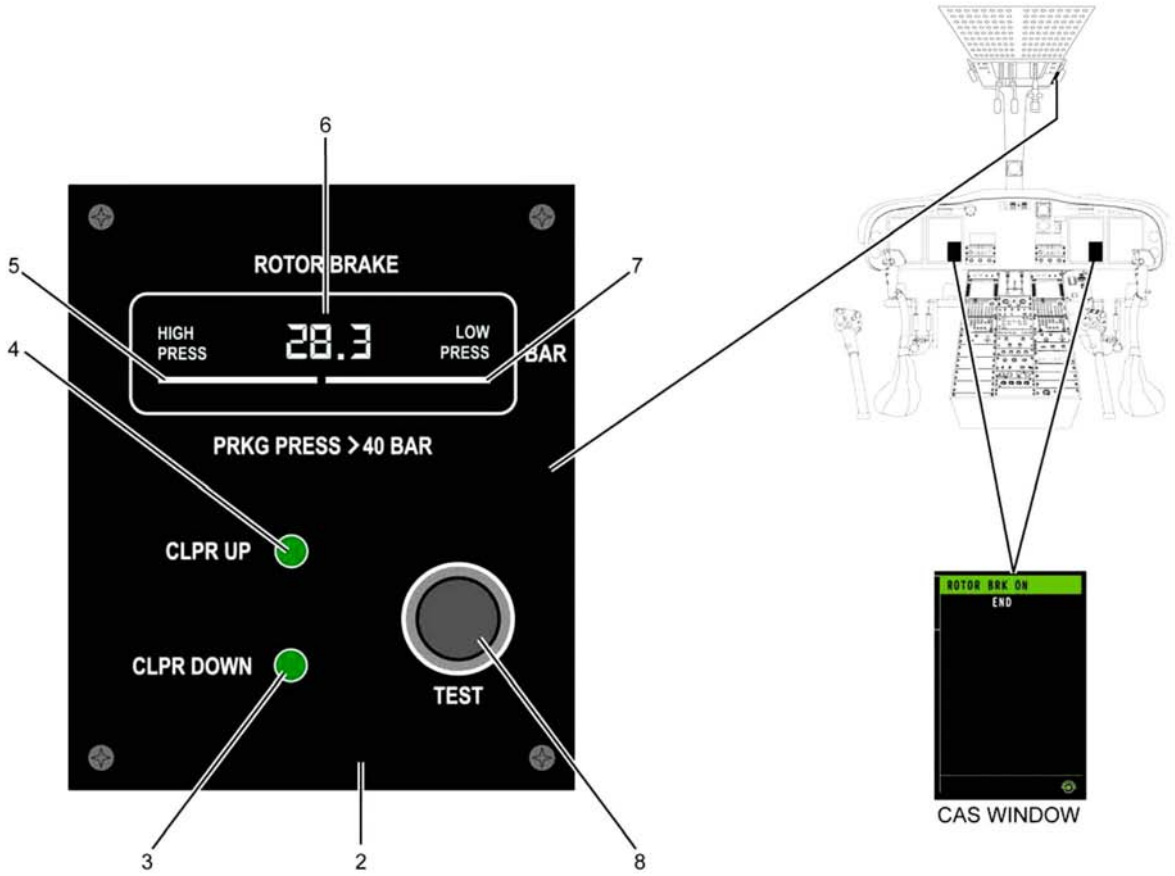
7 PRESS digital readout ..... gredisplays the operating pressure in green

8 TEST pushbutton ..... when pressed the RBPI test light is performed



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ROTOR BRAKE – CONTROLS AND INDICATORS (SHEET 1 OF 2)



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ROTOR BRAKE – CONTROLS AND INDICATORS (SHEET 2 OF 2)

ROTOR BRAKE – OPERATIONS

The braking action is manually controlled by the Rotor Brake Control Lever (RBCL). The movement of the lever allows to generate the required hydraulic pressure, using a mechanical pump that is installed inside the Rotor Brake Control Module (RBCM). The RBCL and the pump are connected via a control rod. The stroke between PUMPING and BRAKE positions, produce the pressure for:

NORMAL BRAKING

- When the pilot moves the RBCL to BRAKE, the RBCM supplies the hydraulic pressure needed to stop the rotor disk (≈ 26-28 bar). The rotors stop in 11 ÷ 15 sec approx. (DYNAMIC BRAKING).

PARKING

- To perform a “pumping action” the RBCM permit to reach a value of 46 bar required to maintain the rotors stopped for approximately 8 hours.

When the lever is set to OFF, the outlet valve (mechanically connected to the lever), goes in open position and permit to de-pressurize the circuit.

For safety reason, the RBA can move in the up and down position. This feature, permit to maintain the Rotor Brake Assembly (RBA) away from the RB disk when all the engines are running and the helicopter is in flight. The RBA is moved by an electrical motor, called Rotor Brake Actuation Assembly (RBAA).

This function is managed by the Rotor Brake Relay Box (RBR) that uses the following signals:

- RBCL / RBAA UP/DOWN positions;
- Engine Control Panel selectors positions;
- WOW (Weight-On-Wheels).

The RBA moves in UP position only when the helicopter is on the ground and the engines mode selector in set to OFF.

The electrical shut-off valve prevents inadvertent pressurization of the RBA (valve open) when:

ON GROUND

- all engines are OFF and WOW ON and hydraulic circuits are pressurized (pressure detected by the Interlock Pressure Switch)

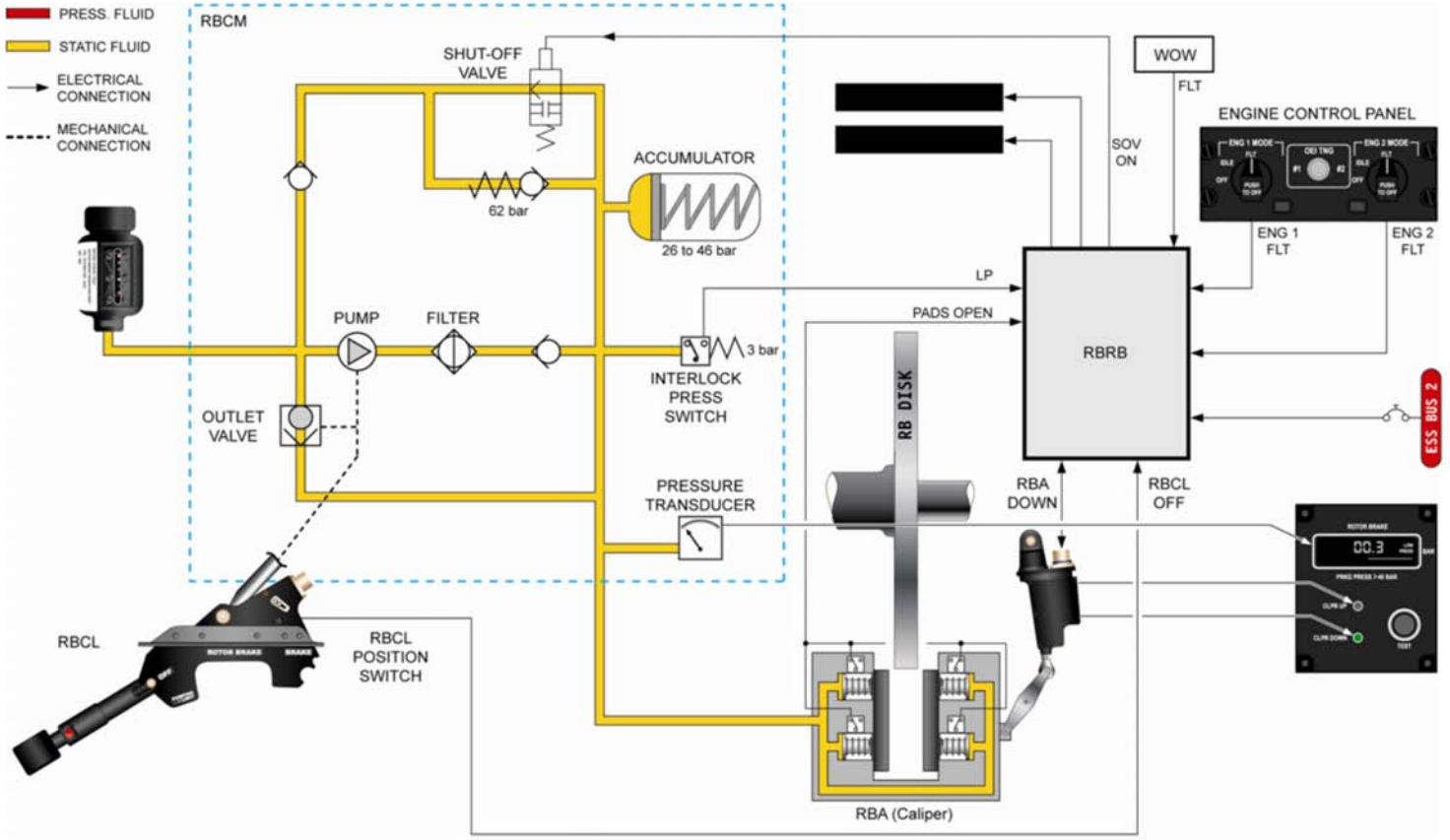
IN FLIGHT

- all engines ON and WOW OFF.

In case the system is already pressurized and the “ENGINE START-UP PROCEDURE” begins, the shut-off valve automatically opens and de-pressurizes the hydraulic circuit.

IN FLIGHT OPERATIONS

- the RBCL is in OFF position
- no hydraulic pressure is furnished to RB circuit
- the caliper is DOWN
- the CLPR DOWN green LED on the Rotor Brake Pressure Indicator (RBPI) is lighted



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IN-FLIGHT OPERATIONS

ON - GROUND OPERATIONS

WITH ONE OR BOTH ENGINES IN FLT OR GI

- the RBCL is in OFF position
- no hydraulic pressure is supplied to rotor brake circuit
- the caliper is DOWN
- the CLPR DOWN green LED on the RBPI is lighted
  - The CLPR DOWN and CLPR UP flash during transition;
  - CLPR DOWN green LED lights on

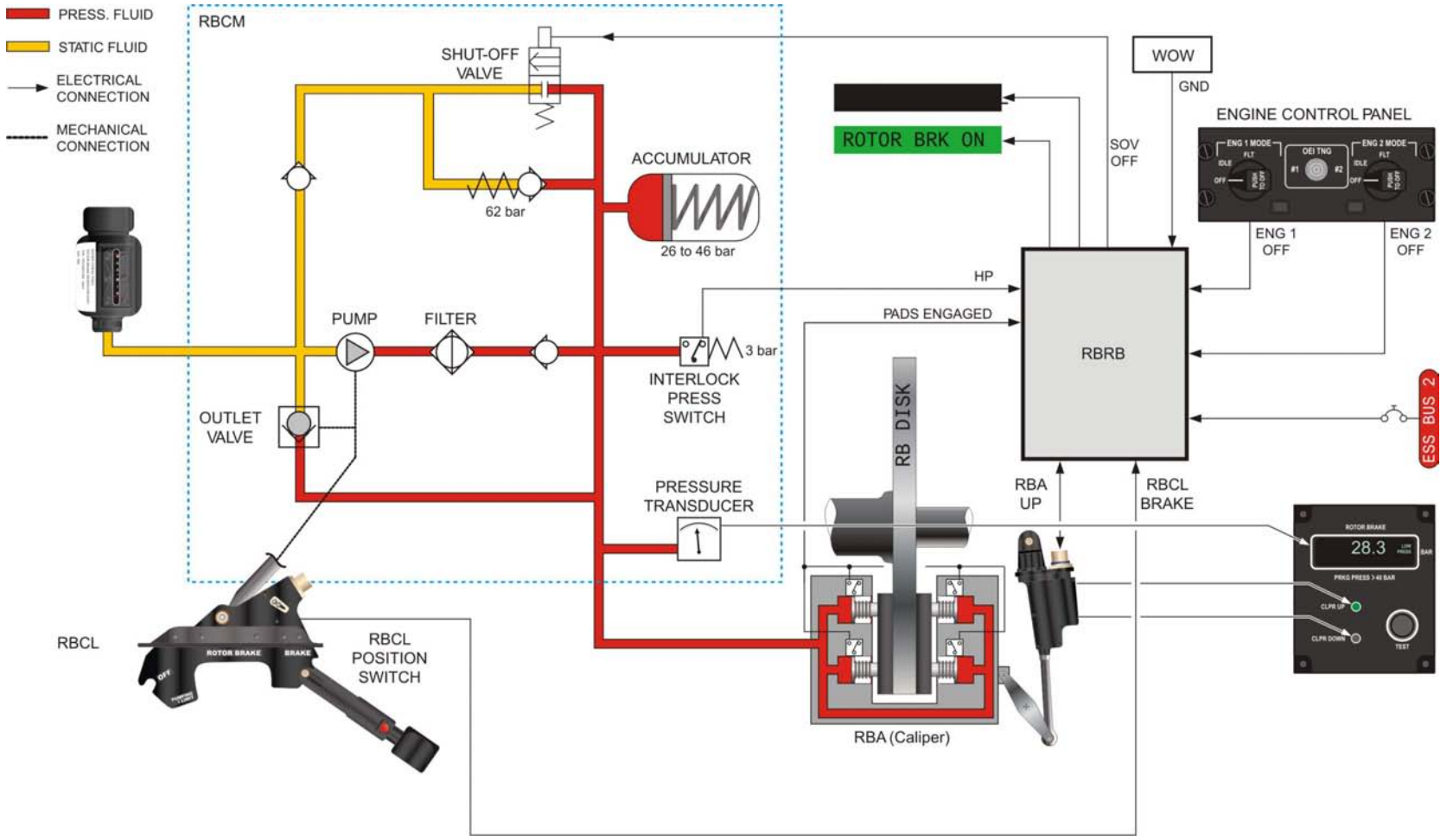
ON GROUND AFTER ENGINES SHUT-DOWN

- the RBCL is in OFF position
- no hydraulic pressure is furnished to RB circuit
- the caliper rises UP automatically
- the RBPI changes the status of the indicators as follows:
  - the CLPR UP and CLPR DOWN flash during transition
  - the CLPR UP lights on
  - LOW PRESS lights on

BRAKE ACTION

When the engines stop the displacement of the RBCL from OFF to BRAKE positions causes the braking action and

- the RB circuit is pressurized at 26-28 bar
- LOW PRESS on the RBPI goes off
- the digital display of the RBPI shows the braking pressure
- the message ROTOR BRAKE ON is displayed on the CAS window



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ON GROUND OPERATIONS

PARKING OPERATIONS

Using pumping action of the RBCL sets the system to PARKING condition.
When the pressure becomes grater than 40 bar but less than 62 bar:

- the RBPI digital display shows the braking pressure;
- ROTOR BRAKE ON is displayed on the CAS window;
- up to 8 hours of parking pressure are guaranteed;

When the pressure decreases below 20 bar:

- LOW PRESS is displayed on the RBPI;
- a pumping action is required to re-pressurize the circuit until HIGH PRESS is displayed on the RBPI.

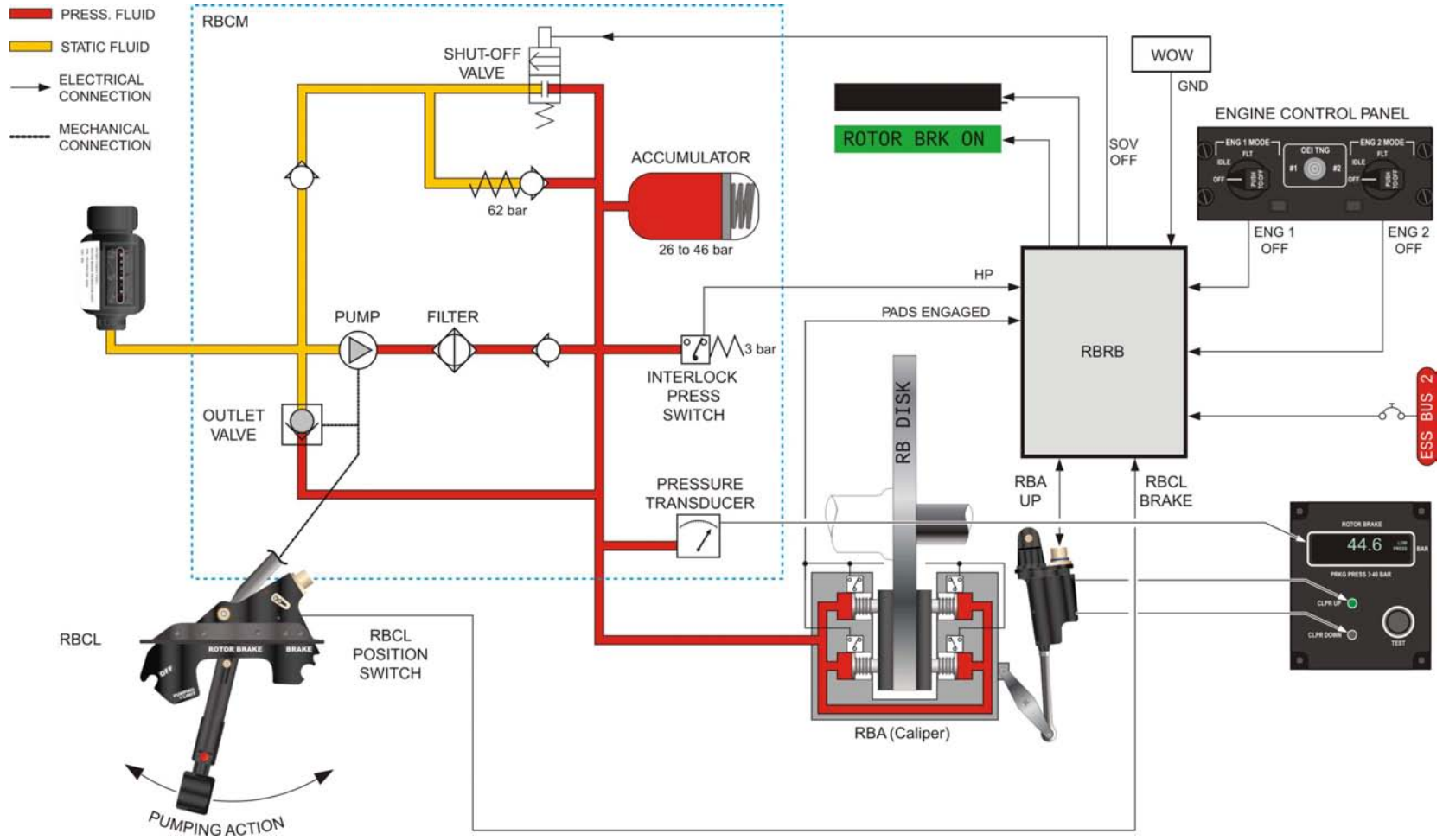
BRAKE RELEASE

To release the brake the pilot must set the RBCL to OFF:

- the CLPR UP green LED on the RBPI remains lighted;
- LOW PRESS is displayed on the RBPI.

AFTER ENGINE STATING:

- the RBCL is in OFF position;
- no hydraulic pressure is supplied to rotor brake circuit;
- the caliper moves automatically DOWN;
- on the RBPI
  - the CLPR DOWN and CLPR UP flash during transition;
  - CLPR DOWN green LED lights on.



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PARKING OPERATIONS

CAS CAUTION MESSAGES	
CAS CAPTION	**FAILURE DESCRIPTION**
ROTOR BRK FAIL	Rotor brake system in one of the failure conditions noted • Brake pads not withdrawn • With both engines OFF brake caliper not UP • With an engine in FLT, brake caliper not DOWN • With an engine in GI, brake caliper not DOWN • Rotor brake lever not in OFF, one engine to FLT or in GI (displayed on ground only)

ROTOR BRAKE ADVISORY MESSAGES ON ROTOR BRAKE MONITOR PANEL	
CLPR UP	RotoRotor brake caliper ready for brake (only on ground)
CLPR DOWN	Rotor brake caliper out of braking position (only on ground)

CAS ADVISORY MESSAGES	
CAS CAPTION	**MESSAGE DESCRIPTION**
ROTOR BRK ON	Rotor brake selected to BRAKE position

ROTOR BRAKE - OPERATIONG LIMITATIONS

Rotor brake can only be actuated with both engines OFF.

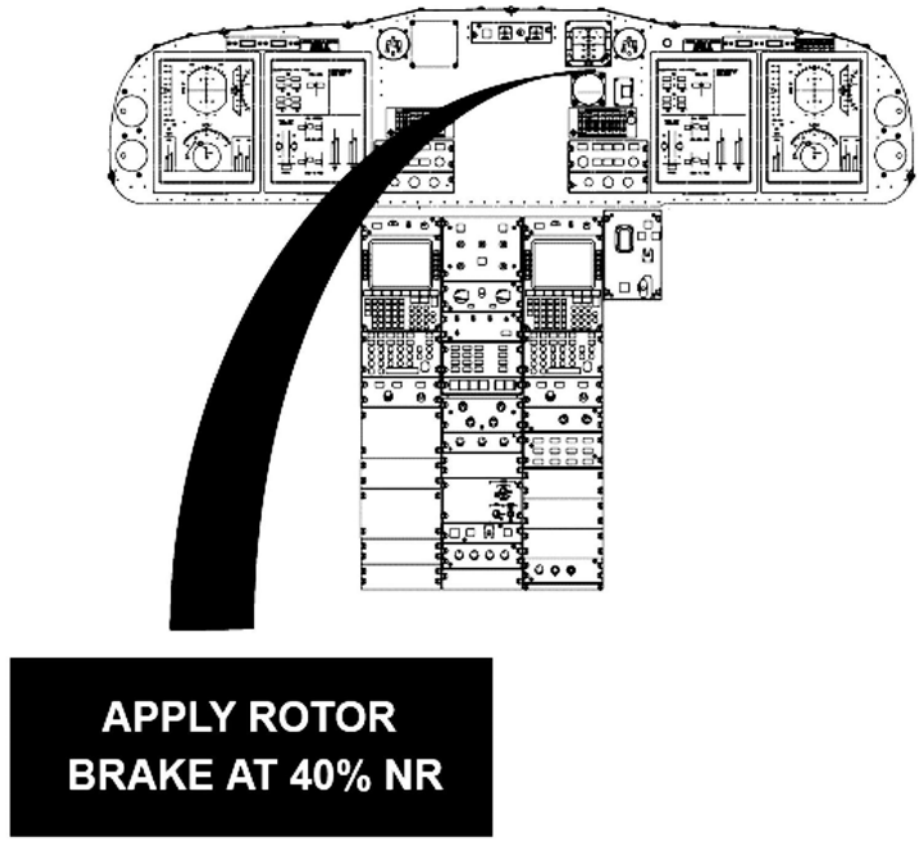
- Max NR (%) for selection to BRAKE position..... 40 %

Braking from 90% NR permitted in an EMERGENCY

- Max pressure when in BRAKE position (BAR)..... 62.5
- Min pressure for level in BRAKE position (BAR)..... 40

NOTE B

When in BRAKE position, the rotor brake is effective for a max of 8 hrs.
Parking on slopes up to 10° is permitted for a max of 1 hr.
The minimum pressure allowed after 8 hrs in BRAKE position is 26 bar,



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CHAPTER

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TAIL ROTOR

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SECTION 00 - TAIL ROTOR

TAIL ROTOR – GENERAL

The Tail Rotor (TR) compensates the reaction torque caused by the main rotor and enables the aircraft to be controlled in its yaw axis.
The TR is a four blade fully articulated rotor with elastomeric spherical bearings, which allow flapping, lead-lag and feathering movements. The TR comprises:

- the tail rotor head
- the blades
- the rotating controls

TAIL ROTOR – MAIN COMPONENTS

TAIL ROTOR HEAD

The tail rotor head is driven by the Tail Gear Box (TGB) shaft and includes

- a splined hub connecting the TGB shaft to the four blades. The hub is constituted by four arms and secured in place by locknut and bolts. An upper and a lower limiter are used to limit the flapping movements between - 10° and + 12°
- four elastomeric bearings make the connection between the blades and the hub and permit lead/lag, flap, pitch change and movement of the blades. In the centre of the elastomeric bearing is inserted a titanium bolt that greatly add to the stiffness of the hub
- four elastomeric dampers are installed between the blades and the hub and dampen the lag movements of the blades. The limiting stops limit the flap movement of the blades to the maximum lateral flap position of the blades
- a cover, made by aluminium alloy, is installed on the top of the rotating controls

BLADES

The tail rotor is a four blade rotor where each rotating blade reacts to centrifugal, bending and torsion loads. The blades are made of composite material (unidirectional fiberglass tapes wrapped in cross ply fiberglass layers) with a thin nickel shield bounded on the outer surface at the leading edge gives protection against erosion.
An upper and lower fiberglass skins are bonded at the rear of the leading edge spar and extended back to joint together and form the trailing edge. The root section is formed in the spar by fiberglass strips that are wound so as to allow the attachment of the elastomeric bearing.
The blade tip includes a pocket where masses are introduced to statically balance the blade; the same occur in the root section.
A lightning conductor jumper is attached to the top of each blade attachment.

ROTATING CONTROLS

The TR interfaces the flight control system via the rotating controls. The rotating controls receive inputs from the TR actuator and transform them to give the required blade pitch-angle changes.
The TR controls include:

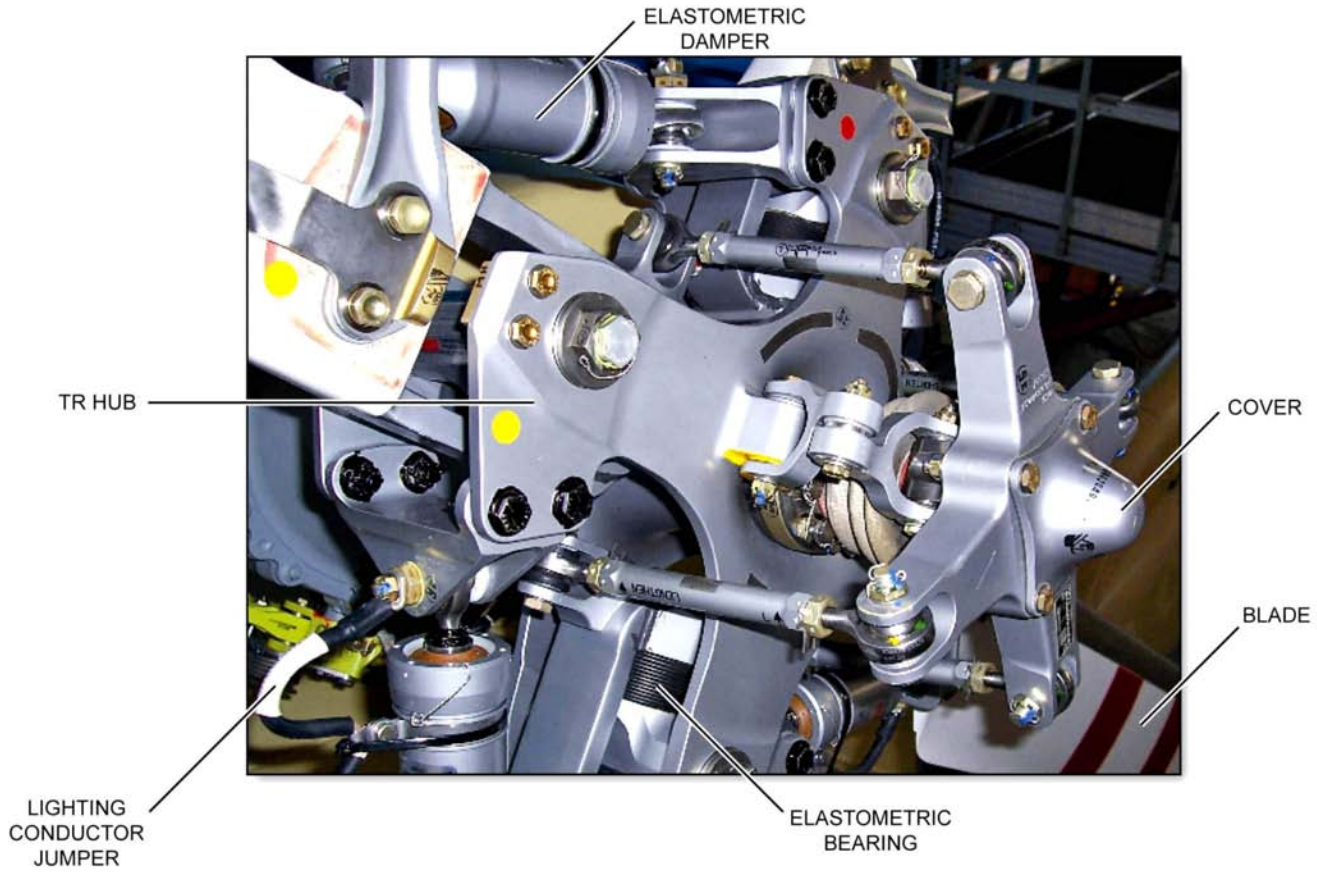
- a spider installed over a sliding tube. The sliding tube is installed on the TGB mast and supplies the attachments for one end of the rotating scissors
- the rotating scissors supply a rotary drive force from the hub to the spider. At the same time the rotating scissors turn to allow lateral pitch-change movement

- four pitch links, one for each blade, connect the TR blades to the spider



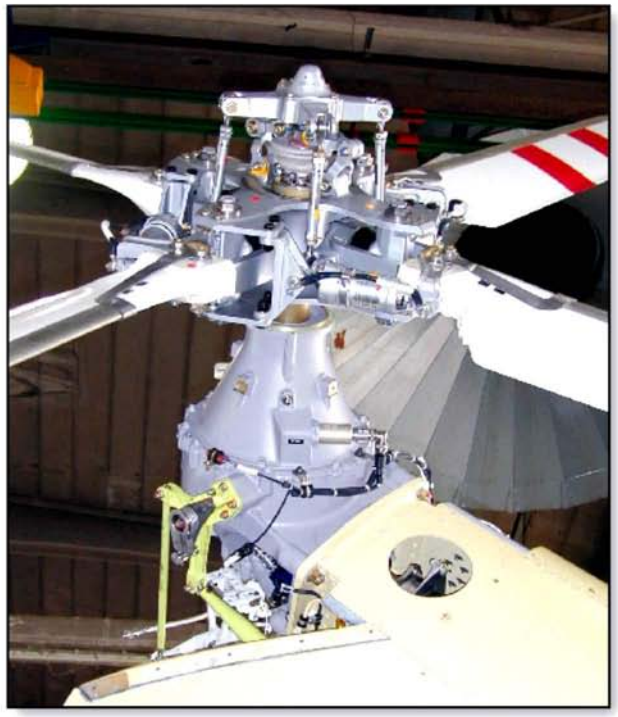
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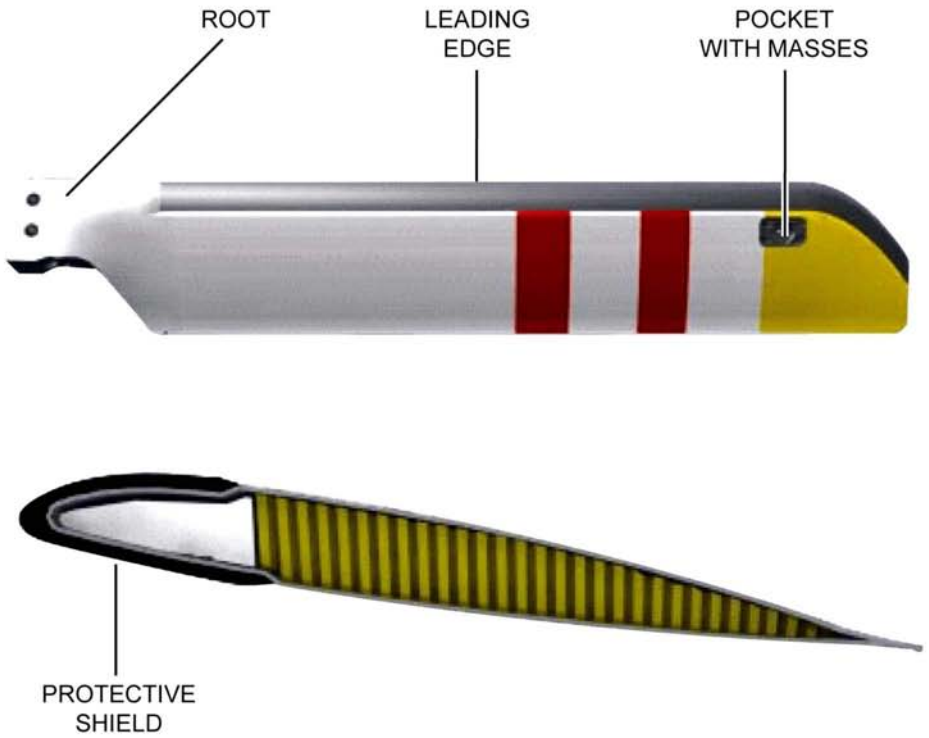
TAIL ROTOR – GENERAL



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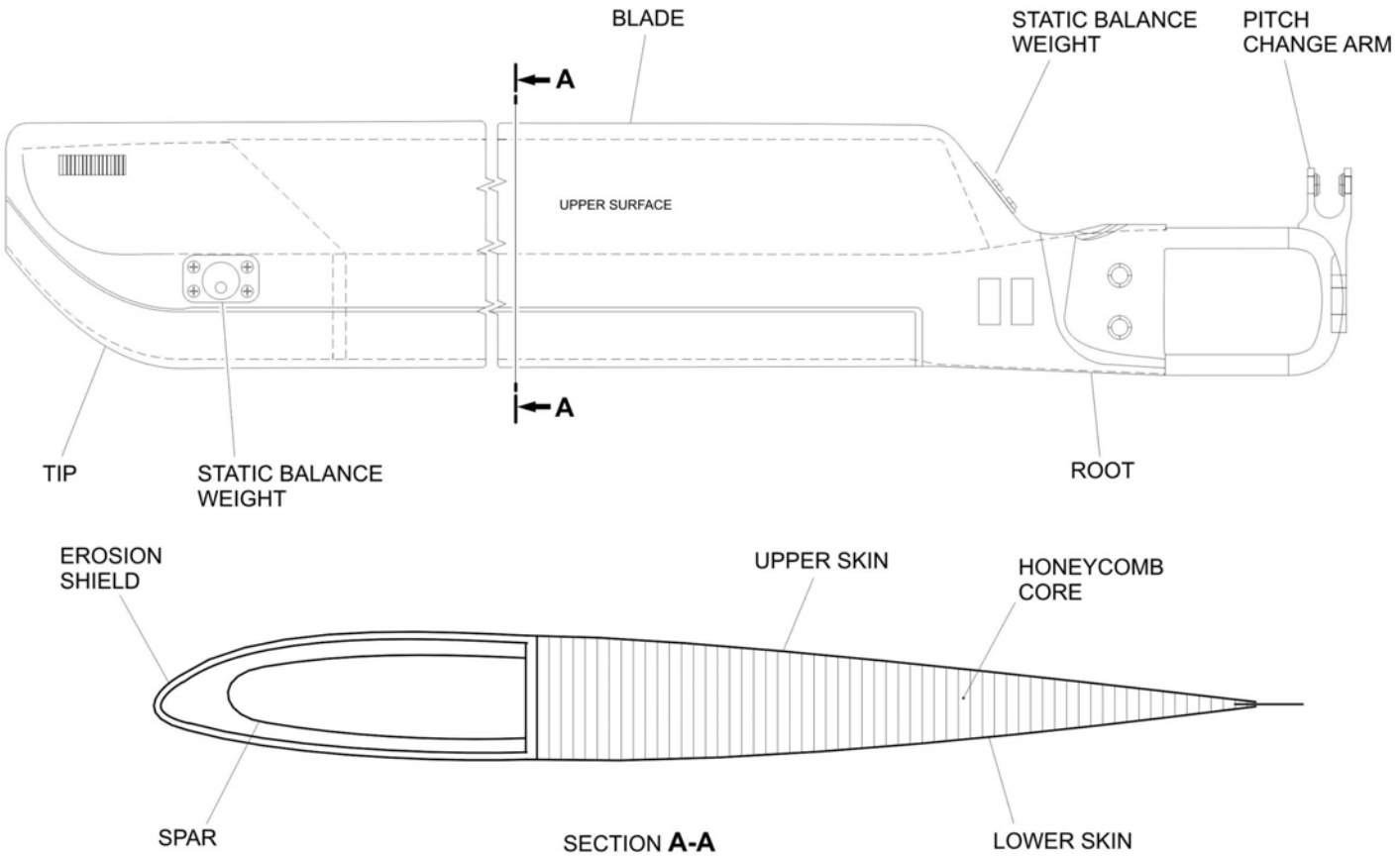
TAIL ROTOR HEAD – MAIN COMPONENTS





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TAIL ROTOR BLADE

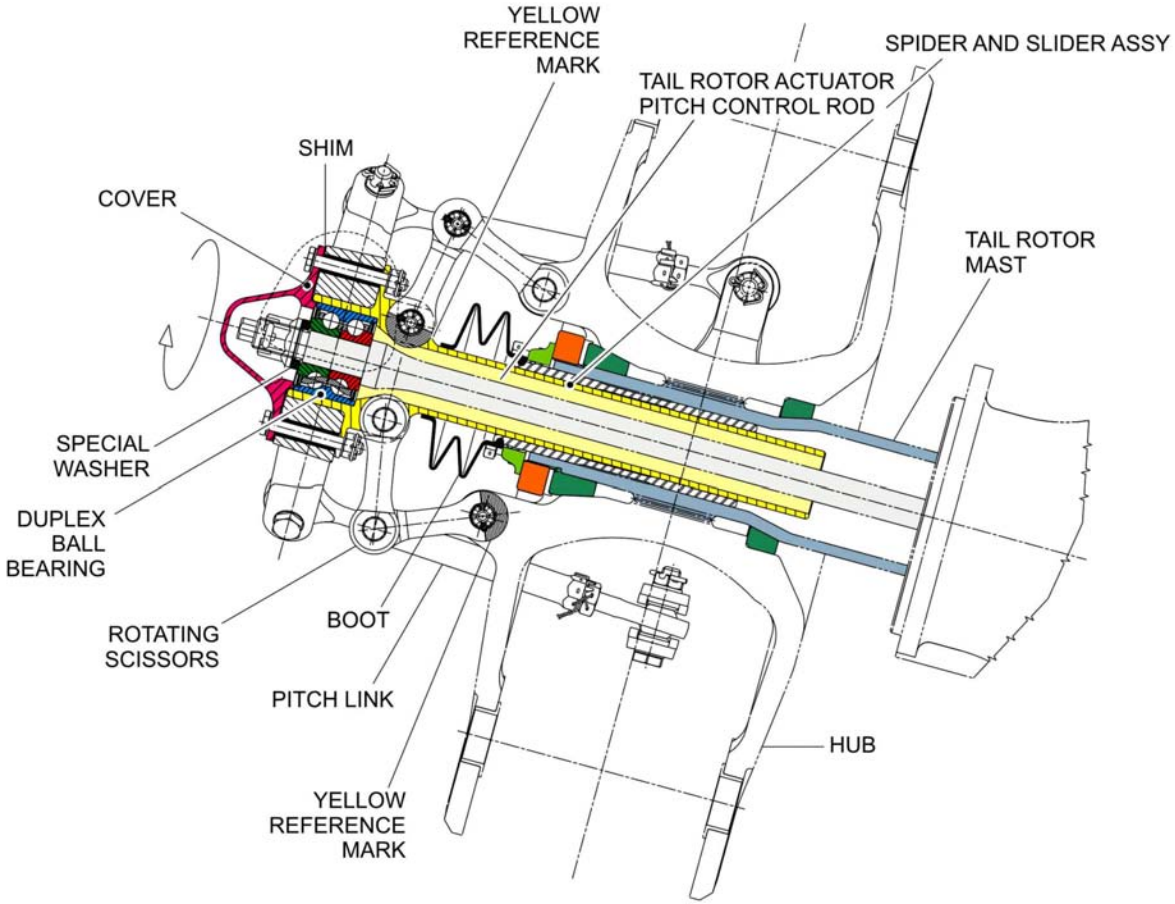


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ROTOR BLADES

ELASTOMERIC BEARING

Elastomeric bearings provide a resilient connection between the blades and hub and react the loads while permitting lead/lag, flap and pitch change movements of the blades.
The elastomeric bearings locate, within the “U” root section of the rotor blades.
The location is achieved by a shaped mounting flange at the smaller end of the elastomeric bearing.
The shaped flange of the bearing has a recessed groove that engages with the closed end of the blade root and pitch change arm.
Four matching holes in the pitch change arm and bearing allows the bolted joining, with the blade secured within the two.
The pitch change arm has a clevis lug extending from the cap section. The clevis allows the attachment of the pitch link.
The larger end of the elastomeric bearing forms a boss that has a hole drilled through it.
The blade, with elastomeric bearing attached, is fitted between the hub arms with the installation bolt passing through hub arms and the bearing to make the attachment.
The installation bolt is hollow to house dynamic balancing weights.
The bolt is externally covered with a solid lubricant film.



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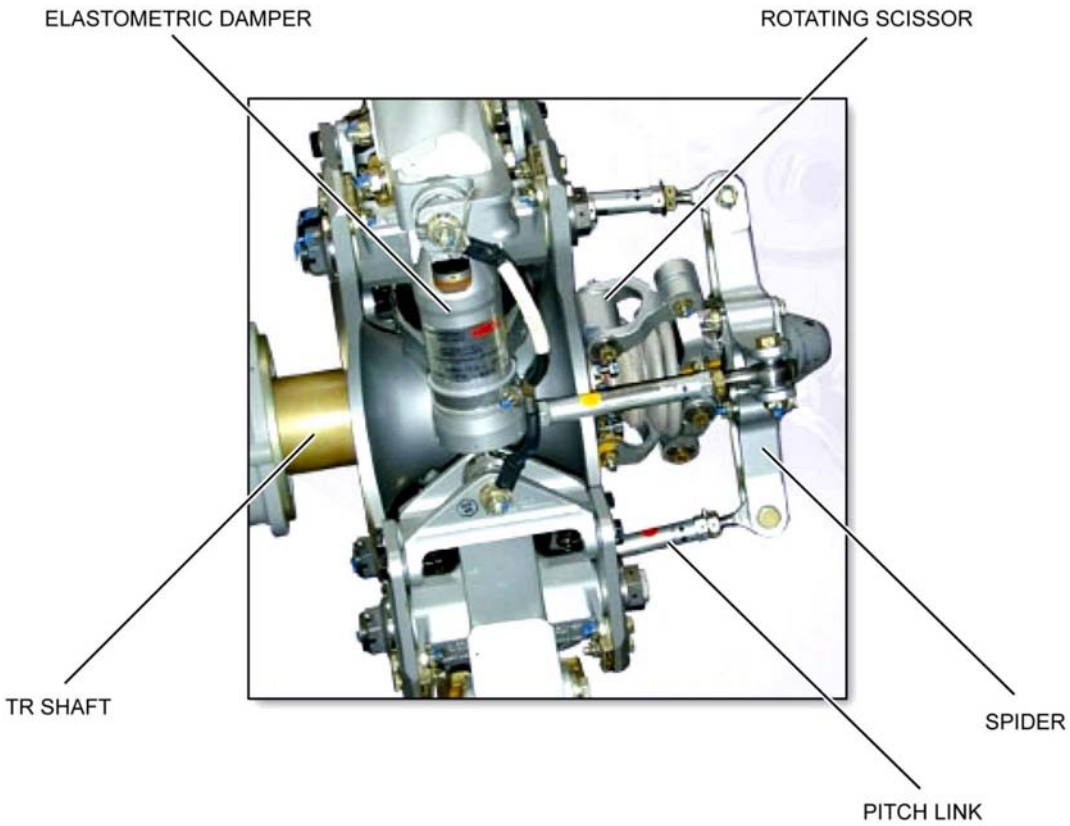
ROTATING CONTROLS

SAFETY PRECAUTIONS

- The materials that follow are dangerous. Make sure that you know all the safety precautions and first aid instructions for these materials:
  - Primer
  - Corrosion inhibitor
  - Solvent
  - Liquid nitrogen
- Be careful when you use the compressed air. Dust and particles can cause injury to your eyes. Always use applicable protective goggles.
- Be careful when you use the liquid nitrogen. Cold parts can cause injury to the persons. Always use applicable protective clothing.

SPECIAL TOOL/TEST EQUIPMENT

- TR pitch link adjustment tool
- TR duplex bearing removal tool
- TR duplex bearing installation tool
- TR slider bushing removal tool
- TR spider and slider assembly removal tool.



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TAIL ROTOR – ROTATING CONTROLS

CHAPTER

65

TAIL ROTOR DRIVE

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SECTION 00 - TAIL ROTOR DRIVE

TRANSMISSION

The transmission supplies the drive from the engines to the Main Rotor (MR), the Tail Rotor (TR) and the accessories (pumps, ECS compressor, and alternators - if installed).
The transmission is composed by:

- the Main Rotor Drive System (MRDS)
- the Tail Rotor Drive System (TRDS)

as visualized in the schematic that follows. This chapter describes the Tail Rotor Drive System.

TAIL ROTOR DRIVE SYSTEM – GENERAL

The Tail Rotor Drive System transmits the rotational drive from the Main Gear Box to the Tail Rotor. The TRDS comprises:

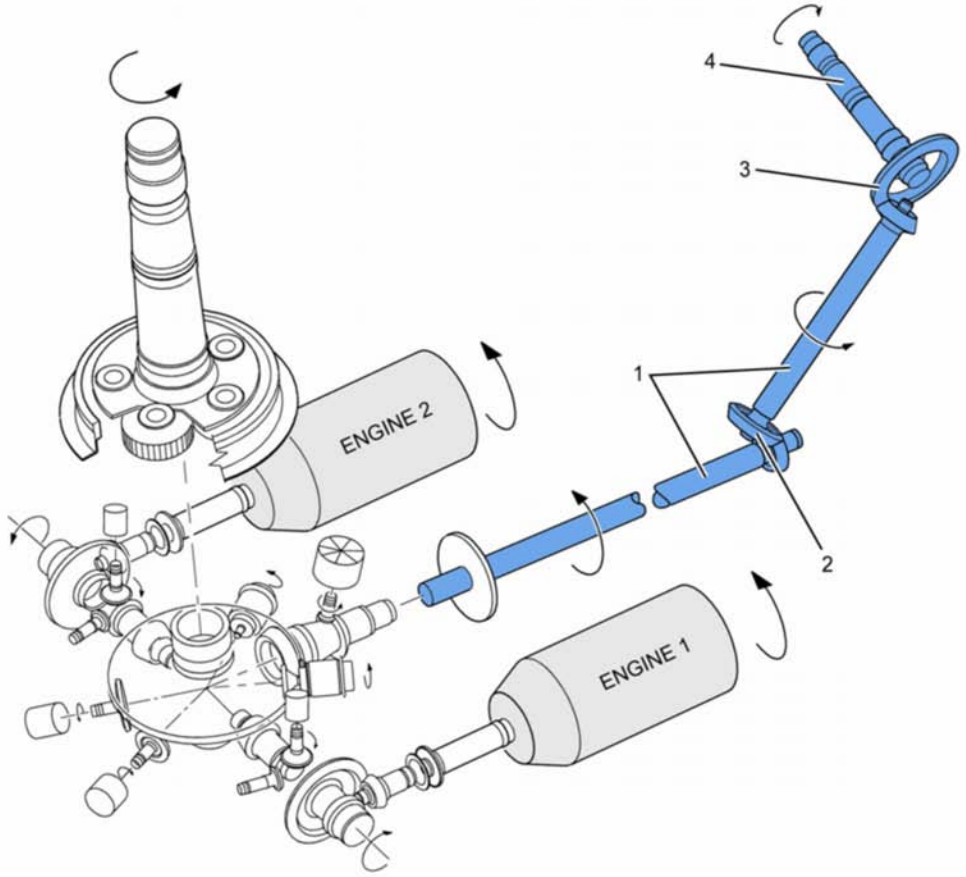
- the tail rotor drive shafts;
- the Intermediate Gear Box (IGB);
- the Tail Gear Box (TGB).

The Intermediate Gearbox (IGB) and the Tail Gearbox (TGB) transmit the drive while changing the drive angle. The gearboxes also reduce the rotational speed between the input and output drive. Each gearbox has a self-contained splash lubrication and condition-monitoring oil system.

■ Tail Rotor Drive System

1. Tail Rotor drive shafts
2. Intermediate Gear Box (IGB)
3. Tail Gear Box (TGB)
4. Tail Rotor Mast

□ Main Rotor Drive System



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TRANSMISSION – GENERAL

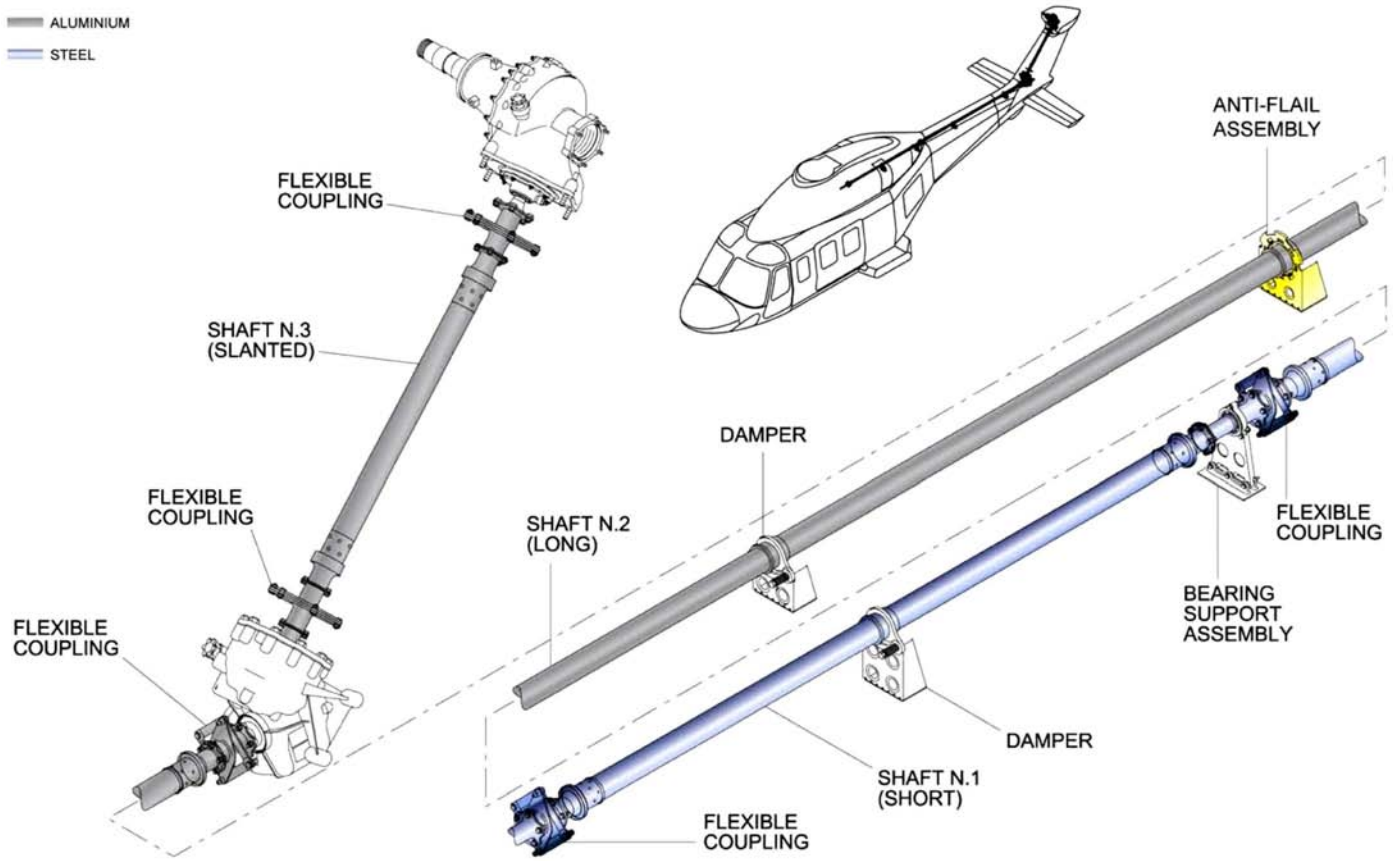
TRDS – MAIN COMPONENTS

TAIL ROTOR DRIVE SHAFTS

The tail rotor drive shafts make the connection and transmit the torque between the gearboxes. Two different length shafts connect the Main Gear Box (MGB) to the IGB. The drive shaft no. 1 starts at the forward end of the drive-train and connects the MGB tail flange coupling to a support bearing assembly. The same support assembly makes a connection to the forward end of shaft no. 2. The rear end of this no.2 shaft connects to the IGB input drive pinion.
One slanted shaft connects the IGB to the TGB.
Shafts no.1 and no.2 have damper-assemblies used to dampen the shaft flexing movements that can happen at critical rotational speeds.
The drive shaft no.2 is a critical component because of a resonance can occur. For this reason an anti-flail assembly is installed in case of failure or disconnection of shaft no. 2.
The damper no.1 is installed on the drive shaft no.1 and decreases the vibrations transmitted to the airframe.

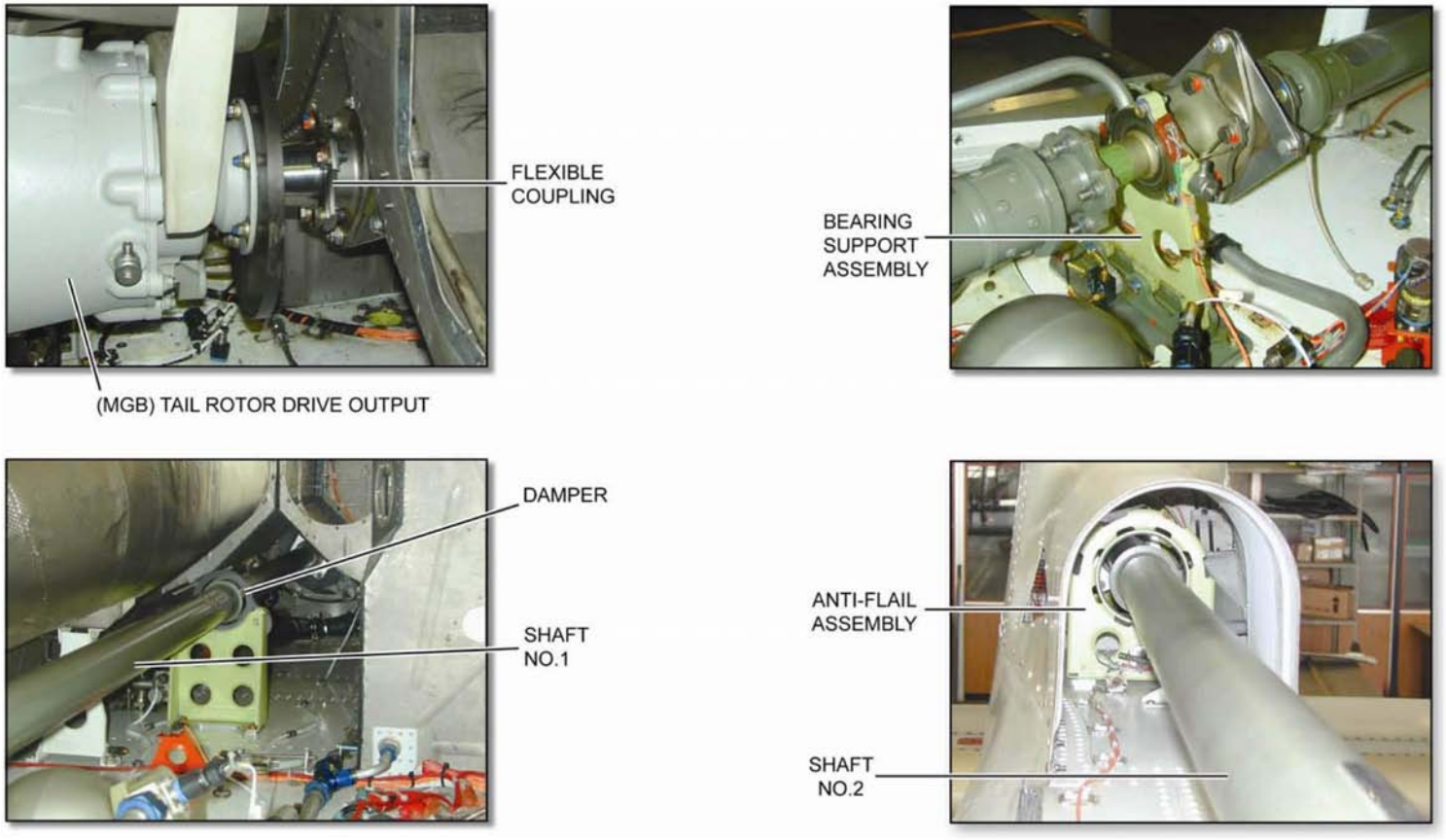
INTERMEDIATE GEAR BOX (IGB)

The Intermediate Gear Box (IGB) is installed on the lower tail-fin structure and changes the direction of the drive to the TGB. The IGB reduces the rotational speed from the input speed of 4532 rpm to the output speed of 3458 rpm.
The IGB is splash lubricated and includes one oil low level sensor, one oil temperature sensor, one accelerometer and one chip detector.
A visual oil level indicator allows to monitor the oil level for maintenance purposes.



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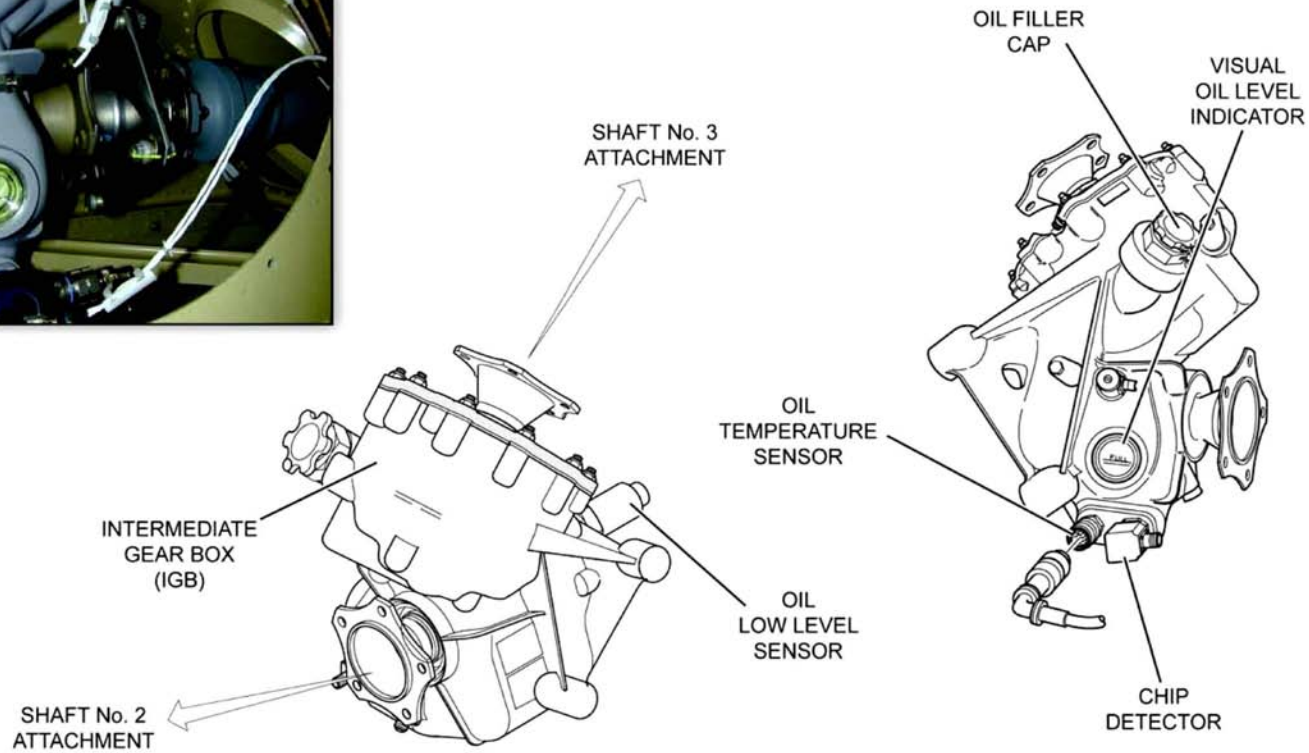
TAIL ROTOR DRIVE SHAFTS



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TAIL ROTOR DRIVE SYSTEM – MAIN COMPONENTS



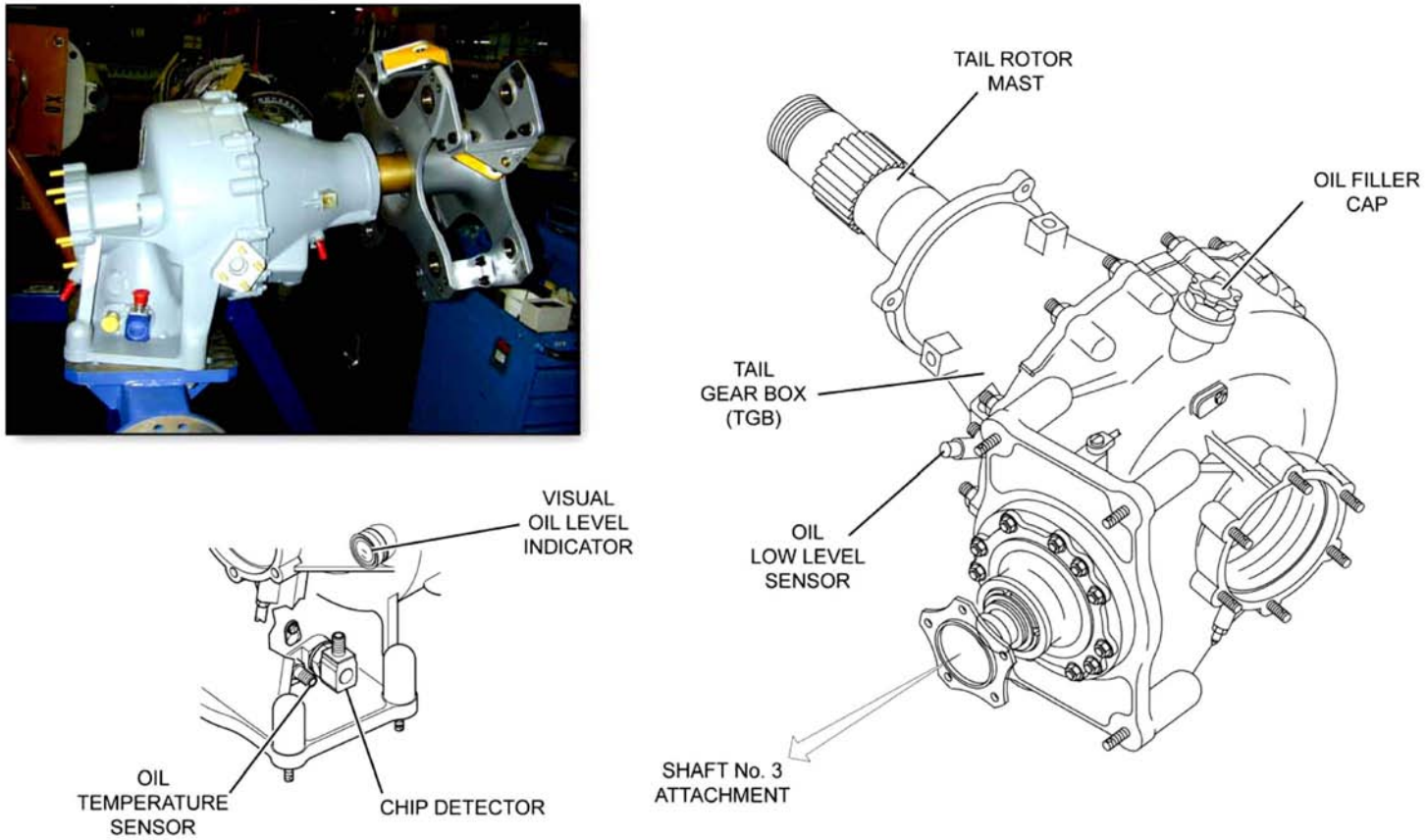


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INTERMEDIATE GEAR BOX (IGB)

TAIL GEAR BOX (TGB)

The Tail Gear Box is installed on the top of the fin. The TGB reduces the input speed of 3458 rpm to the output speed of 1435 rpm which is the nominal speed of the TR.
The TGB is splash lubricated and includes one oil low level sensor, one oil temperature sensor, one accelerometer and one chip detector.
A visual oil level indicator allows to monitor the oil level of the TGB for maintenance purposes.



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TAIL GEAR BOX (TGB)

TAIL ROTOR DRIVE – INDICATIONS

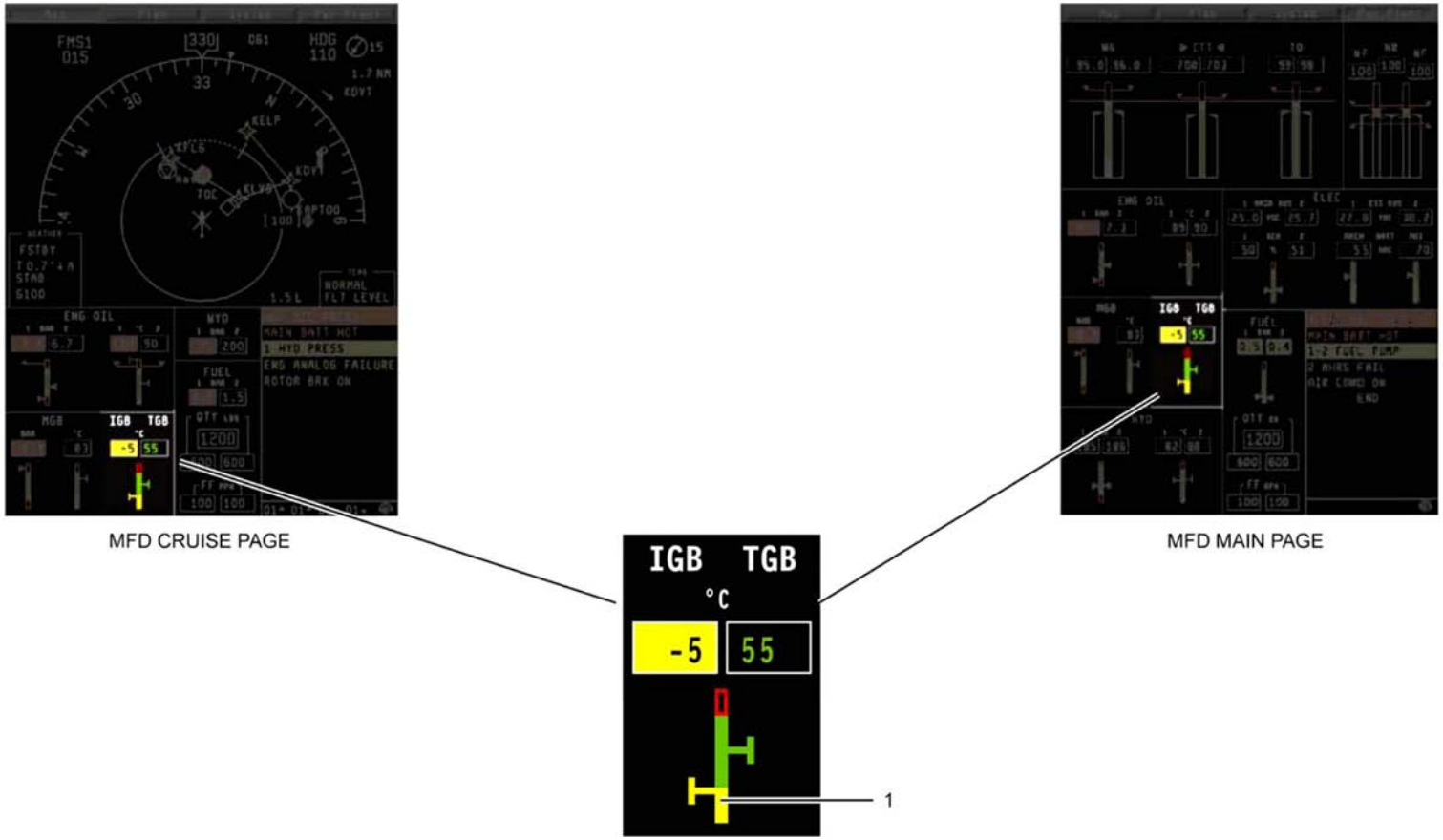
The PWR PLANT PAGE displays the values of temperature in the IGB (Intermediate Gear Box) and TGB (Tail Gear Box) areas.

1 TEMPERATURE

The oil temperature values in the IGB and in the TGB are represented by two digital readouts under the label °C (on the left are displayed the IGB values of temperature; on the right are displayed the TGB values of temperature). Graphically the values are represented on a vertical scale by means of pointers (T symbols) that match the color of the area on the scale.

Green band of the analogue vertical scale represents a normal condition for the oil temperature and so the associated digital readout values in Celsius degrees.

The amber band represents a caution condition while the red band is associated to a warning condition.



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MAIN ROTOR DRIVE – INDICATIONS

TAIL ROTOR DRIVE – CONTROLS AND INDICATORS

1 CHIP BURNER push-button

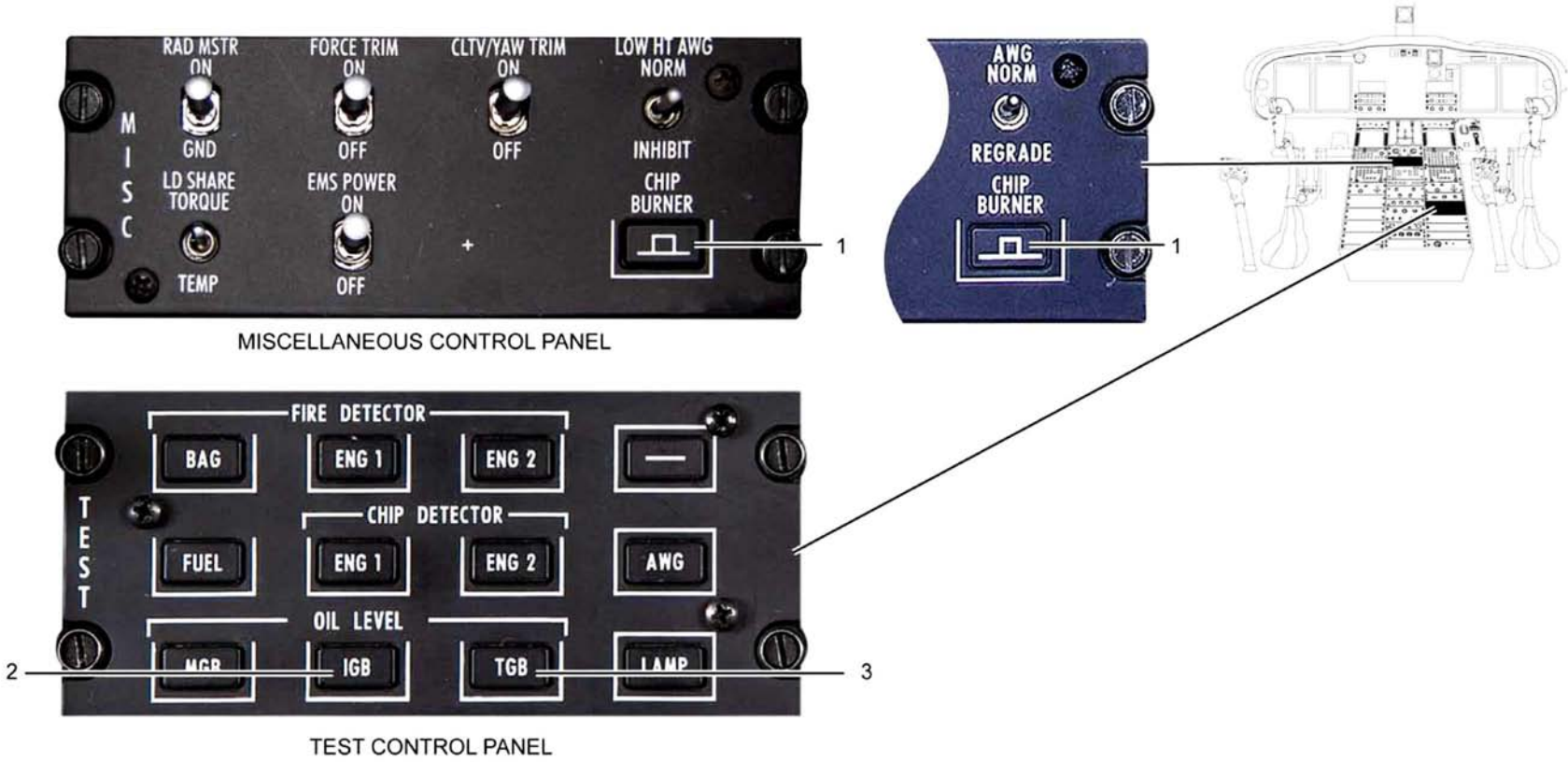
- PRESSED ..... a chip burning attempt for the IGB CHIP and TGB CHIP is performed

2 OIL LEVEL IGB push-button

- PRESSED ..... the caution IGB OIL LOW in inverse video is displayed in the CAS window
  The test is possible only when the helicopter is on the ground and the NR is less than 2%

3 OIL LEVEL TGB push-button

- PRESSED ..... the caution TGB OIL LOW in inverse video is displayed in the CAS window
  The test is possible only when the helicopter is on the ground and the NR is less than 2%

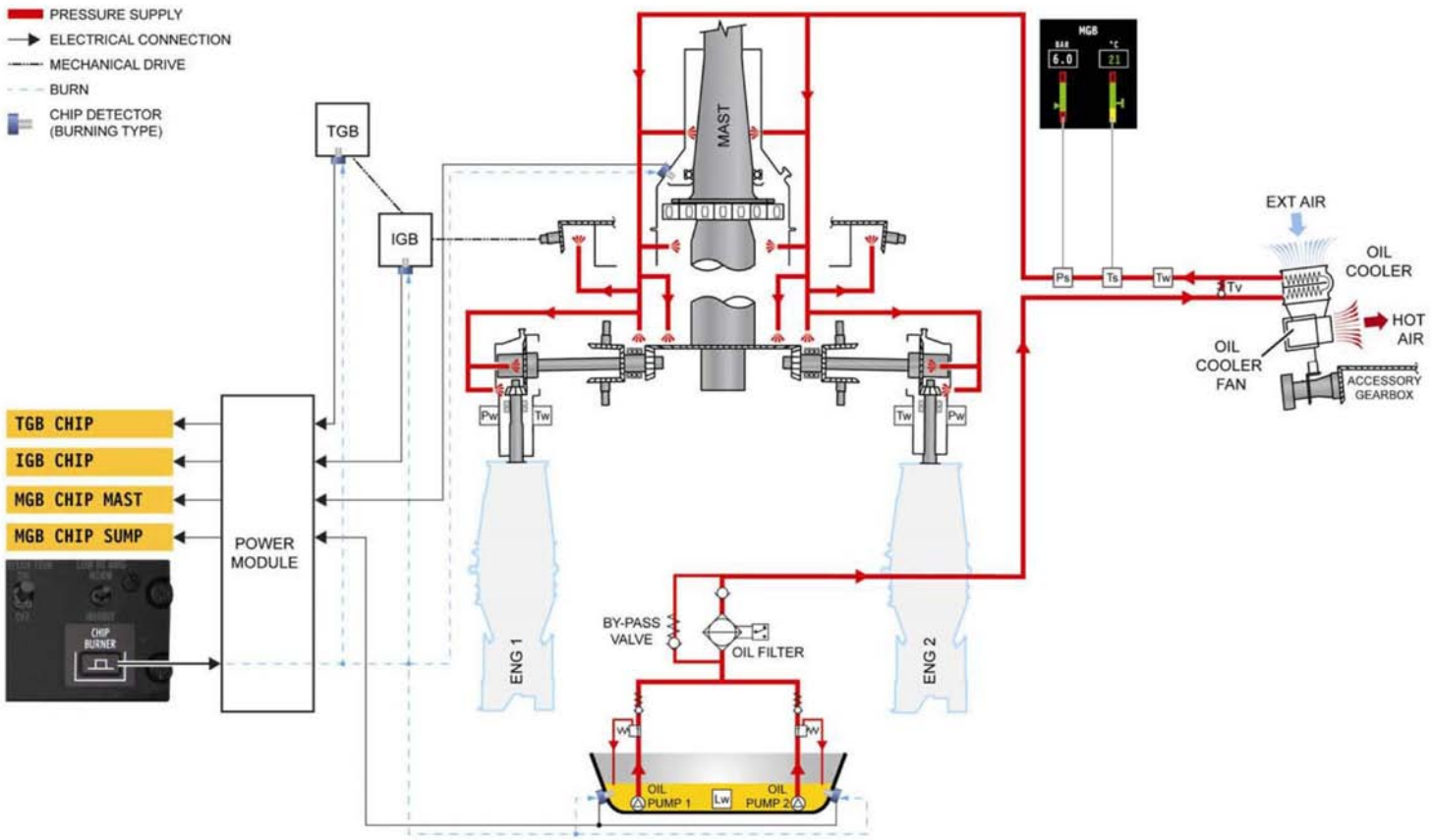


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TAIL ROTOR DRIVE – CONTROLS AND INDICATORS

CHIP DETECTOR SYSTEM

The chip detector system is described in the Ch.63-00-00. On the TRDS there are two chip detectors: one chip detector is installed on the IGB and one chip detector is installed on the TGB.
When the IGB CHIP or the TGB CHIP caution messages are displayed in the CAS window of the MFD, the pilot can try to burn the ferrous particle(s) pushing the CHIP BURNER switch located on the MISC control panel.
If the burning is successful the particle is classified as “small” and the relevant caution extinguishes.
In the opposite case, the particle is considered “large” and cannot be burned.
When the particle is burned, a message is written in the Non Volatile Memory (NVM) for maintenance purposes.



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CHIP DETECTOR SYSTEM

CAS CAUTION MESSAGES		
CAS CAPTION	**FAILURE DESCRIPTION**	**PROCEDURE NAME**
IGB CHIP	Activated CHIP burner (on MISC control panel) It is permitted to activate the CHIP BURNER up to 3 times to clear a chip in one flight. On the 4th CHIP caution Land As Soon As Practicable	INTERMEDIATE GEARBOX CHIP
TGB CHIP		TAIL GEARBOX CHIP
IGB OIL LOW	IGB oil temperature above limit (greater than109°C)	INTERMEDIATE GEARBOX OIL LOW
TGB OIL LOW	IGB oil temperature above limit (greater than109°C)	TAIL GEARBOX OIL LOW
IGB OIL TEMP	IGB oil level low (caution only active with engines shut down and NR below 2%)	INTERMEDIATE GEARBOX OIL TEMPERATURE HIGH
TGB OIL TEMP	TGB oil level low (caution only active with engines shut down and NR below 2%)	TAIL GEARBOX OIL TEMPERATURE HIGH
CHIP DET UNIT CHIP DET TEST	Chip detect system malfunction	GEARBOX CHIP DETECT UNIT MALFUNCTION

CAS CAUTION MESSAGES		
CAS CAPTION	**FAILURE DESCRIPTION**	**PROCEDURE NAME**
IGB CHIP FAIL TGB CHIP FAIL	Associated gearbox chip sensor failed	GEARBOX CHIP DETECTOR SENSOR FAILURE

CHAPTER
67
ROTORS FLIGHT CONTROL

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SECTION 00 - ROTORS FLIGHT CONTROL

ROTOR FLIGHT CONTROLS – GENERAL

The rotor flight controls allow to manage the flight attitude, altitude and direction of the aircraft. The control is transmitted by means of mechanical linkages that interface with manual input controls (collective, cyclic and pedals both pilot and copilot) and Automatic Flight Control System (AFCS) input controls.
The rotor flight controls include:

- the main rotor controls system
- the tail rotor controls system
- the rotor flight controls indicating system



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ROTOR FLIGHT CONTROLS - GENERAL

MAIN ROTOR CONTROLS SYSTEM – GENERAL

The main rotor controls system includes

- the collective control
- the cyclic control

The two controls are mixed, before to be sent to the main rotor servo actuators, to obtain the desired attitude and altitude.

COLLECTIVE CONTROL – GENERAL

The collective control is a conventional metallic rod and bellcrank type. The control is composed of a lever for pilot and copilot that make a common input to a mixing unit. The collective control sticks are made of aluminium with a control grip at the top. The levers are connected together by a torsion tube.

An adjustable friction is installed at the right end of the Collective torsion tube: it used to increase the collective control mechanical resistance; such resistance can be as high as to hold the stick during manual flight.

Two Linear Variable Differential Transformer (LVDT) are installed at the left end of the Collective torsion tube: they provide position of the Collective lever to the engines for power change anticipation.

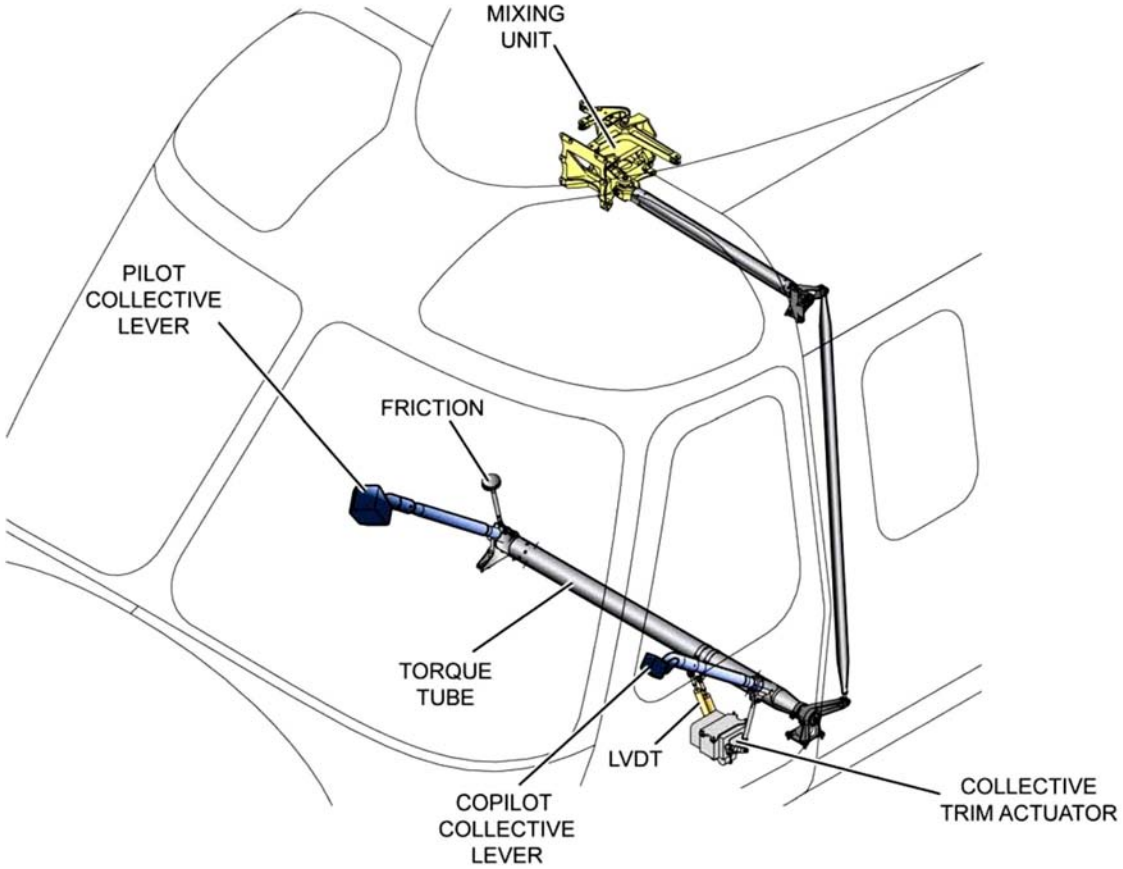
From the control tube a rod extends vertically to the roof and them, through another torsion tube inputs, to the mixing unit.

COLLECTIVE TRIM ACTUATOR

A collective trim actuator is installed in parallel with the torque tube. It has the same control travel authority as the pilot but with a low-limited rate of control for safety in case of a malfunction occurs. The trim actuator can be overridden by pilot commands via a spring-clutch.

COLLECTIVE CONTROL – PRINCIPLE OF OPERATION

The pilot and copilot collective levers are connected together to give a common input to the mixing unit.
The mixing unit, in turn, gives a common output towards the three hydraulic main rotor servo actuators which change the pitch angle of the main rotor blades to obtain the desiderated change of attitude and altitude.
An electrical trim actuator connected to pilot and copilot collective provides an automatic input towards the AFCS.



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COLLECTIVE CONTROL

CYCLIC CONTROL – GENERAL

The cyclic control is a conventional metallic rod and bellcrank type. The control is composed of a stick for pilot and copilot that make a common input to a mixing unit. The collective control sticks are made of aluminium with a control grip at the top. The pilot and copilot stick are connected together by a torsion tube. The sticks are connected to the floor using an attachment plate containing a spherical bearing which is connected with a cross shaped, stainless steel fitting to a torsion tube.

The pilot attachment plate contains a shaped flange used to limit pitch and roll movements combination.

To statically balance the control linkage weight, springs are installed on the stick cross-shaped fitting.

At the left end of the torsion tube, the cross-shaped fitting allows for the connection with rods to the longitudinal and lateral trim actuators.

From the right side the control is connected with a lower bellcrank installation and then with rods that extend vertically to an upper bellcrank installation.

Each lower bellcrank installation has a friction clamp (fixed friction) for counteracting the output force of the series actuators, when the pitch/roll trim actuators are declutched, thus ensuring the movement is translated to the mixing unit.

At the upper bellcrank installation there is a compound linkage connection which transmits the longitudinal and lateral control to the mixing unit. Interposed between the compound linkage and the mixing unit are the AFCS series actuators. Acting on the rod that connects between each series actuators and the compound linkage is an anchor spring which provides a safety function for the event of an actuator disconnect, in that the spring will react the manual input as a pivot point to ensure the movement is translated to the mixing unit.

The compound linkage then mixes the pilot input with the series actuators (pitch and roll dual linear actuators) input and transmits this output to the mixing unit. Limit stops are present in the upper bellcrank to limit the longitudinal and lateral cyclic pitch to the limits of its envelope.

PITCH AND ROLL TRIM ACTUATORS

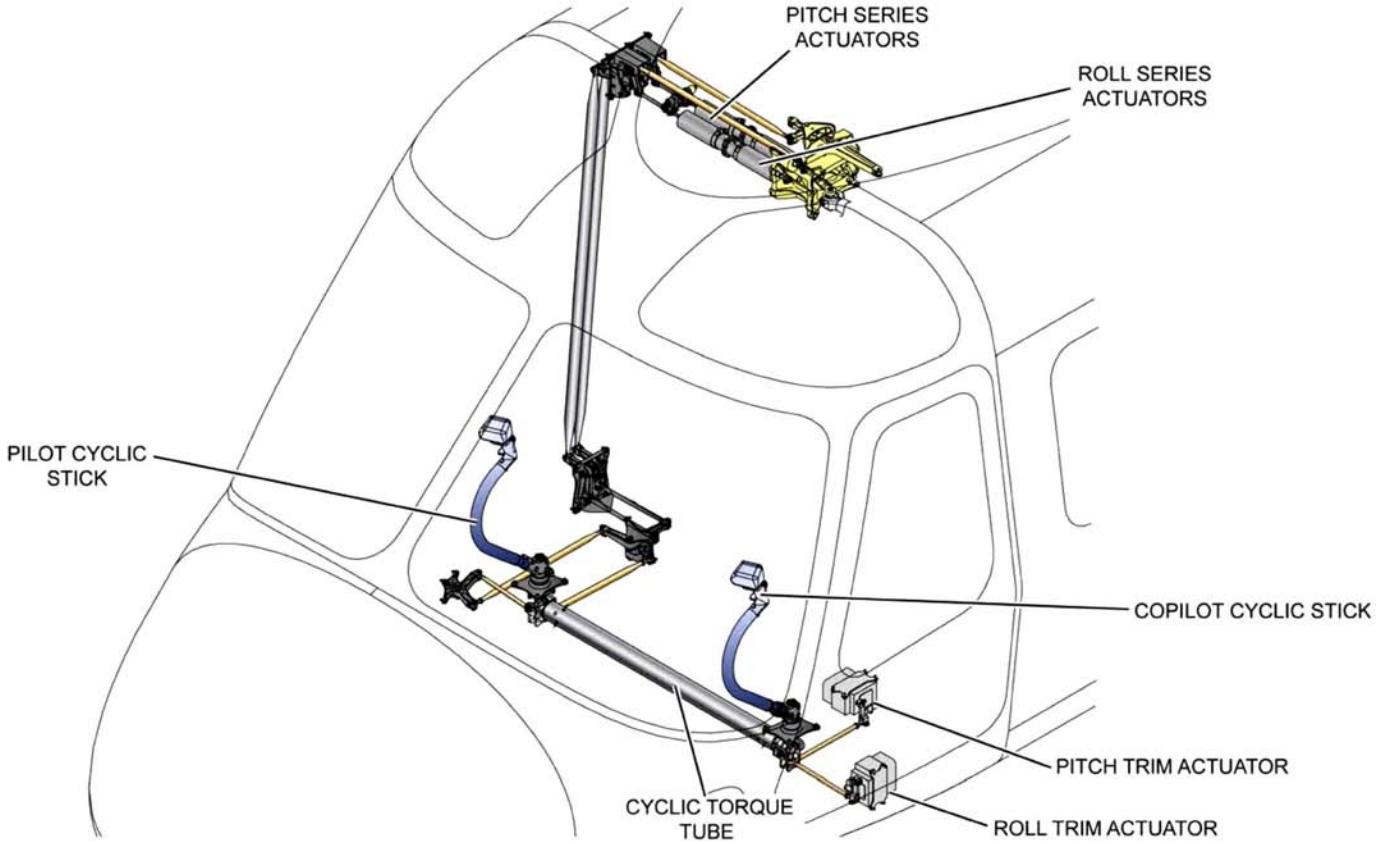
Pitch and roll trim actuators are installed in parallel with the cyclic stick. Trim actuators have the same control travel authority as the pilot but with a low-limit rate of control for safety in case of malfunctions.
Trim actuators can be overridden by pilot commands via a spring-clutch. Actuators are operated by pilot commands or by the AFCS. They convert the command to a mechanical output.

CYCLIC CONTROL – PRINCIPLE OF OPERATION

The cyclic control is a mechanical, dual linkage system, actuated by the cyclic control sticks.

The two cyclic inputs are transmitted to the mixing unit which sends the output to the three main rotor actuators.

The mechanical linkage also gives a connection to the pitch and roll trim actuators which can also give pitch and roll commands.

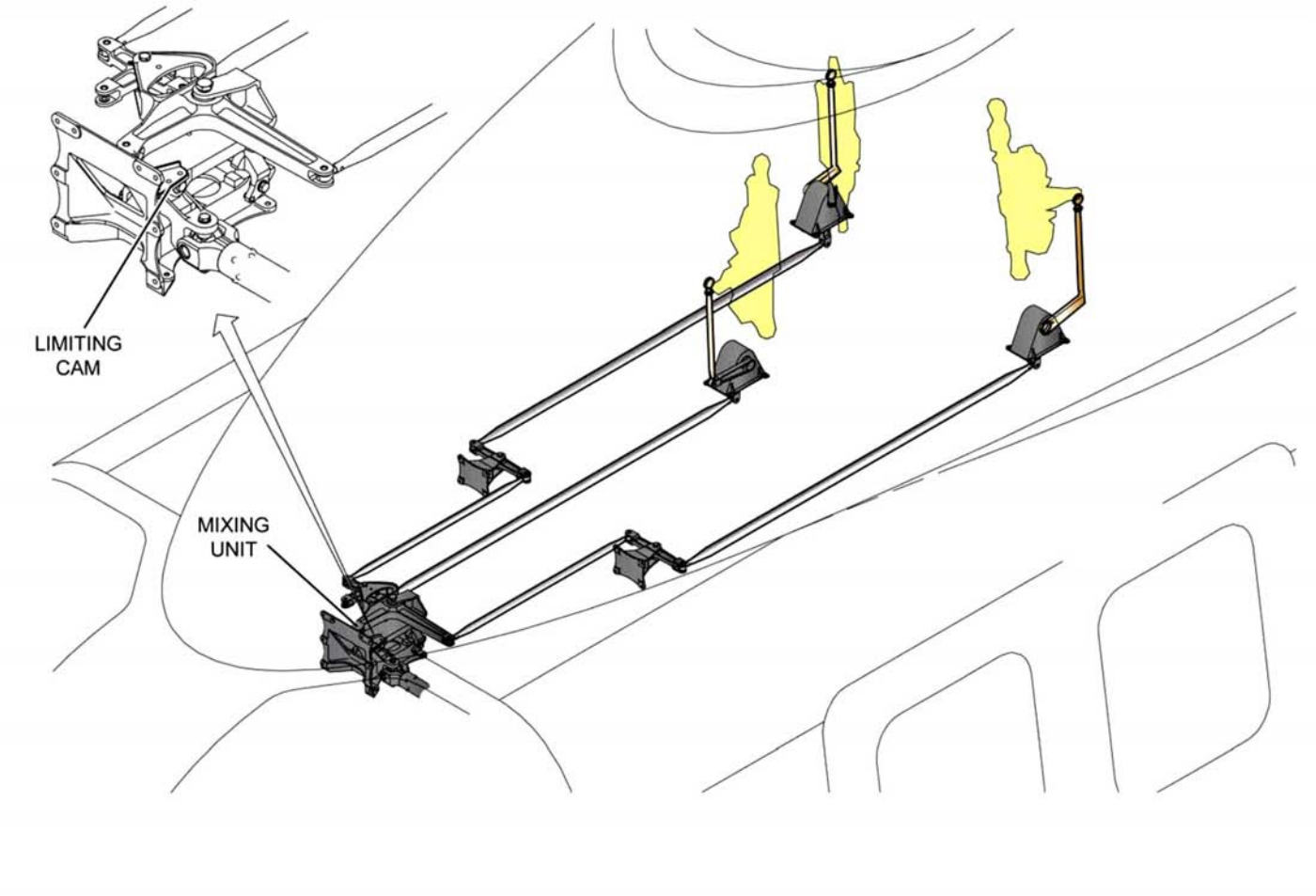


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CYCLIC CONTROL

MIXING UNIT

The mixing unit is a mechanical equipment made by a linkage on a movable support assembly.
The mixing unit receives a mechanical input from the collective controls and cyclic controls and, after mixing them, sends a mechanical output to main rotor servo actuators.
The mixing unit includes a mechanical stop that restricts the longitudinal pitch control as a function of collective displacement; full forward longitudinal cyclic control is achieved only if collective control is not at minimum.
The output from the mixing unit to the servo actuators comprises three rods that use a pivot lever and bell cranks to transmit the mechanical movement.



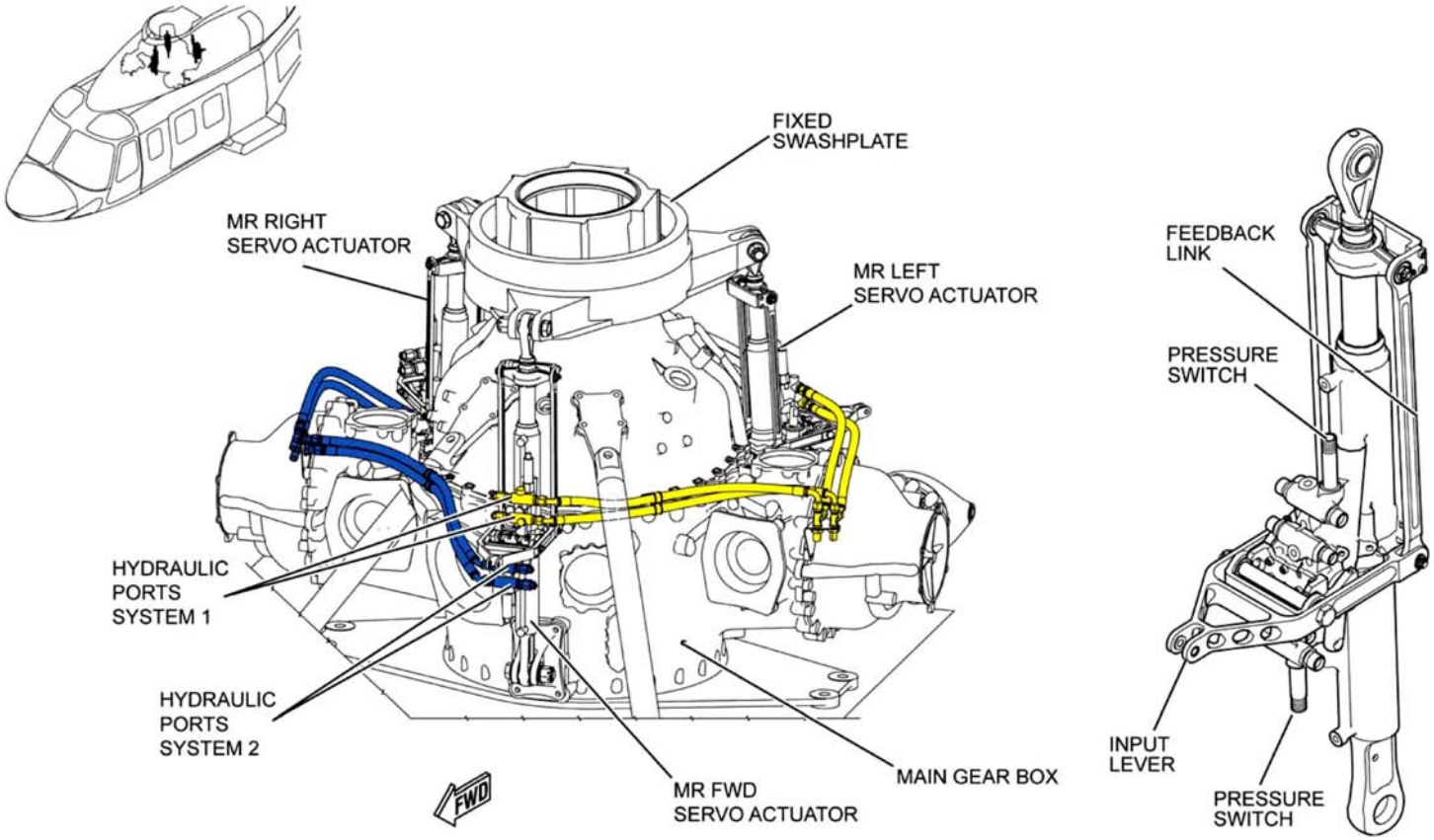
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MIXING UNIT

SERVO CONTROLS

MAIN ROTOR SERVO ACTUATORS - GENERAL

There are three Main Rotor (MR) servo actuators: LEFT, RIGHT and FORWARD (LH, RH and FWD).
Each main rotor servo actuator is a fixed body actuator comprising two separate cylinder assemblies bolted together at the actuator center and tandem pistons.
The upper piston is connected to the fixed swashplate; the lower end of the cylinder is connected with a support to the MGB.
Each cylinder assembly has an integral flow distributor that contains a dual concentric valve which provides the normal servo control and a jam tolerant function.
The flow distributor assembly is controlled through an input lever which receives the input from pilot's flight controls and through a feedback link which is connected to the output piston.
The anti-jamming device consists of a inner and outer sleeve that are held in place under the action of a spring. In case of a valve spool jam the force required at the input lever to operate the main valve exceeds a preset value thus compressing the spring and allowing the two sleeves to move relatively to control the hydraulic fluid within the cylinder chambers.
The actuator is designed to operate normally with two hydraulic systems (hydraulic system no.1 and hydraulic system no.2) which are completely separated within the actuator.
In case of failure of one system, the level of performances with only one hydraulic system operating ensures no degradation of handling qualities.
The indication system of the rotor flight controls is supplied by a pressure switch installed on each actuator control-valve.
If the control spool of an actuator control-valve jams or moves abnormally, the pressure switch will send an input to the indicating system which, in turn, will generate a caution and a 1 SERVO or 2 SERVO message depending on which system has had a jam.



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MAIN ROTOR SERVO ACTUATORS

TAIL ROTOR CONTROLS SYSTEM – GENERAL

The tail rotor controls system includes the yaw control. The yaw control is obtained by changing the pitch angle of the tail rotor (TR) blades.

YAW CONTROL – MAIN COMPONENTS

PILOT PEDAL ASSY

The pilot pedal assembly is mounted on an articulated support that allows to fit the physical characteristics of the pilot.
The pilot pedal assembly is connected to the copilot pedal assembly to give a common input to the TR servo actuator.
The yaw pedal incorporates a microswitch used to operate the trim actuator friction in order to disengage the anchor point.

COPILOT PEDAL ASSY

The copilot pedal assembly is mounted on an articulated support that allows to fit the physical characteristics of the copilot.
The yaw pedal incorporates a microswitch used to operate the trim actuator friction in order to disengage the anchor point.

YAW TRIM ACTUATOR

The yaw trim actuator receives electrical commands from the pilots or from the AFCS and convert these commands into a mechanical output used to change the position of the yaw control linkage

YAW DUAL LINEAR ACTUATOR

The yaw dual linear actuator is installed in series with the rods of the mechanical linkage inside the tail boom.

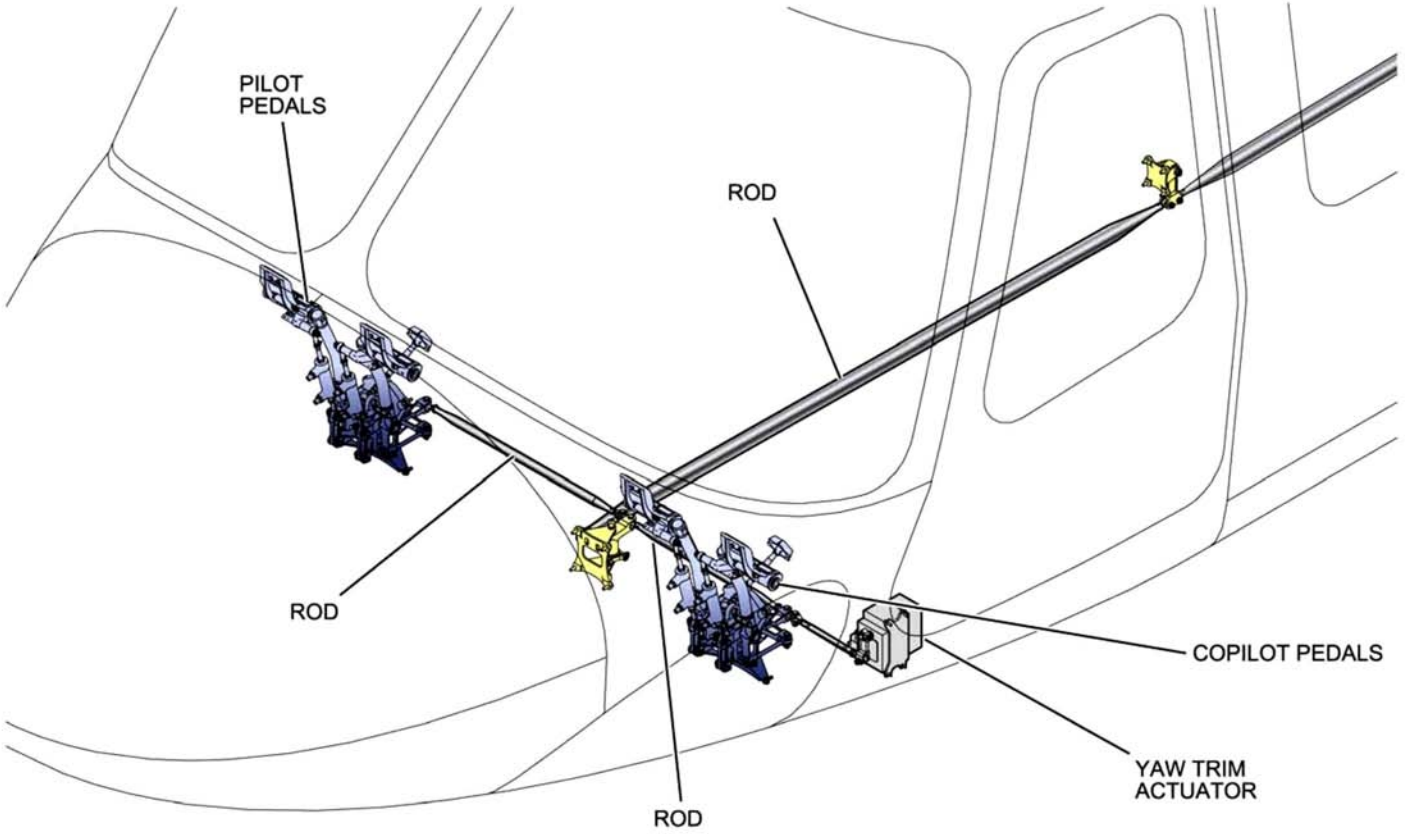
It is connected with the linkage through an anchor spring which provides a safety function in case of the actuator disconnects: in this case the spring reacts to the manual input as a pivot point to ensure the movement to the tail rotor servo actuator.

YAW CONTROL – PRINCIPLE OF OPERATION

The tail rotor control is operated by means of a mechanical linkage that gives an input to the dual-channel tail rotor hydraulic servo actuator. The TR servo actuator acts the pitch-change mechanism to give the required yaw action.

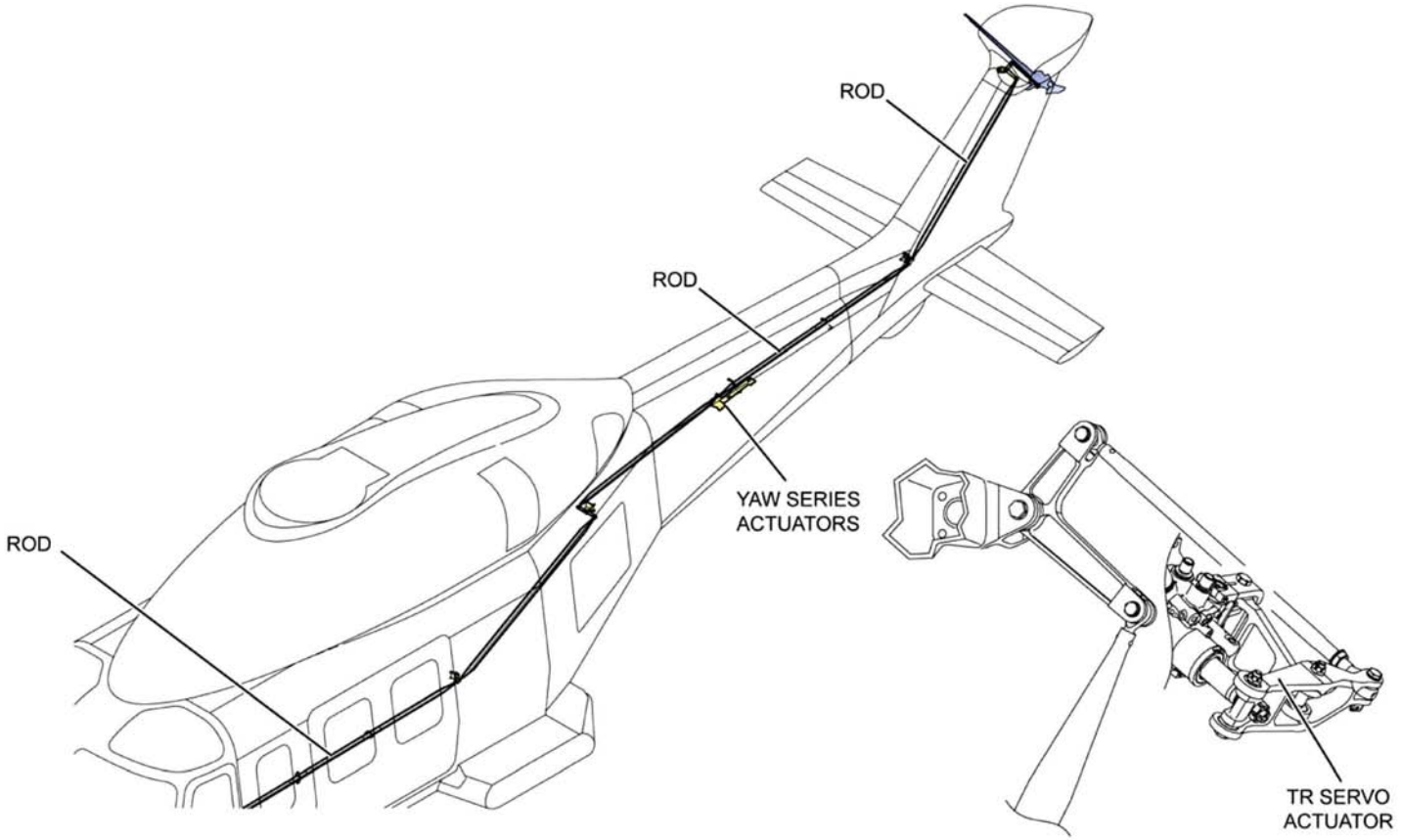
The mechanical linkage is operated by pilot and copilot pedals.

The mechanical linkage is also controlled by a yaw trim actuator.



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YAW CONTROL – MAIN COMPONENTS (SHEET 1 OF 2)



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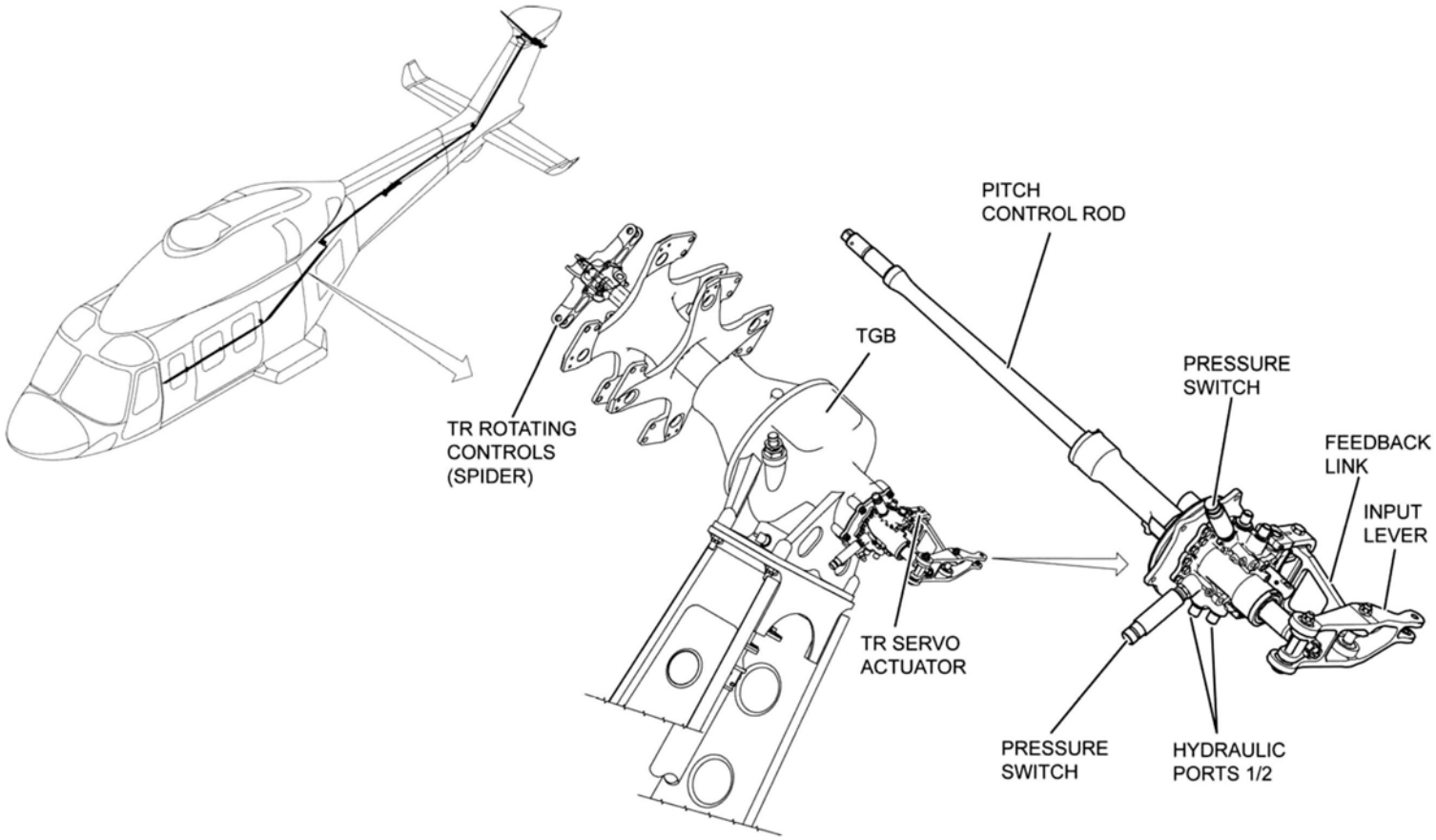
YAW CONTROL – MAIN COMPONENTS (SHEET 2 OF 2)

TAIL ROTOR SERVO ACTUATOR - GENERAL

The tail rotor servo actuator is a fixed body actuator comprising two separate cylinder assemblies, which are bolted together at the actuator center and tandem pistons.

TR SERVO ACTUATOR – PRINCIPLE OF OPERATION

The actuator body is provided with a mounting flange for the installation to the left side of the TGB. One piston end extends with a control rod which is connected to the rotating control spider; the other piston end is connected with the input lever. Each cylinder assembly has an integral flow distributor that contains a dual concentric valve which provides the normal servo control and a jam tolerant function.
The flow distributor assembly is controlled through an input lever which receives the input from pilot's flight controls and through a feedback link which is connected to the input lever and hence to the output piston.
The anti-jamming device consists of a inner and outer sleeve that are held in place under the action of a spring.
In case of a valve spool jam the force required at the input lever to operate the main valve exceeds a preset value thus compressing the spring and allowing the two sleeves to move relatively to control the hydraulic fluid within the cylinder chambers.
The actuator is designed to operate normally with two hydraulic systems which are completely separated within the actuator. In case of failure of one system, the level of performances with only one hydraulic system operating ensures no degradation of the helicopter handling qualities.



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TAIL ROTOR SERVOACTUATOR

ROTOR FLIGHT CONTROLS – CONTROLS AND INDICATORS

1 FORCE TRIM switch

- OFF ..... the cyclic force trim actuators are disengaged
- ON ..... the cyclic force trim actuators are engaged. When the pilot moves the cyclic stick out of the detent position in the pitch and the roll axis, he feels the spring force applied on the cyclic flight control

2 CLTV / YAW TRIM switch

- OFF ..... the collective and yaw pedals force trim actuators are disengaged
- ON ..... the collective and yaw pedals force trim actuators are engaged. When the pilot moves the collective lever and the yaw pedals from their detent positions, he feels the spring force applied on the collective and yaw pedals flight controls

3 FTR push-button switch

- PRESSED ..... disengages the clutch of the cyclic trim actuator. When the pilot operate the cyclic stick, he feels the flight controls free to move and the spring force is not applied
- RELEASED ..... engages the clutch of the cyclic trim actuators. The pilot feels the spring force applied to the cyclic stick

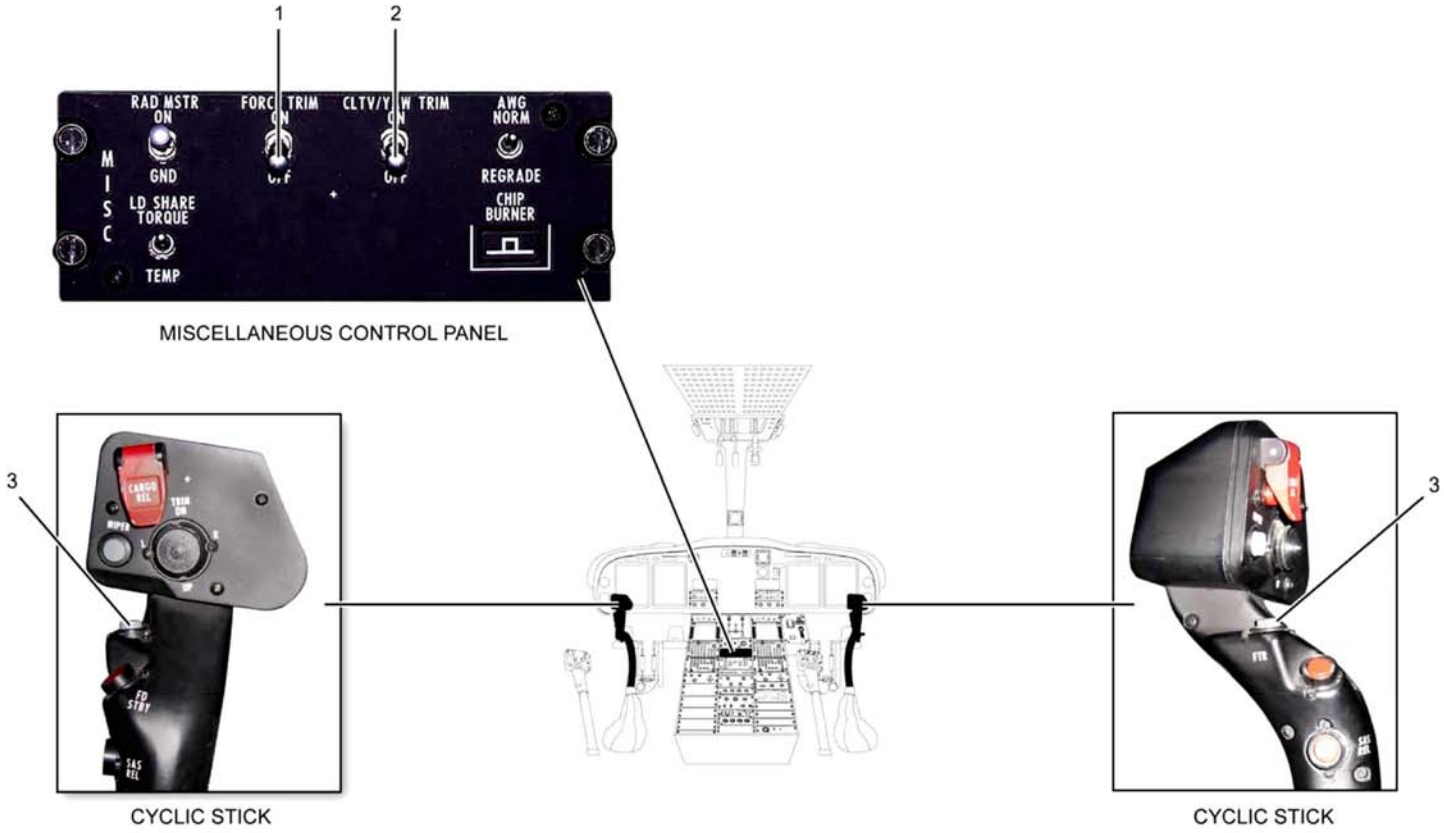
4 FTR push-button switch (CLTV PLT)

- PRESSED ..... disengages the clutch of the collective and yaw pedals trim actuators.
  NOTE. When the pilot operate the collective lever and the yaw pedals, he feels the flight controls free to move and the spring force is not applied.
- RELEASED ..... engages the clutch of the collective trim actuator and yaw pedals trim actuators. The pilot feels the spring force applied to the collective lever and yaw pedals

5 FTR push-button switch (CLTV CPLT)

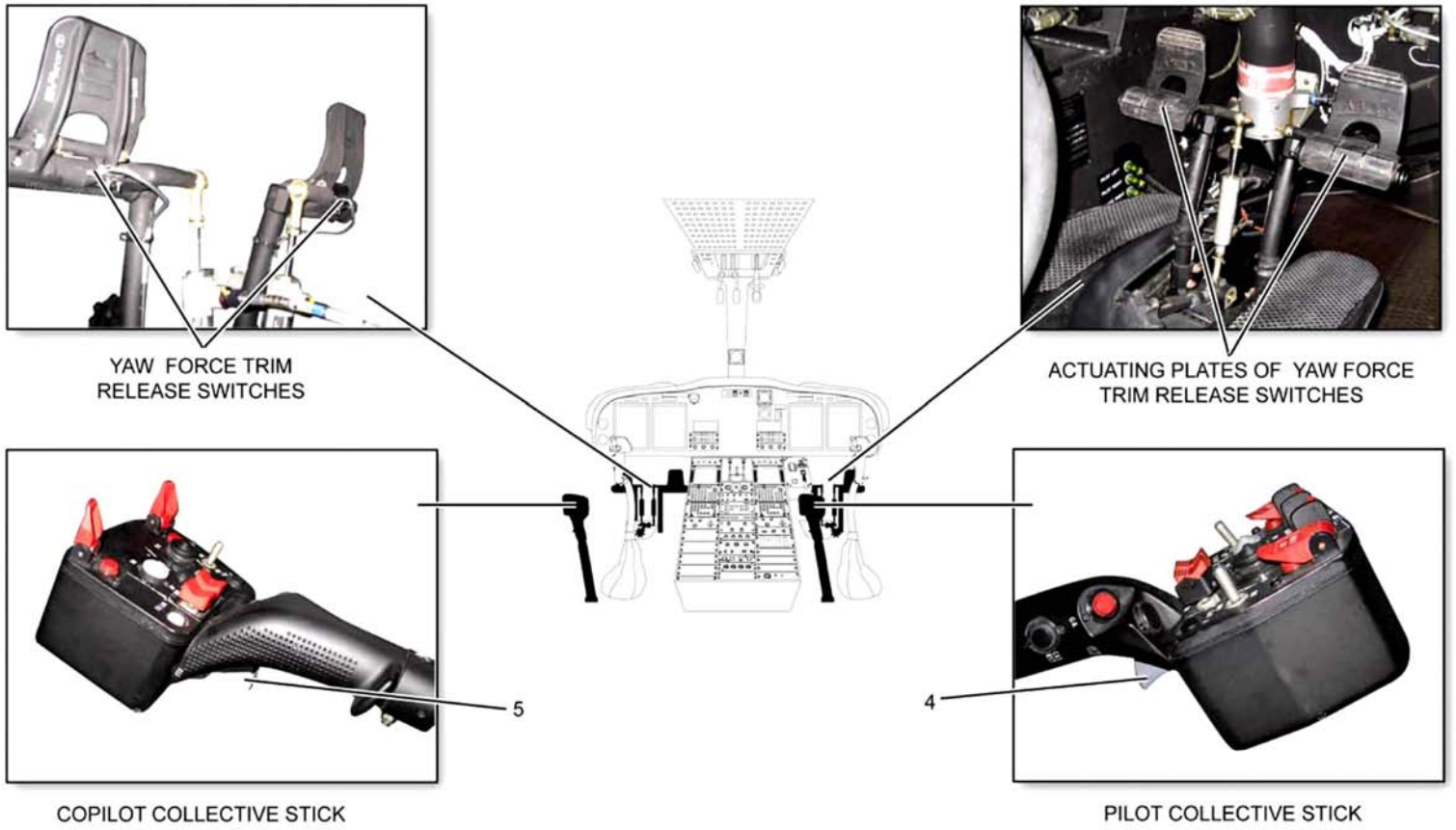
- PRESSED ..... disengages the clutch of the collective and yaw pedals trim actuators.
  NOTE. When the copilot operate the collective lever and the yaw pedals, he feels the flight controls free to move and the spring force is not applied.
- RELEASED ..... engages the clutch of the collective trim actuator and

yaw pedals trim actuators. The copilot feels the spring force applied to the collective lever and yaw pedals.



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ROTOR FLIGHT CONTROLS – CONTROLS AND INDICATORS (SHEET 1 OF 2)



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ROTOR FLIGHT CONTROLS – CONTROLS AND INDICATORS (SHEET 2 OF 2)

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SECTION 30 - SERVO-CONTROL SYSTEM

DESCRIPTION

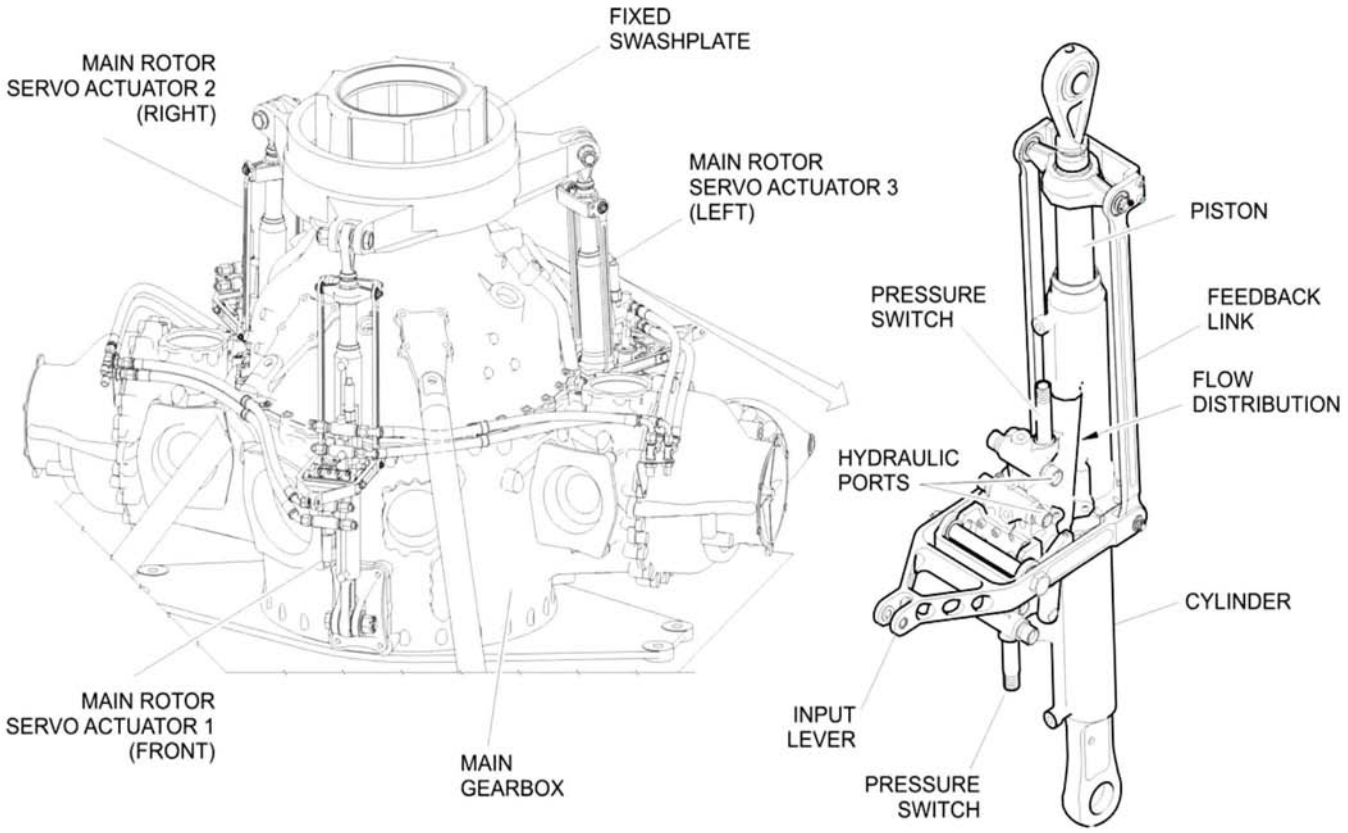
Main Rotor Servo Actuator

The main rotor servo actuator is a fixed body actuator comprising two separate cylinder assemblies, which are bolted together at the actuator centre, and tandem pistons.
The output upper piston end is provided with a terminal eye for connection to the swashplate.
The lower cylinder end is shaped as a terminal eye for connection with a support to the Main Gearbox.
The cylinder and piston are manufactured in stainless steel; the piston is chromed to improve wear properties.
Each cylinder assembly has an integral flow distributor that contains a dual concentric valve which provides the normal servo control and an anti-jamming device.
The flow distributor assembly is controlled through an input lever which receives the input from pilot's flight controls and through a feedback link which is connected to the output piston.
The anti-jamming device consists of an inner and outer sleeve that is held in place under the action of a spring. In the event of a valve spool jam the force required at the input lever to operate the main valve exceeds a preset value thus compressing the spring and allowing the two sleeves to move to control the hydraulic fluid within the cylinder chambers.
The actuator is designed to operate normally with two hydraulic systems which are completely separated within the actuator.
The part where the two cylinders are joined is vented to atmosphere to ensure that leakage past the rod seals does not cause inter-system mixing of the fluid.
In case of failure of one system, the level of performance with only one hydraulic system operating ensures no degradation of the helicopter handling qualities.
The Main Rotor Servo actuators are interfaced with:

- Main Gearbox
- main rotor rotating controls
- PCM1 and PCM2
- main rotor flight controls
- Modular Avionics Unit

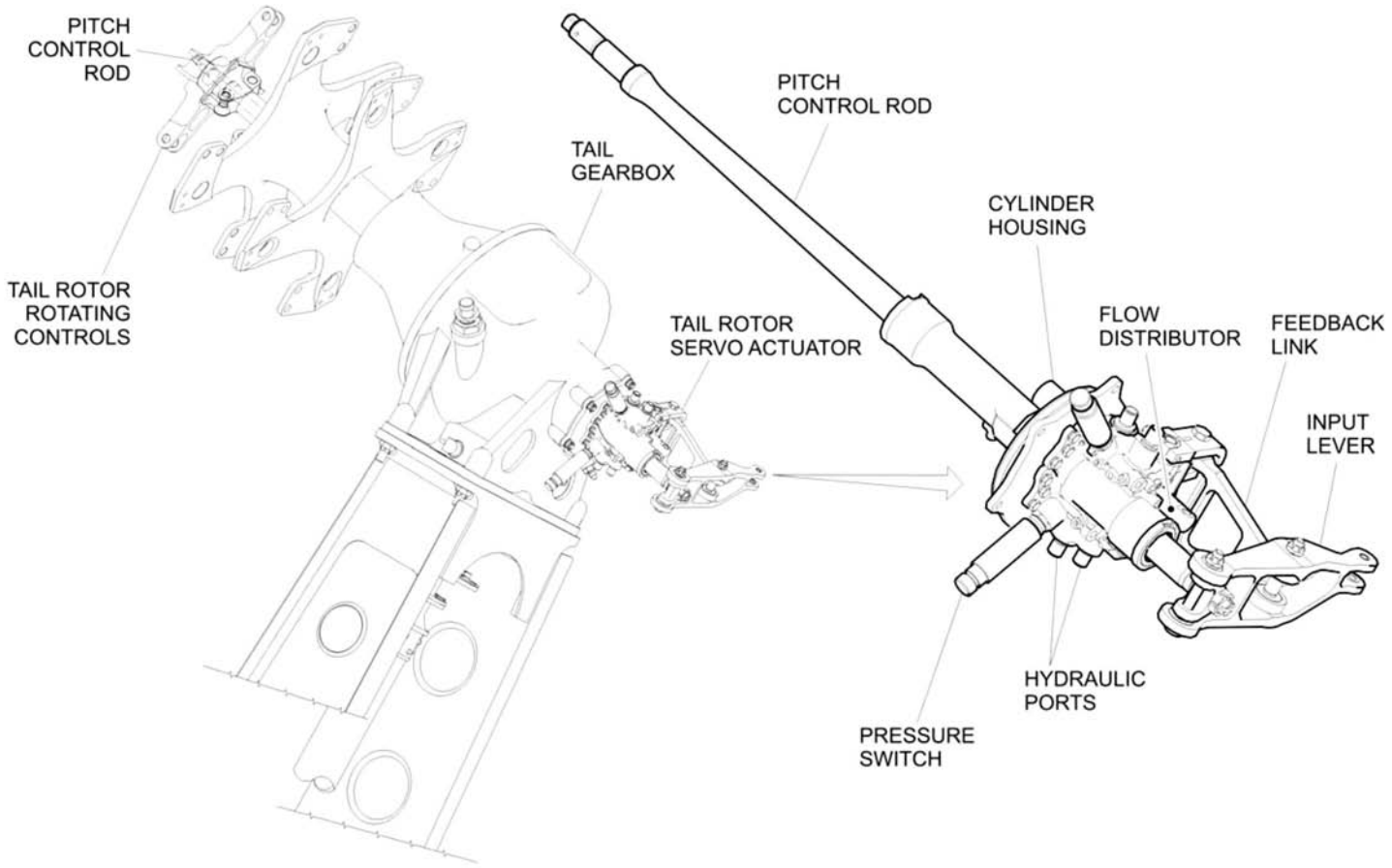
Tail Rotor Servo Actuator

The tail rotor servo actuator is a fixed body actuator comprising two separate cylinder assemblies, which are bolted together at the actuator centre, and tandem pistons. The actuator body is provided with a mounting flange for the installation to the left of the Tail Gearbox.
One piston end extends with a control rod which is connected to the rotating control spider; the other piston end is connected with the input lever.
The cylinder and the piston are manufactured in stainless steel; the piston is chromed to improve wear properties.
Each cylinder assembly has an integral flow distributor that contains a dual concentric valve which provides the normal servo control and an anti-jamming device.
The flow distributor assembly is controlled through an input lever which receives the input from pilot's flight controls and through a feedback link which is connected to the input lever and hence to the output piston.



ICN-39-A-673000-G-A0126-01009-A-01-1

MAIN ROTOR SERVO ACTUATOR



ICN-39-A-673000-G-A0126-01010-A-01-1

TAIL ROTOR SERVO ACTUATOR

OPERATION

Normal Operation

The input lever transmits the pilot's input through a linkage and layshaft to the flow distributor assembly in each half of the actuator.
Displacement of the input lever results in movement of the servo valves from the neutral position routing the high pressure hydraulic fluid into the appropriate cylinder chamber.
The resulting piston motion drags a dual load path feedback link to rotate the input lever and reset the valve command.
Movement of the input lever results in the piston moving in the opposite direction.

EMERGENCY OPERATION - HYDRAULIC FAILURE

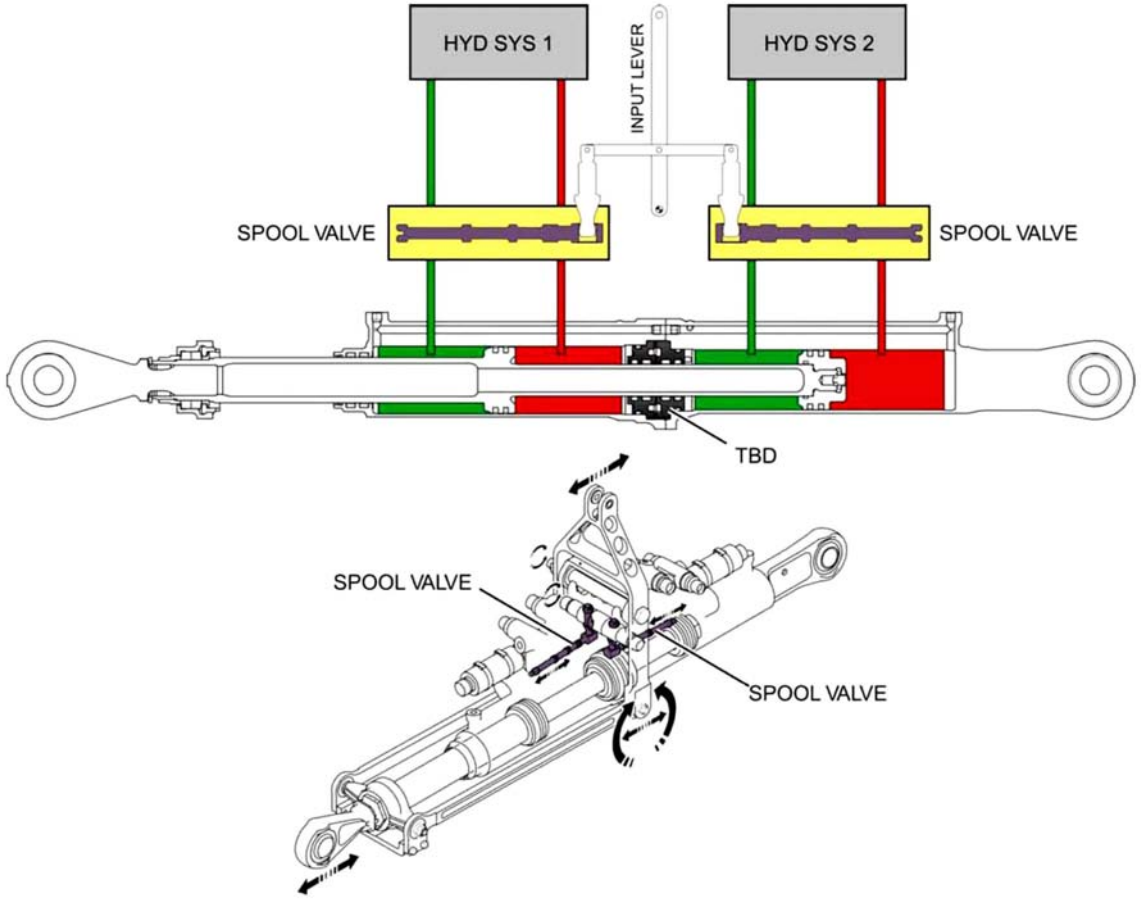
The servo valve ports in the failed half of the actuator (pressure loss) are controlled in the normal way allowing fluid to be forced from the dead cylinder by the active half.
This bypass condition is also monitored to the cockpit (in addition to pressure indication) by the microswitch installed on the flow distributor providing the caution 1 (2) SERVO to the MFD.

EMERGENCY OPERATION - VALVE JAM

Depending on the main spool valve jammed position, the dual concentric sleeve valve allows two ways of operation:

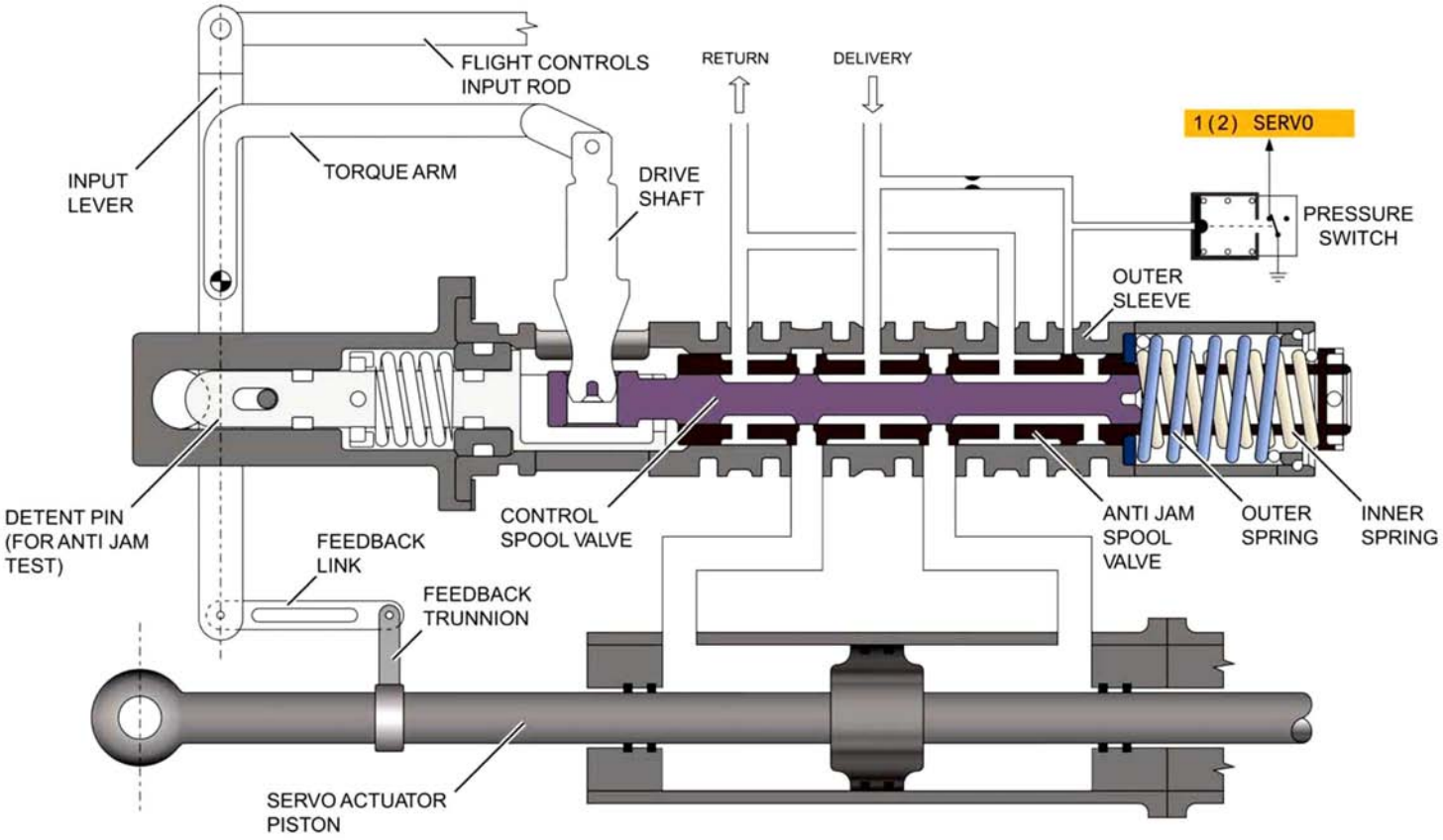
- If the main spool valve is jammed in the centre position (ports in the inner sleeve closed), the sleeve moves to distribute the fluid to and from the cylinder in the normal way in response to input commands.
  This condition is defined as “actuator active”.
- If the spool valve is jammed away from the centre position, the sleeve moves to connect together both control ports, to and from the cylinder, to the return line.
  This is a bypass condition and the piston is moved by the remaining active half of the actuator.

In the first condition the pressure line connected to the microswitch is drained to return only if flight controls are operated so that the caution 1 (2) SERVO is provided to the MFD.
In the second condition the pressure line connected to the microswitch is always drained to return so that the caution 1 (2) SERVO is always provided to the MFD.
A maintenance check may be performed on the anti-jam device to ensure that the function is working correctly and that no dormant failures exist in the components.
The maintenance check consists of applying pressure to the actuator and operating the anti-jam device through a plunger which is externally operated by a special tool and observing the cockpit indication 1 (2) SERVO to the MFD.



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SERVO ACTUATOR – SCHEMATIC



ICN-39-A-673000-G-A0126-01544-A-01-1

SERVO ACTUATOR – SCHEMATIC

CAS CAPTION	FAILURE DESCRIPTION
1(2) SERVO	Associated hydraulic servo actuator in bypass NOTE A When the flight controls are operated and a main spool valve in any servo actuator is jammed in center or a main spool in any servo actuator is jammed out of the center regardless of flight controls operation (the caution is also provided when the pressure in the hydraulic circuit is below 138 bar). **WARNING** DO NOT SWITCH SOV TO CLOSE ON THE UNAFFECTED SYSTEM SINCE THIS WILL CAUSE LOSS OF CONTROL IN THE AFFECTED SERVO JACK NOTE B Loss of hydraulic fluid in system no.2 will automatically close the Tail Rotor Shut Off Valve (TRSOV). This will be indicated by a 2 SERVO caution on the CAS and a TRSOV closed indication on the hydraulic synoptic page. Once the TRSOV has operated the SOV no.1 is inhibited

MAINTENANCE INFORMATION

- PCM1 and PCM2 must be drained before main and tail servo actuator removal
- Tail gearbox must be drained before tail rotor servo actuator removal
- After a main rotor servo actuator replacement you must carry out:
  - filling and bleeding hydraulic circuits 1 and 2
  - adjustment of the mixing control system
  - operational check of the hydraulic system 1 and 2
- After a tail rotor servo actuator replacement you must carry out:
  - filling and bleeding hydraulic circuits 1 and 2

- filling the tail gearbox
- operation test of the hydraulic system 1 and 2.

LEADING PARTICULARS

- MR and TR servo actuators operating inlet pressure (static/ dynamic)..... 207 bar / 196 bar (3000 psi / 2843 psi)
- MR and TR servo actuators operating return pressure (static/ dynamic)..... 1 bar / 10 bar (14.5 psi / 145 psi)
- Input lever control load (normal operation)..... from 3N to 16N (o.2 to 3.6 lb)
- Input lever control load (jamming condition)..... from 70N to 100N (15.7 to 22.5 lb)
- Hydraulic fluid..... MIL-PRF-83282
- MR piston rated speed..... 100 mm/sec (3.94 in/sec)
- TR piston rated speed..... 50 mm/sec (1.97 in/sec)
- Bypass/jamming condition microswitch operation
  - decreasing (ON)..... 138 bar (2001 psi)
  - increasing (OFF)..... 179 bar (2596 psi)
- MR weight..... 8.6 Kg (19 lb)
- TR weight..... 7.0 Kg (15.4 lb)

SAFETY PRECAUTIONS

- The solvent is a dangerous material. Make sure that you know all the safety precautions and first aid instructions for the solvent.
- Make sure that you obey all applicable precautions when you use the compressed air.
- Be careful when you remove/install the main servo actuator. The main servo actuator is heavy. An incorrect movement can cause an injury to person and/or damage to the equipment. In this procedure, get a second person to help you.

SPECIAL TOOLS/TEST EQUIPMENT

MR/TR servo actuator by-pass simulator tool (part of flight controls adjustment tool kit).

CHAPTER 71

POWER PLANT

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SECTION 00 - POWER PLANT

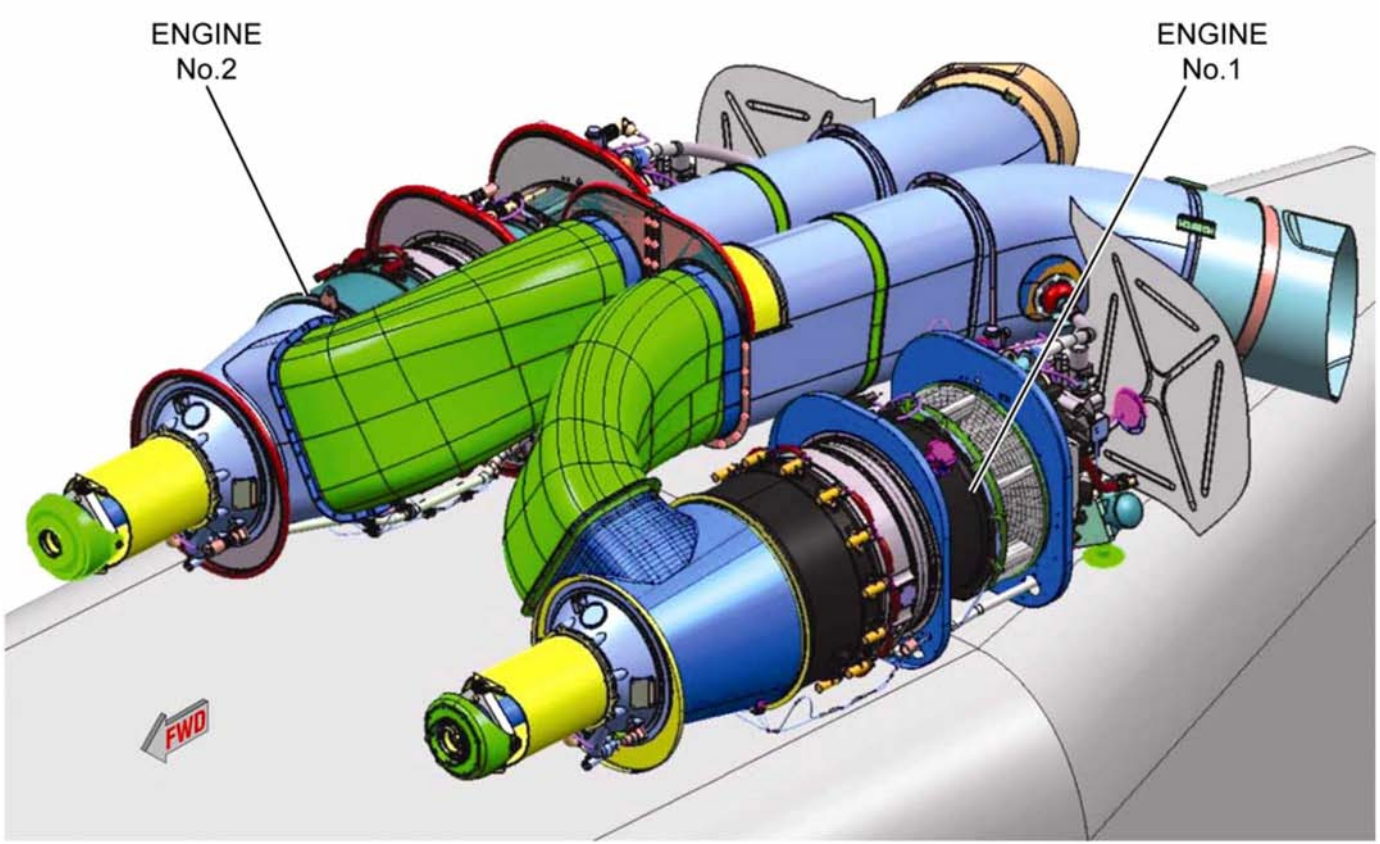
POWER PLANT – GENERAL

The AW139 is powered by two Pratt & Whitney Canada PT6C-67C turbine engines.
Each engine is a free turbine turbo-shaft propulsion engine that includes a four stage axial compressor and a one stage centrifugal compressor. The compressors are driven by a single-stage compressor turbine.
The two stage power turbine drives the output shaft which supplies power directly to the input gears of the MGB.
An Electronic Engine Control (EEC) unit with an hydro-mechanical Fuel Management Module (FMM), ensures automatic control of the engine and fast response changes in power demand.
The Engine Control Lever (ECL) provides manual back-up in case the automatic engine control system fails.
The engine indicating system includes sensors and probes installed on the engine that check the status of the engines.
Each engine supplies 1872 SHP in OEI condition.



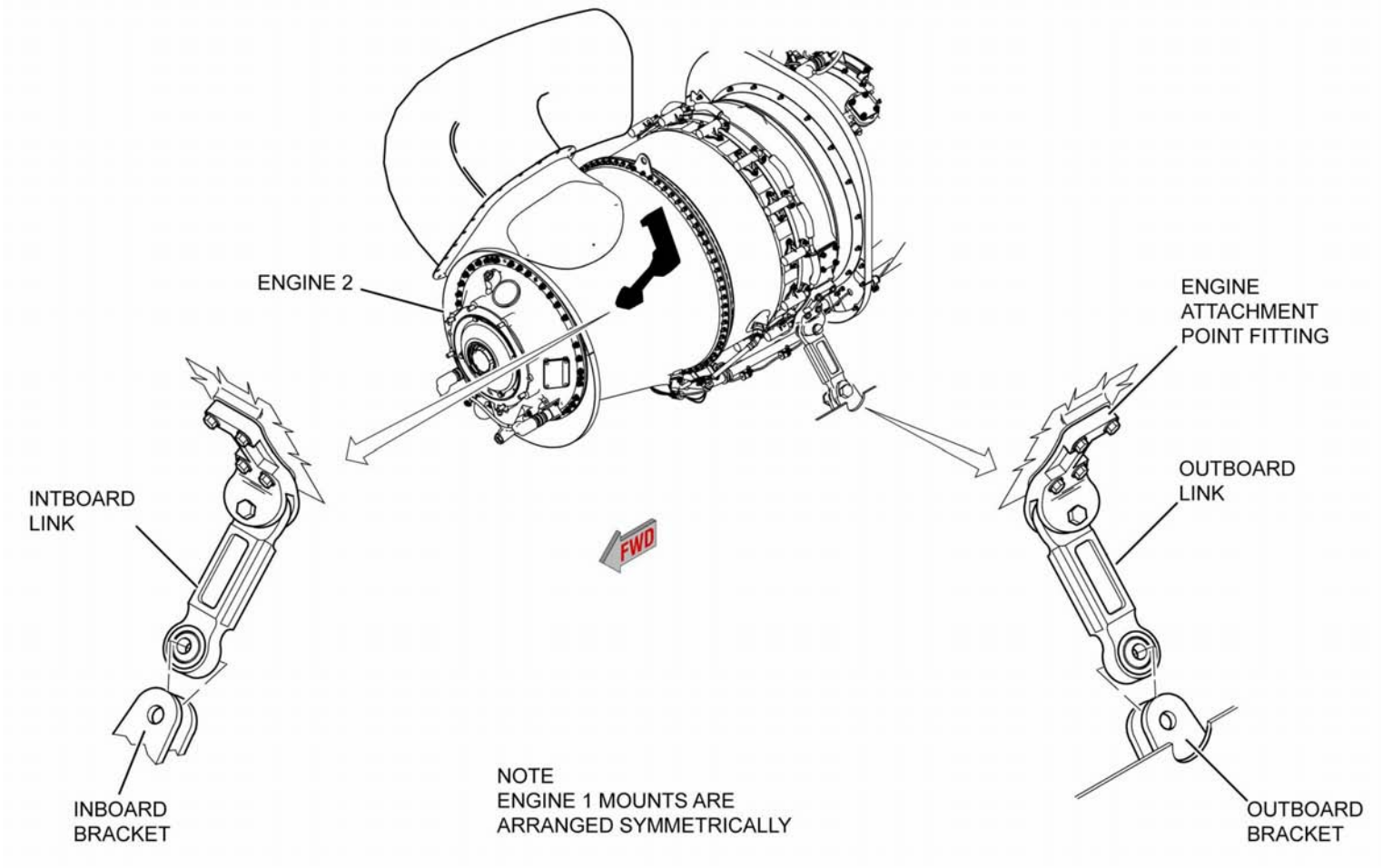
ENGINE INSTALLATION AND MOUNTS

The engines are installed in separated fireproof compartment on the upper deck.
The engines are installed with their output shaft facing forward to drive the MGB directly.
Exhaust gases are therefore routed aft of the engine air intakes via exhaust ducts installed between the two engines.
Each engine is attached to the helicopter airframe by means of two engine mounts. Each engine mount consists of a link to connect the inboard or the outboard engine attachment point to a bracket on the structure.
The engine mounts (located in the rear part of the engine) have the function to absorb shear loads in both vertical and lateral direction, thus allowing for engine thermal expansion.
A third attachment point for the engine is provided by the torque tube and crosshead device that connect the engine output section to the input module of the MGB.
The torque tube and crosshead device distribute and absorbe the axial and radial load produced by engine drive shafts.



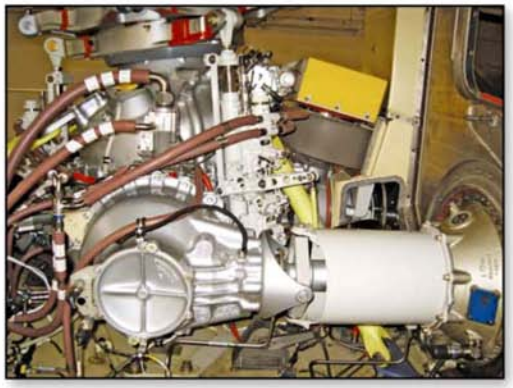
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ENGINE INSTALLATION



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ENGINE MOUNTS



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ENGINE TORQUE TUBE AND CROSSHEAD

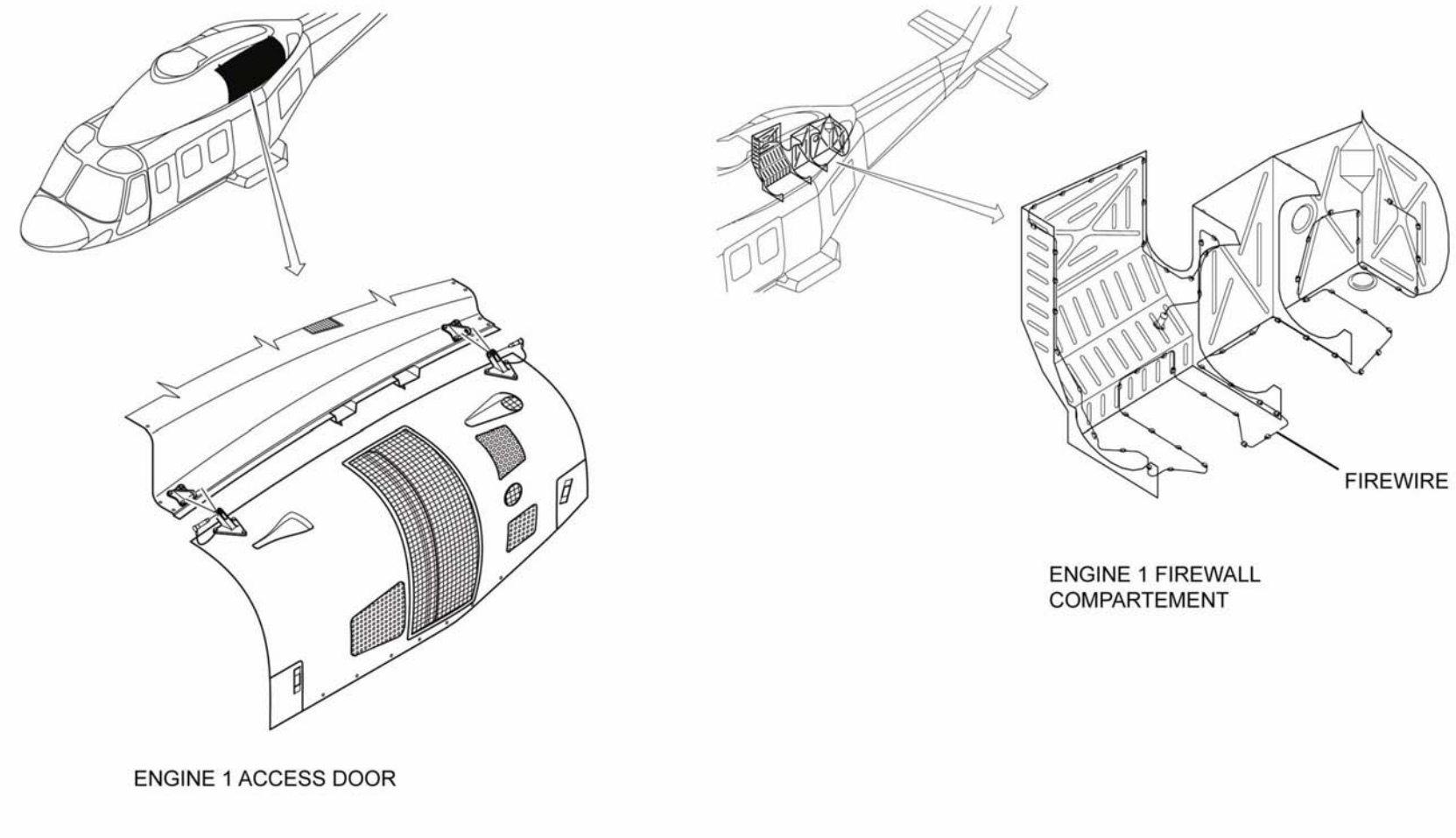
ENGINE COMPARTMENTS

ENGINE FIREWALLS

Engine firewalls structures surround the compartment on three sides and are made of titanium.
The forward side firewall is split to allow engine removal/installation. Firewalls are shaped to accommodate the exhaust collector in the relative area between the two engines.
The firewire element is installed along the engine firewall (see chapter 26).

ENGINE ACCES DOOR

The engine access-door is installed on the left side of the aft fuselage and gives access to the engine. The engine access-door has two hinges on the top and two latches in the two bottom corners. A screen for the engine air intake is installed in the middle of the engine access-door. The screen can be removed.



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FIREWALL COMPARTMENTS

ENGINE – MAIN COMPONENTS

OIL FILLER CAP

Removing the oil filler cap, is possible to perform the refilling of the integrated oil tank.

OIL SIGHT GLASS (LEFT AND RIGHT)

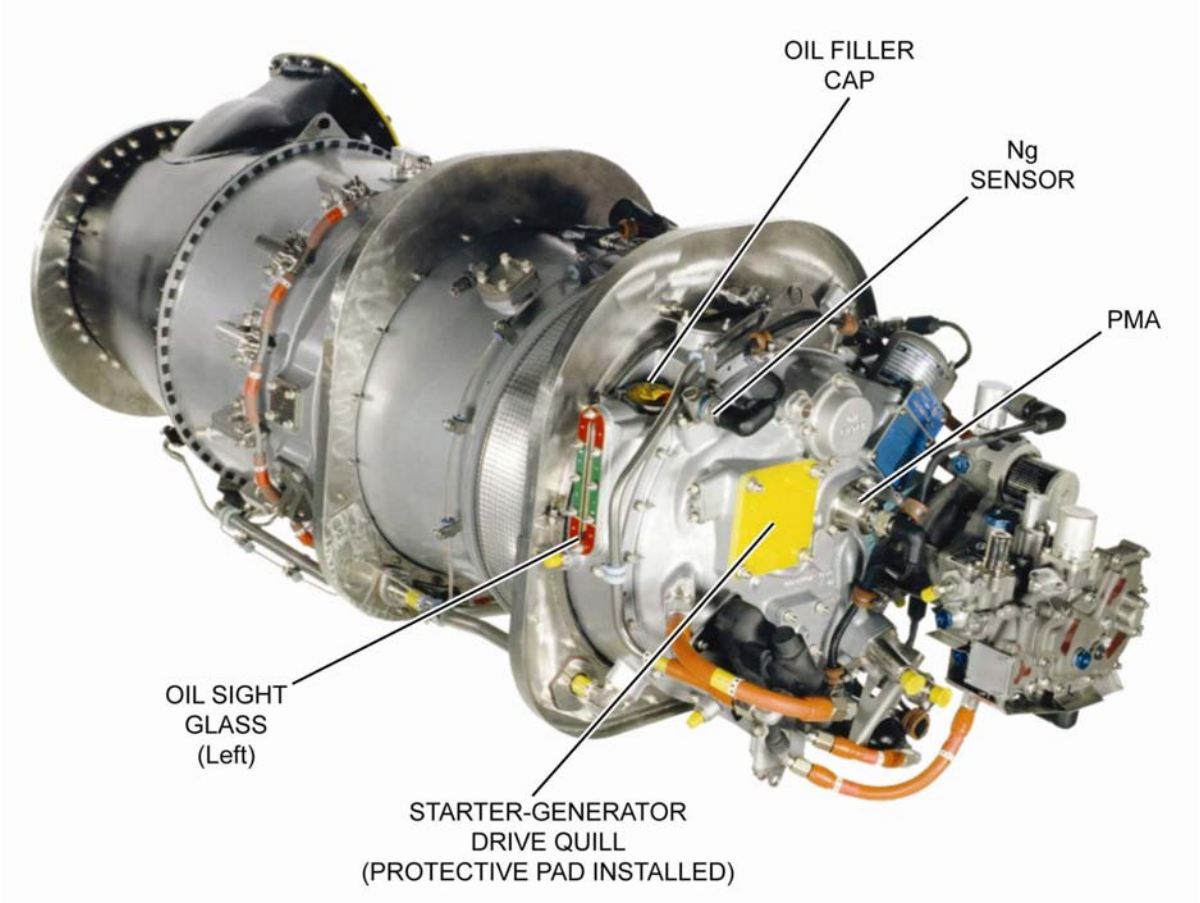
Two oil sight glass are installed, one on the right and one on the left side of the engine oil tank to permit checking the oil quantity.

NG SENSOR

The Ng sensor is located on the top left side of the engine accessory gear box. It is a magnetic sensor which uses the starter - generator gearshaft teeth as speed reference.

PERMANENT MAGNETIC ALTERNATOR (PMA)

The PMA provides the primary power source for the EEC unit when the engine is running above 40% Ng.
The PMA is an integral part of the accessory gearbox.



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ENGINE MAIN COMPONENTS

FUEL HEATER

The fuel heather preheats the fuel to prevent a fuel filter restriction caused by any possible ice VNE formation. It also allows engine operation at low outside air temperatures without using fuel additives.

DATA COLLECTION UNIT (DCU)

The Data Collection Unit is an electronic memory device that is installed on the engine and interfaces with the EEC via the engine electrical harness. It stores engine trim data and records engine fatigue and exceedance data.

COMPRESSOR BLEED VALVE

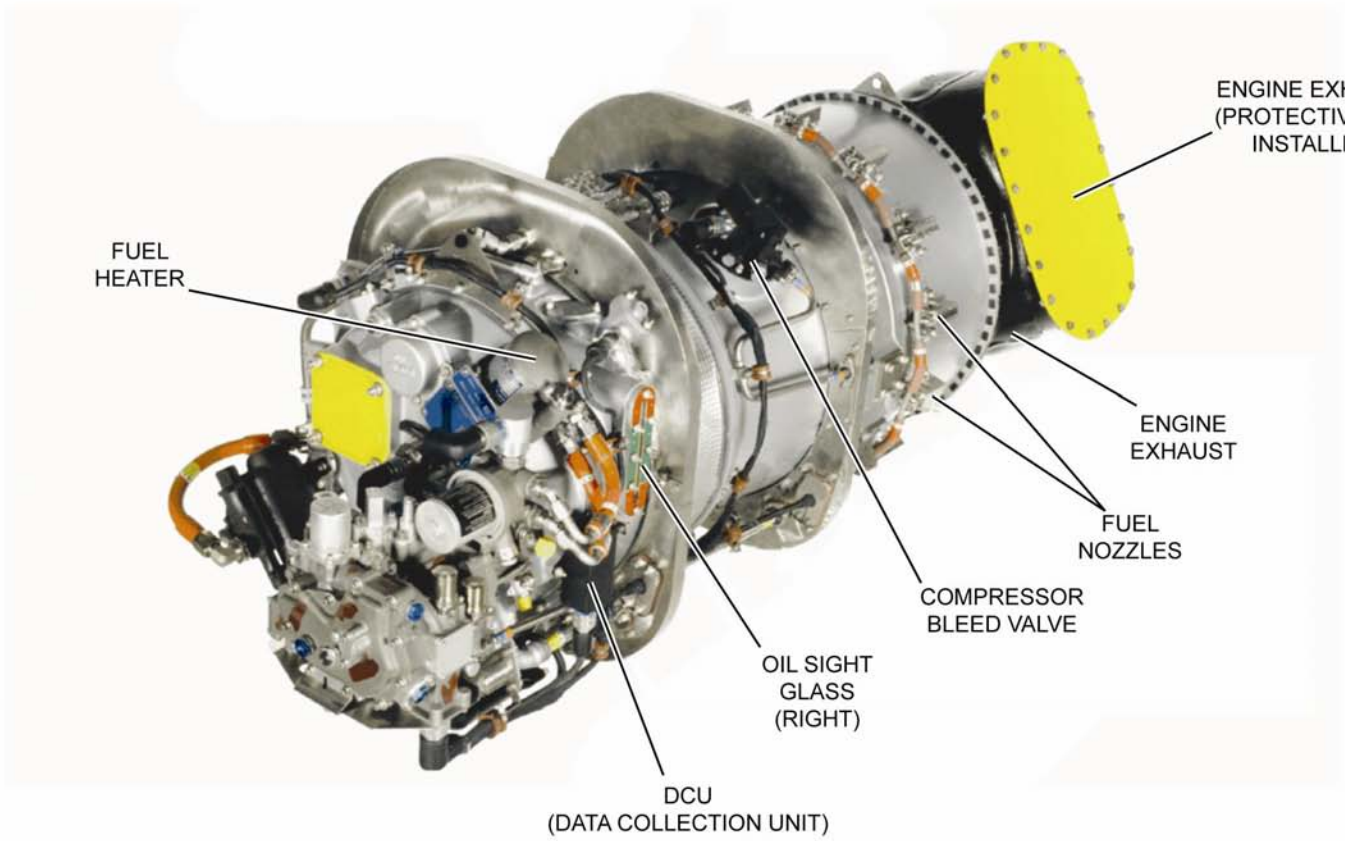
The compressor bleed valve prevents compressor stall or surges at low speed Ng values.

FUEL NOZZLES

The Fuel nozzles deliver and atomize the metered fuel into the combustion chamber.

ENGINE EXHAUST

The engine exhaust directs the exhaust gases to the exhaust duct.



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ENGINE MAIN COMPONENTS

OIL FILTER (WITH BYPASS)

The oil filter traps foreign particles that may contaminate the oil as it lubricates the various engine components. The oil filter assembly is equipped with a bypass valve to prevent oil starvation in the event of oil filter blockage.

OIL FILTER POP-OUT (INMPEDING BY PASS INDICATOR)

The Oil Filter pop-out indicates that the oil flow through the filter is becoming restricted. A red button pops-out when the pressure drops across the filter.

OIL CHIP DETECTOR

The engine oil chip detector is installed on the top RH side of the accessory gearbox.
It detects the presence of metal chips in the engine oil.

STARTER GENERETOR DRIVE QUILL

The starte- Generator drive quill provides drive for the starter-generator.

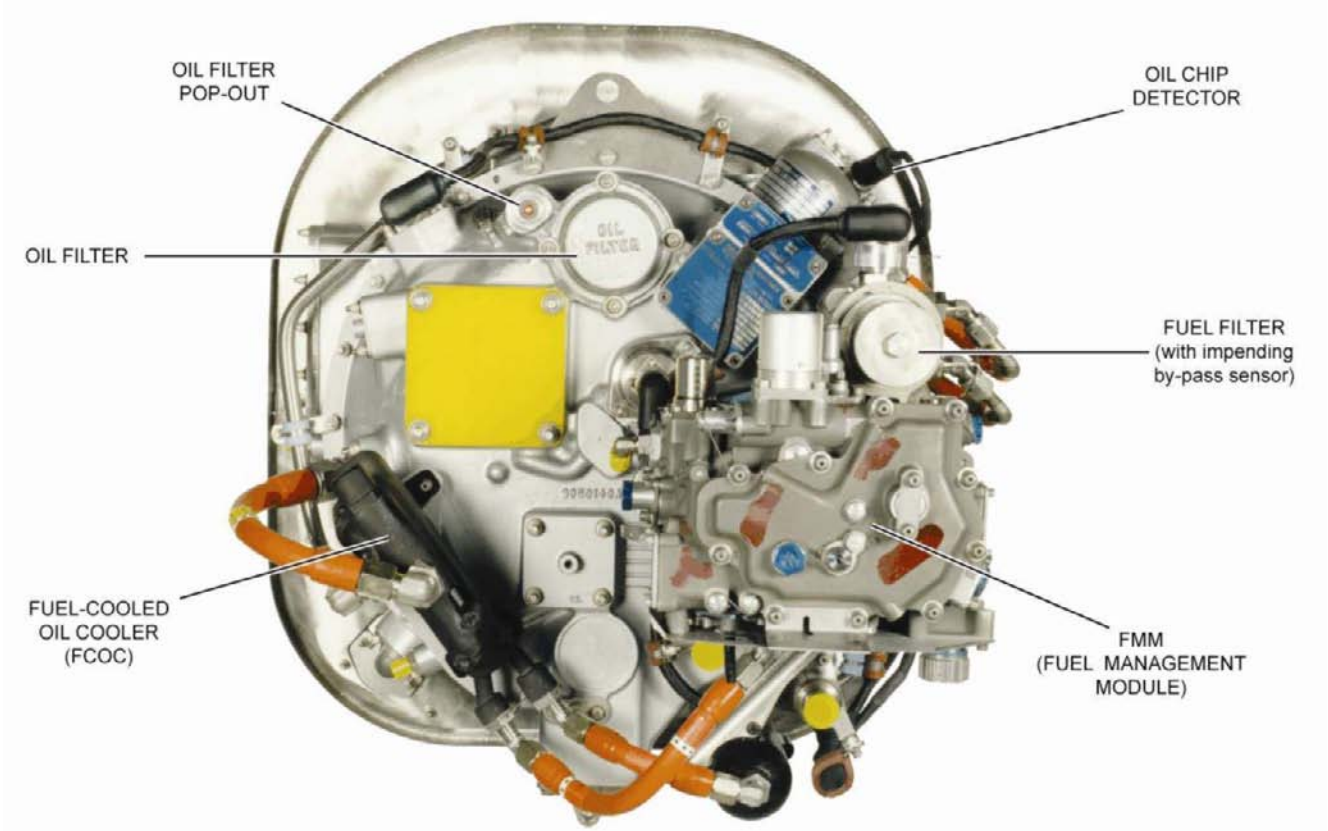
FUEL MANAGEMENT MODULE (FMM)

The FMM works in conjunction with the EEC in order to control the metered fuel flow over the entire operational range of the engine.

FUEL FILTER

The fuel filter traps any possible foreign particles or ice crystal present in the fuel.
The fuel system is equipped with a bypass valve, to prevents fuel starvation in the event of fuel filter blockage.

An impending bypass sensor switch detects pressure drop across the filter and triggers the “FUEL FILTER” caution message in the CAS.

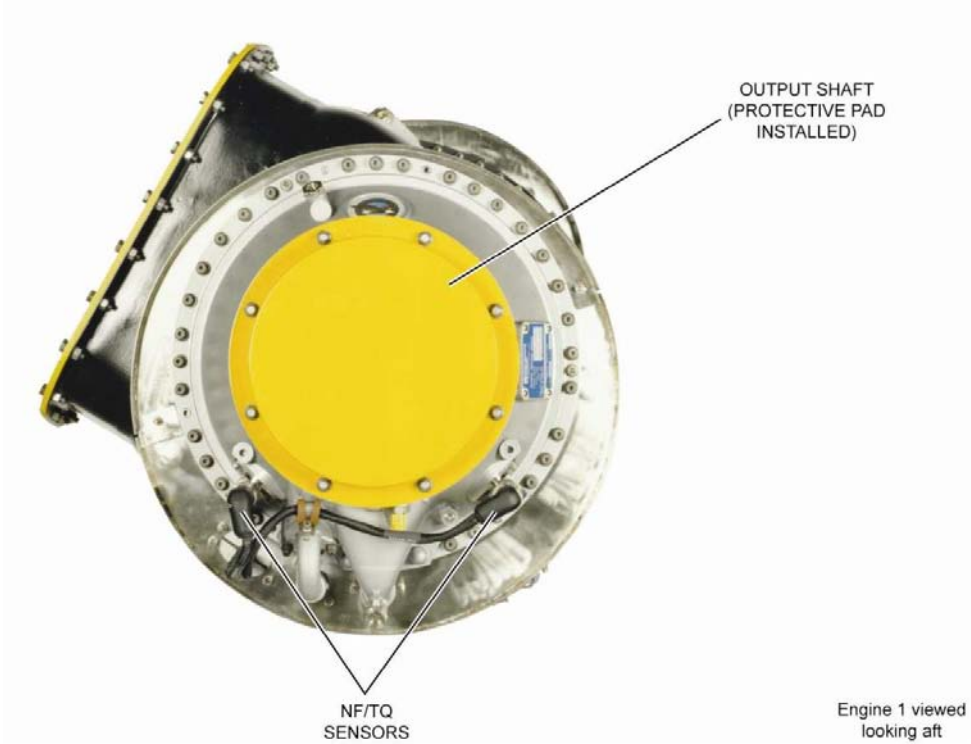


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ENGINE MAIN COMPONENTS

NF/TQ SENSORS

On the PT6C-67C engine two independent NF/TQ sensors are installaed. Each sensor detects both NF speed and engine Torque. They are located on the bottom of the No. 5 bearing support housing.
They are double magnetic coil sensor which use the torque shaft toothed wheel in the output section as the spee/torque reference.
The sensors detect the passage of the torque shaft teeth and generate an AC electrical signal due to the change in the magnetic flux associated with the proximity of the gear tooth in relation to the sensor tip.
The sensors are interfaced with the EEC and the cockpit display, via the engine wiring harness.

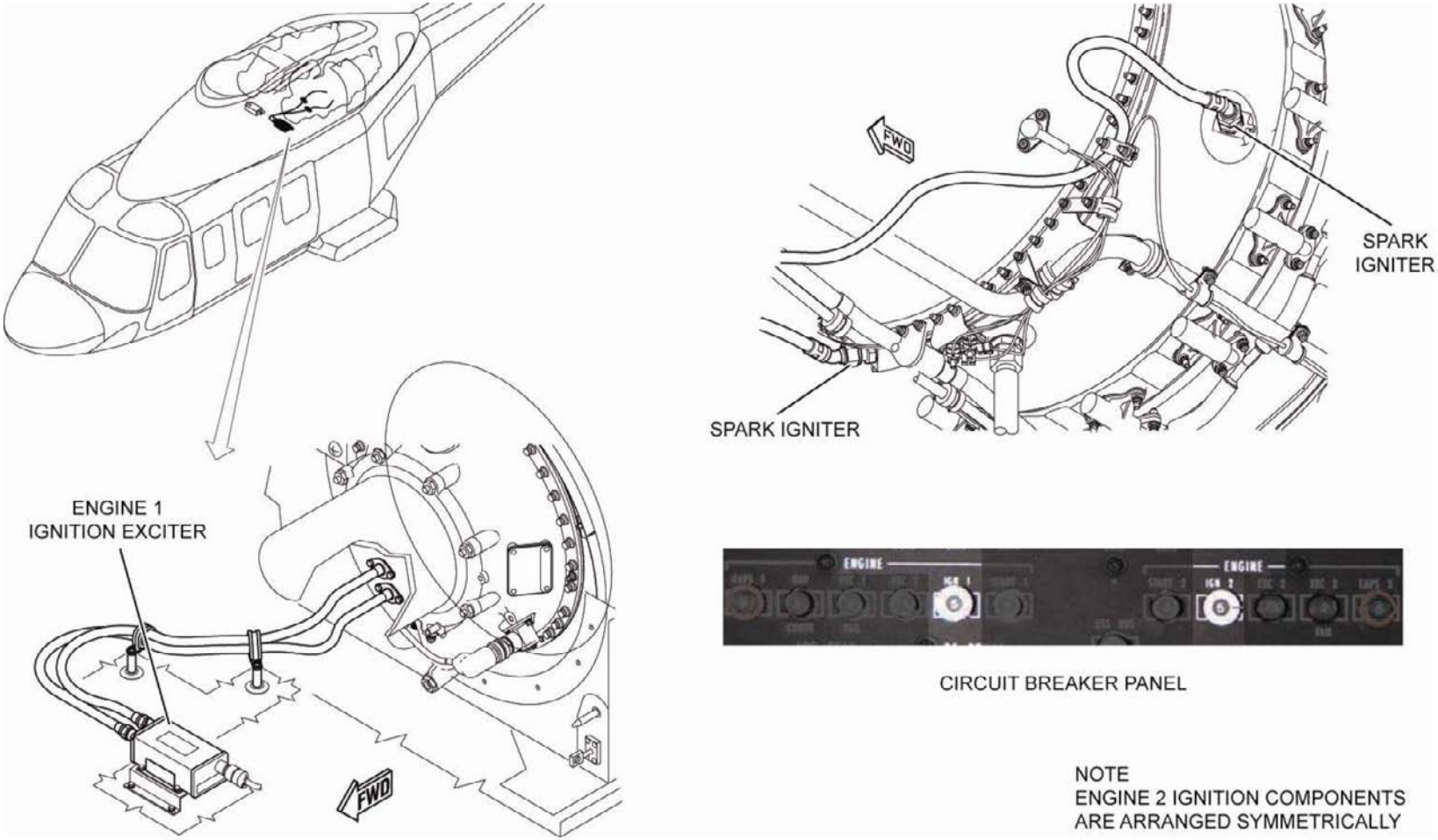


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ENGINE MAIN COMPONENTS

IGNITION

Ignition system provides the initial spark to ignite the fuel-air mixture into the combustion chamber. Each engine has its proper ignition system that consists of one ignition exciter, two high-tension cable assemblies and two spark igniters. The exciter transforms the 28V DC input into a pulsed high voltage output.
The ignition system operates during the engine starting phase only.
The ignition systems of engines 1 and 2 are supplied by ESS 1 and ESS 2 buses via the IGN 1 and IGN 2 circuit breakers, respectively.
The IGN circuit breakers are fitted with a white plastic ring that allows the pilot to quickly locate them and manually trip them for engine motoring.

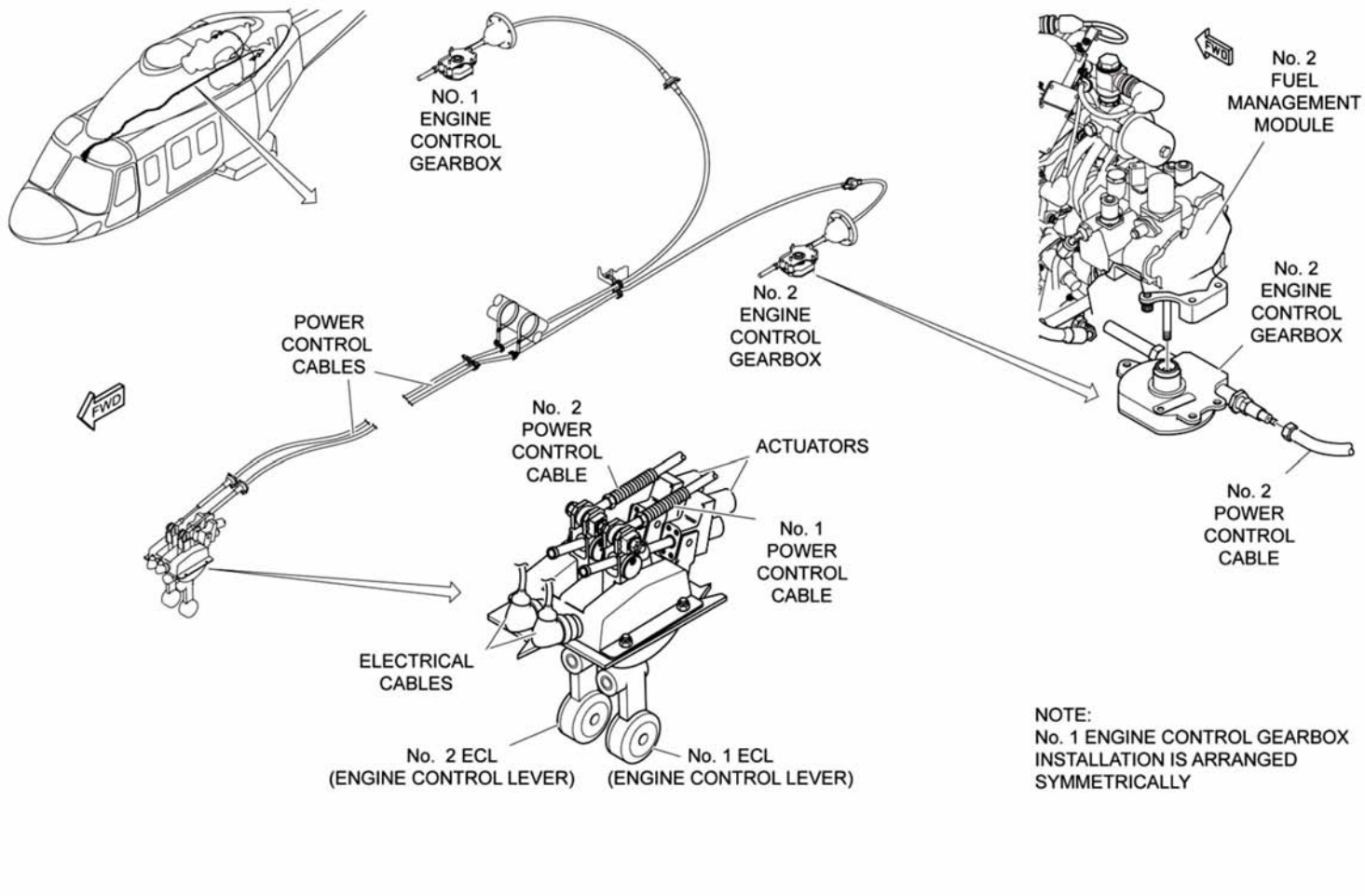


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IGNITION COMPONENTS

ENGINE CONTROL LEVER (ECL)

ECL 1 and ECL 2 are installed on the overhead console. They are connected to the respective engine control gearbox, through a power control cable.
Each engine control gearbox is mechanically connected to the respective Fuel Management Module to permit controlling the engine power manually in case of critical failure of the EEC.
When an ECL is moved to OFF it mechanically closes a fuel shutoff valve on the FMM, thus ensuring the engine shut down.



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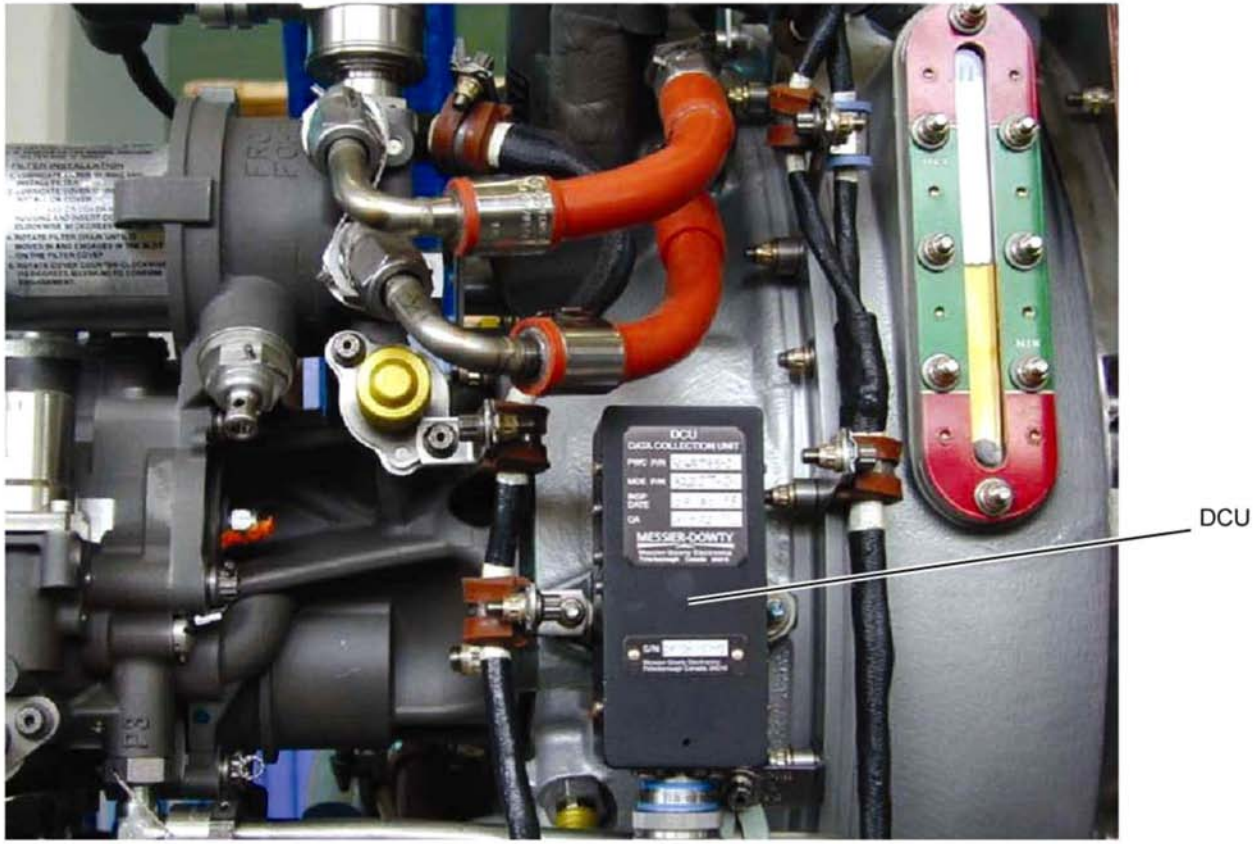
ENGINE CONTROL LEVER INSTALLATION

DATA COLLECTION UNIT

Data Collection Unit, is an electronic memory device installed on the engine and interfaces with the Electronic Engine Control via the engine electrical harness.
DCU function:

- Provides ITT and torque trim values to the EEC
- Store the folling information:
  - Engine serial number
  - Cycles
  - Engine running time (not flight hours)
  - Exceedances detected by the EEC
  - System fault codes detected by the EEC.

The trim values stored into the DCU are specific to the engine serial serial number.



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DATA COLLECTION UNITS

ELECTRONIC ENGINE CONTROL

The Electronic Engine Control (EEC) is an electronic package which contains all the components necessary for effective automatic control of the engine and helicopter rotor system. The EEC is installed in the aft electronic compartment.

EEC FEATURES

- Single channel control computer
- Starting and shutdown supervision
- Auto-start with ITT limiting logic
- Power management
- Independent NF overspeed protection
- Torque limiting logic
- Fault detection and display management
- Cross-talking function

In case of EEc failure, back-up control is achieved by the pilot who manually adjusts (sets) the FMM.

DESCRIPTION

The EEC consists of two functionally indipendent subsystems:

- The “control subsystem” which provides control of the engine fuel flow.
- The “limiter subsystem” which provides independant hardwere protection against inadvertent power turbine overspeed.

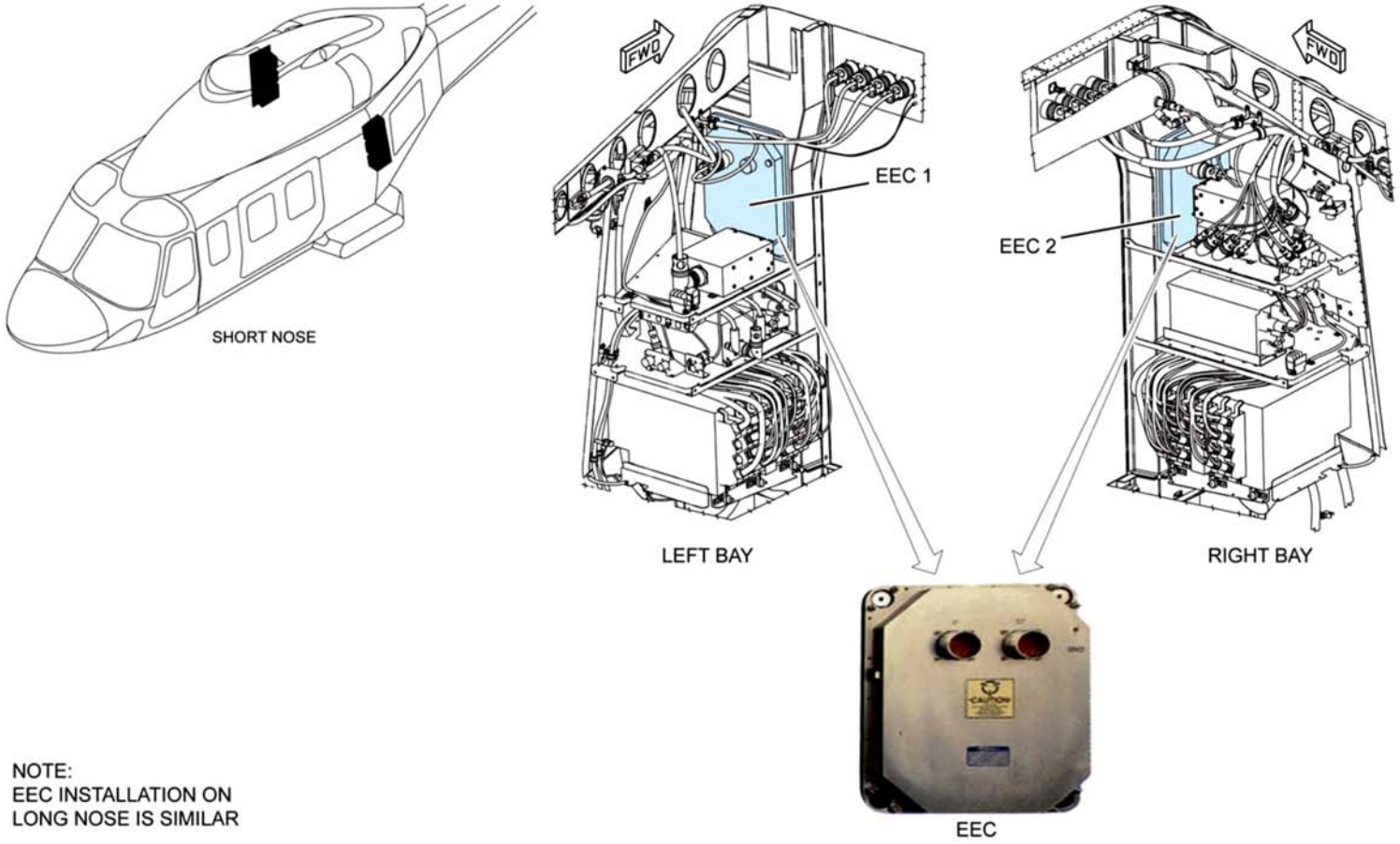
Both subsystems are isolated from each other, including power supplies, signal conditioning interface and signal inputs

OPERATION

The EEC monitors all engine parameters and controls the FMM providing automatic adjustment of the fuel flow delivered to the engine.
The EEC controls fuel flow from start to full power within established upper and lower limits. The fuel flow is adjusted by a torque motor interface valve, housed in the FMM, which responds to EEC commands.
In the event of a critical failure of the EEC, a fuel metering manual backup mode is automatcally activated.
In case of power turbine overpseed, the limiter subsystem of the EEC automatically reuces the fuel flow, indipendently of the EEC fuel flow control.

ECC LIMITATIONS

EEC is sensitive to high temperatures. When helicopter is powered on the ground (not starting the engines) with high OAT, the EEc circuit breakers must be disconnected. See RFM, section 1, limitations.

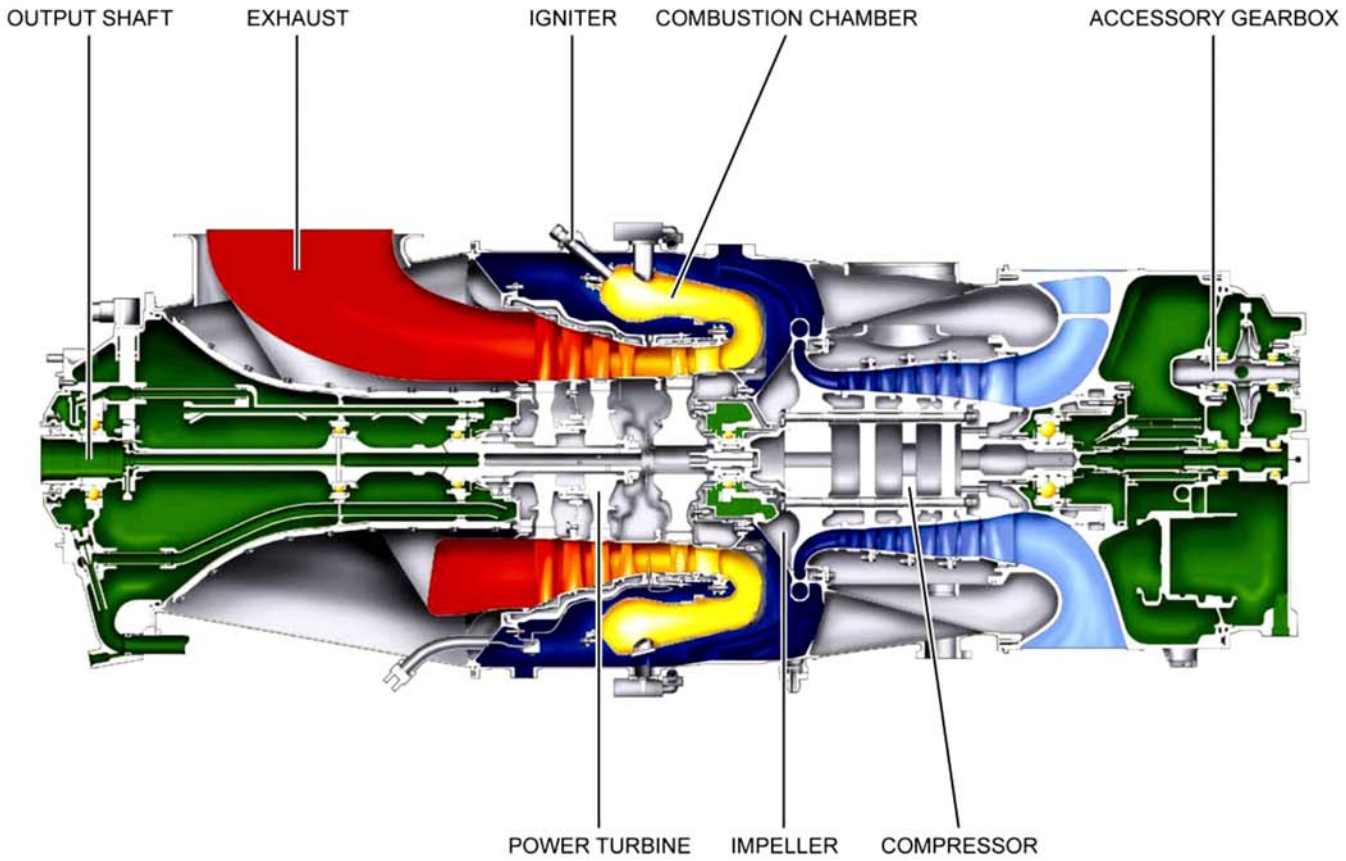


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ELECTRONIC ENGINE CONTROL INSTALLATION

ENGINE CROSS SECTION

Each engine is a free turbine turbo-shaft propulsion engine that includes a four stage axial compressor and a one stage centrifugal compressor. The compressors are driven by a single-stage compressor turbine.
The combustion chamber is a reverse flow combustion camber type where flow direction changed 180 degrees as it enters and mixes with the fuel. Metered fuel is sprayed into the combustion chamber through 14 fuel nozzles.
The power turbine turns the output shaft at the front of the engine. The output shaft supplies power to the input gears of the MGB.
A high voltage ignition unit and two spark igniters are used to start combustion. When combustion is self sustaining the ignition is turned off.



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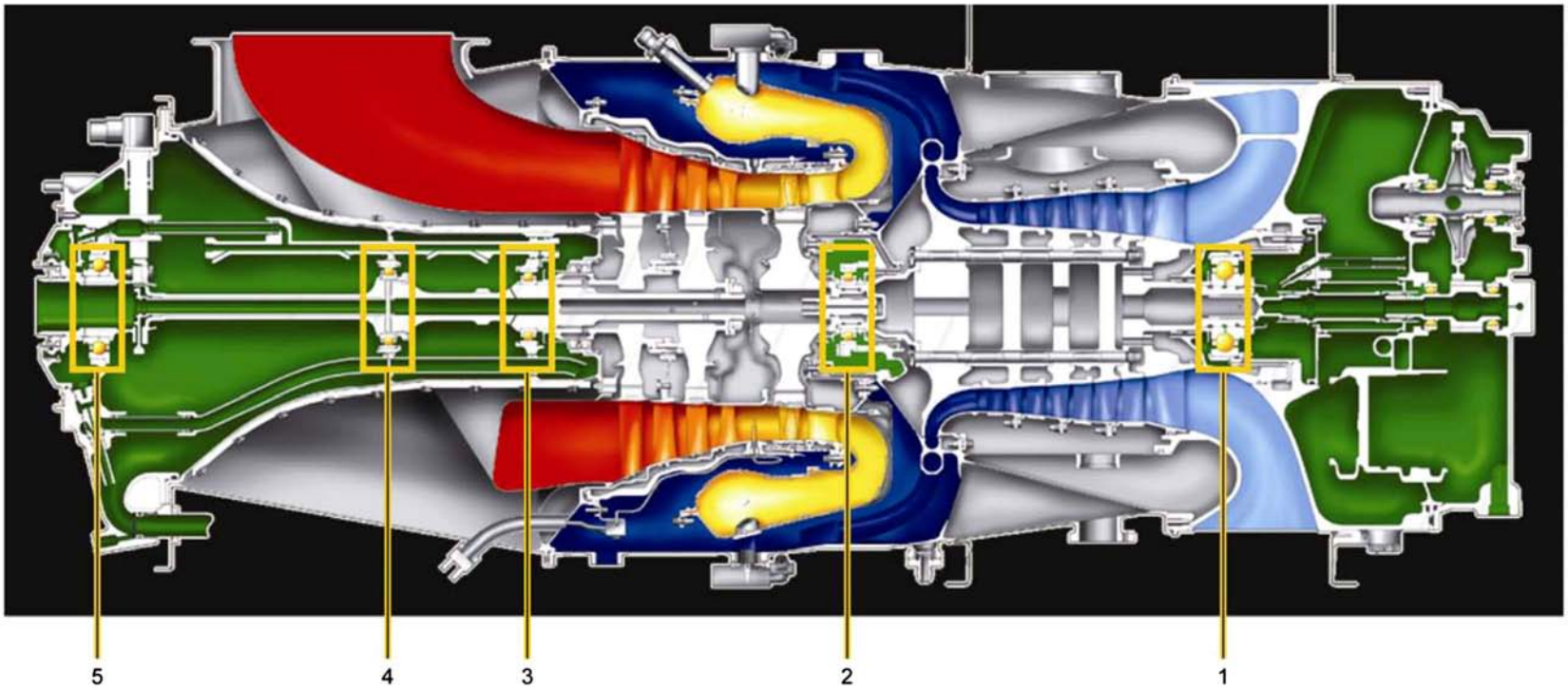
ENGINE CROSS SECTION

ENGINE CROSS SECTION – BEARINGS

The main function of the bearings is to support the major rotating assemblies.
There are 5 main bearings in the engine as follows:

- Ball bearings No. 1 & 5:
  - Absorb axial and radial loads
  - Position rotational assemblies in place
- Roller bearings No. 2, 3 & 4:
  - Absorbs radial loads only.
  - Permits axial movement caused by thermal expansion.

All bearings are pressure lubricated.



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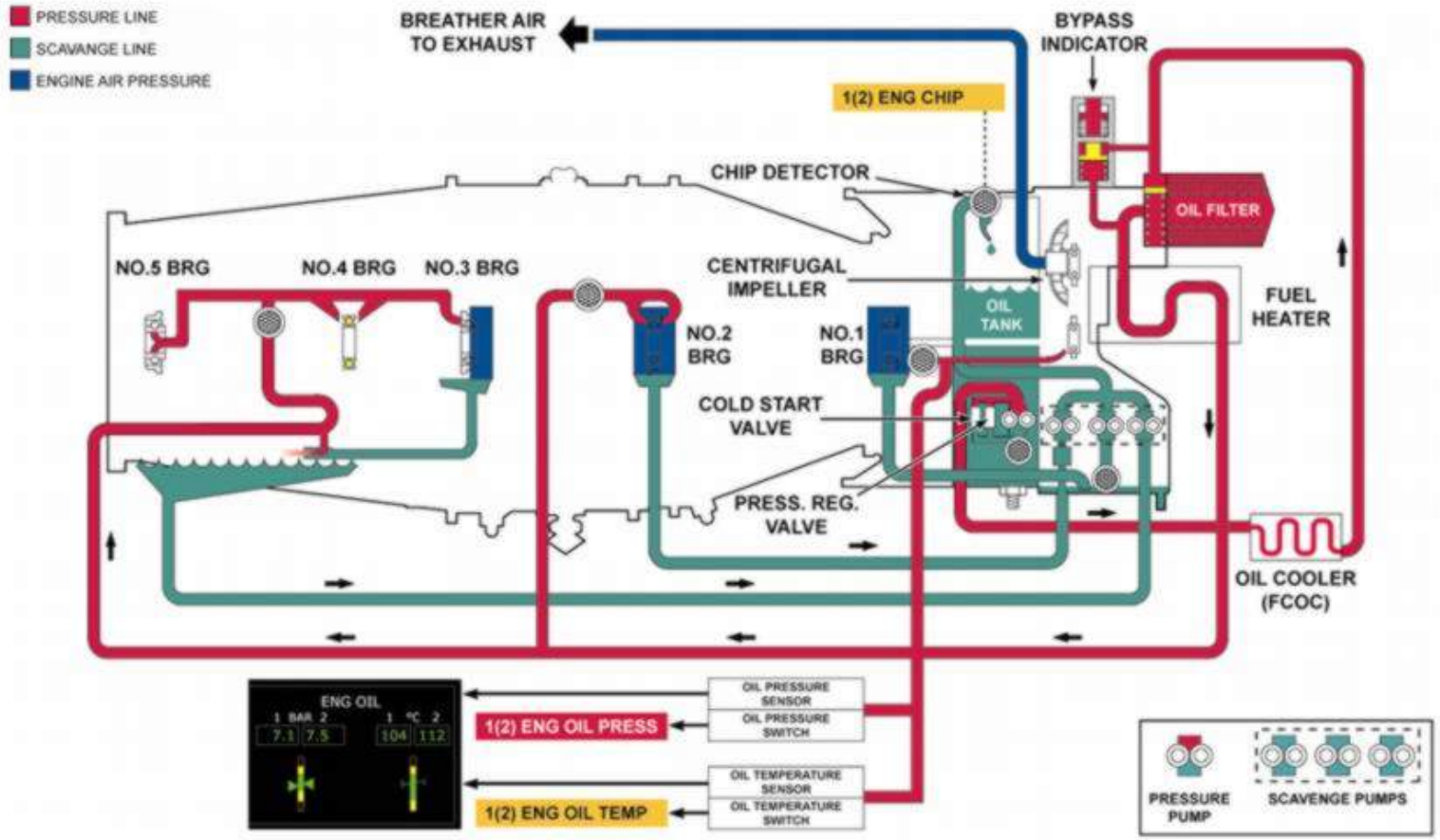
ENGINE CROSS SECTION – BEARINGS

ENGINE OIL

The engine oil system supplies a flow of filtered oil to the engine in order to cool, lubricate and clean different components.
The oil system consists of:

- integral oil tank
- pressure system
- scavenge system
- breather system.

The oil tank is integrated with the engine. It is an annular cavity created between the air inlet case and the accessory gearbox rear case.
Oil level indication is achieved with two oil level sight glasses, one per each side of the AGB (Accessory Gear Box).
Oil is presurized by a gear pump mechanically driven by the engine AGB.
A pressure-regulating valve regulates the pressure system.
Pressurizze oil is cooled by the FCOC (Fuel-Cooled Oil Cooler) and filtered by the Oil Filter which is provided with a bypass and an impending by-pass indicator (popout red button).
Leaving the Filter, engine oil flows through the Fuel Heater and then it is distributed to the pressure-lubricated bearings.
Oil pressure and temperature are sensed by two sensors that drive the indicators on the MFD and by two switches that control the 1 (2) ENG OIL PRESS warning and the 1 (2) ENG OIL TEMP caution messages in the CAS, respectively.
Engine air pressure is used to pressurize various bearing cavities for sealing. The air/oil mixture from the bearing cavities is routed to the AGB via the scavenge system.
The scavenge system return the oil to the AGB by either gravity or dedicated pumps. All returned oil flows near a magnetic chip detector located at the inlet of the oil tank.
The oil breather system separates air from oil and vents it overboard through the centrifugal impeller.



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ENGINE OIL SYSTEM – GENERAL SCHEMATIC

ENGINE FUEL AND CONTROL – GENERAL

The engine fuel and control system supplies:

- fuel to the engine at the required pressure and flow to avoid main rotor droop (isochronous governing);
- permits reducing pilot workload on engine control
- optimizes engine power and to improves engine response.

ENGINE FUEL AND CONTROL – MAIN COMPONENTS

The engine fuel and control system comprises the following main components

- Fuel Management Module (FMM)
- Fuel pumps
- Fuel filter
- Fuel heater
- Fuel Cooled Oil Cooler (FCOC)
- Ecology accumulator
- Fuel nozzles

FUEL MANAGEMENT MODULE (FMM)

The FMM is an hydro-mechanical unit that controls the fuel delivered to the engine using Electronic Engine Control (EEC) signals, Engine Control Lever (ECL) position, engine compressor discharge P3 pressure and NG speed as input parameters.

FUEL PUMPS AND FILTER

A Low Pressure pump supplies fuel at adequate pressure to the Fuel Heater and Filter element before routing it to the High Pressure pump.

The filter is provided with a bypass valve.

FUEL HEATER

The Fuel Heater uses engine lubricating oil to heat the fuel to prevent fuel filter restriction due to ice formation, thus allowing engine operation at lower outside temperature.

FUEL COOLED OIL COOLER (FCOC)

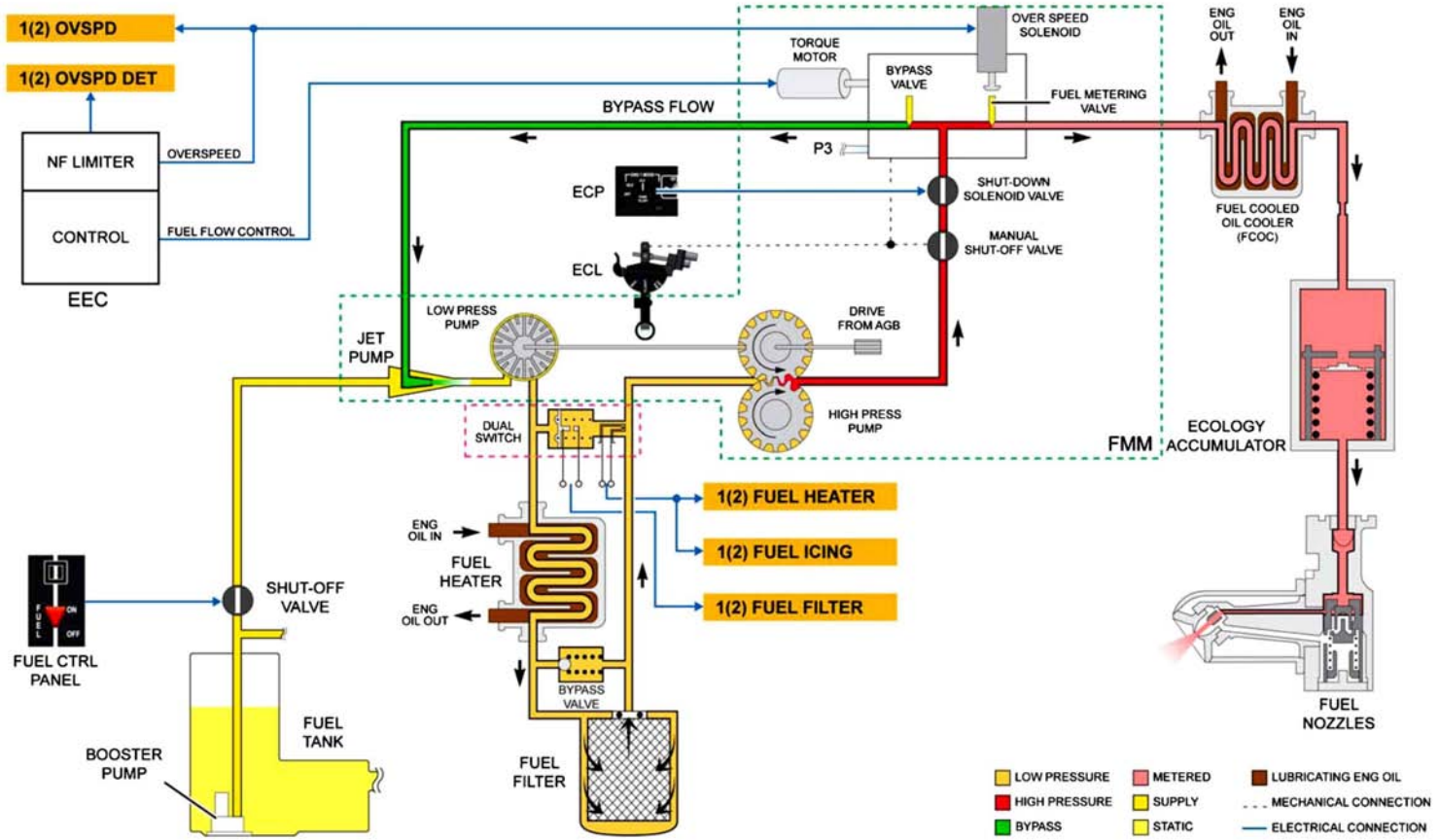
The Fuel Cooled Oil Cooler is a heater exchanger with two flow circuits: engine oil and fuel that provides for engine oil cooling with temperature regulation.

ECOLOGY ACCUMULATOR

The Ecology Accumulator prevents fuel nozzle coking after shutdown and acts as manifold reservoir to collect residual fuel from the manifold on shutdown, thus allowing reusing it for the next engine start.

FUEL NOZZLES

There are 14 fuel nozzles with integral flow divider that deliver and atomize metered fuel into the combustion chamber.



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ENGINE FUEL – GENERAL SCHEMATIC

ENGINE FUEL AND CONTROL – OPERATIONS

The FMM operates in two modes:

- AUTO mode (normal mode of operation)
- MANUAL mode (backup mode of operation)

In both situations, the fuel coming from the tank is driven by a low pressure pump through a heater and then a filter.

The Dual Switch consists of a Differential Pressure Switch and a Temperature sensor. The Differential pressure switch monitors the difference in pressure between, the Low Pressure Pump outlets and the Fuel Filter: it triggers the FUEL FILTER caution message when the Fuel Filter bypass valve is about to open (impending bypass condition).

The temperature sensor supplies the MAU with fuel temperature data to generate the FUEL HEATER and/or FUEL ICING caution messages.

After being pressurized by the high pressure pump, the pressurized fuel passes through:

- a “Manual Shut-off Valve”, directly connected to the ECL, which permits to isolate the engine from the fuel supply (Valve is closed when ECL is at OFF)
- a “Shut-Down Solenoid Valve”, directly controlled by the ENG MODE switch on the Engine control panel (ECP) is energized closed during the engine shut-down procedure. This valve must be open during manual engine start procedure, ie ENG MODE switch must be at IDLE or FLIGHT.

A Bypass Valve keeps a constant value of differential pressure across the Fuel Metering Valve.

A by-pass flow line sends the exceeding fuel back to the low pressure pump.

The metered fuel flow is sent through the FCOC to the fuel nozzles.

The FMM include a Fuel Metering Valve (FMV) which, under the action of the Electronic Engine Control (EEC) (when the engine is in automatic mode) or the ECL (when the engine is in manual mode), sets the fuel flow over the full range of the engine operations.

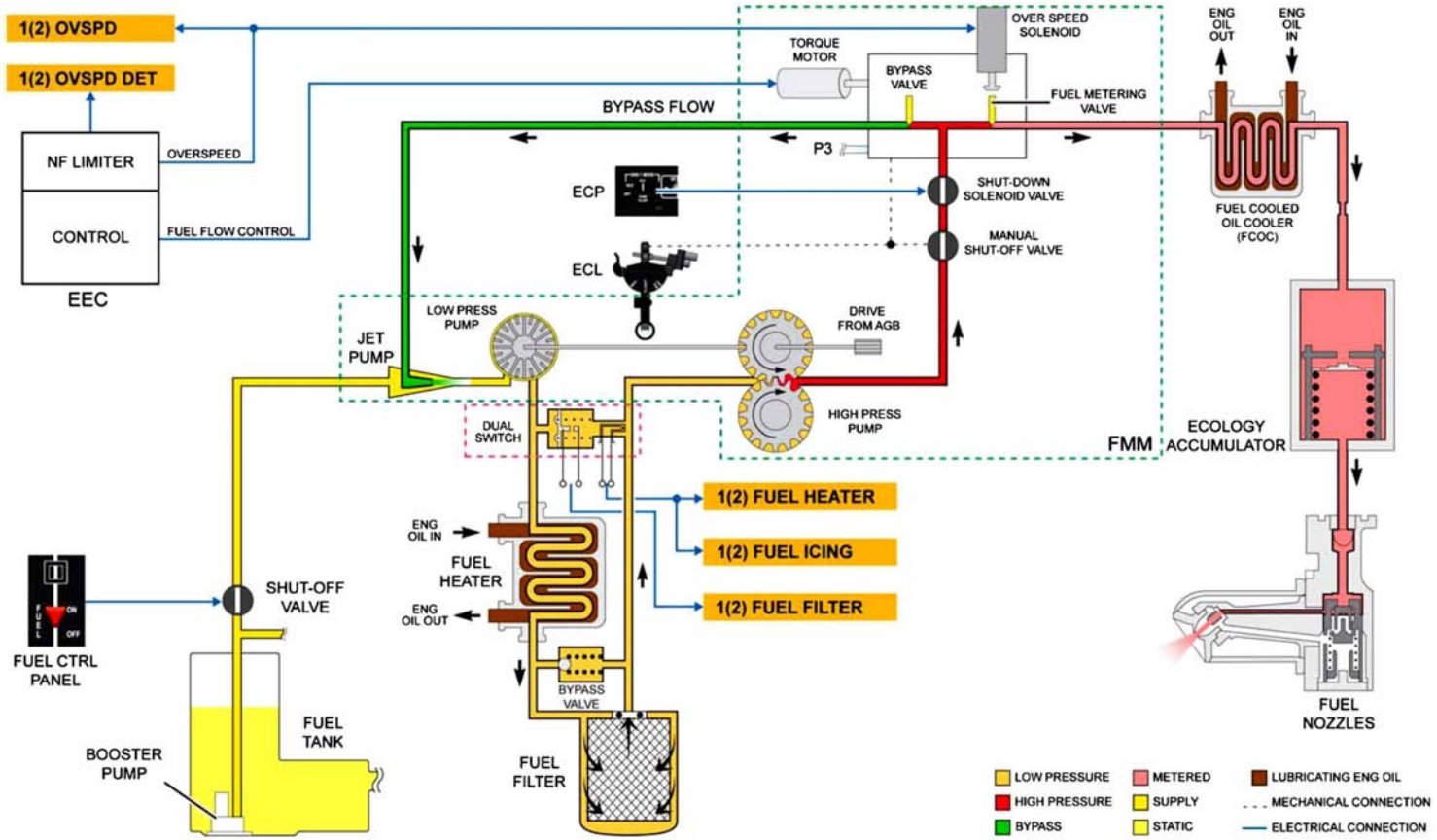
The FMM includes a Fuel Metering Valve (FMV) that is controlled by the Electronic Engine Control (EEC) or by the ECL to meter the fuel flow over the full range of the engine operation.

Depending on the FMM mode of operation (AUTO or MANUAL), the Fuel Metering Valve uses different methods to meter the fuel flow:

- in Automatic mode the EEC commands a torque motor
- in Manual mode the ECL moves mechanically.

NF LIMITER OPERATIONS

The Nf limiter is an indipendent subsystem of the EEC that receives an indipendent NF input from the TQ sensor and compares it to a threshold value of 111% NF.
In the event that an overspeed is detected, the NF limiter energizes the overspeed solenoid valve in the FMM, which reduces the fuel flow to minimum. When NF drops below 109%, the overspeed solenoid valve is deenergized and the fuel flow recovers. This cycling will occur for as long as the overspeed condition persists.
The overspeed protection is active both in AUTO and MANUAL modes.



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ENGINE FUEL – GENERAL SCHEMATIC

ENGINE – CONTROLS

1 LD SHARE switch

- TORQUE ..... Engines match in torque.
- TEMP ..... Engines match in temperature.

2 ENG 1/2 MODE switch

- OFF ..... Engine is stopped.
- IDLE ..... Engine is commanded to run at minimum speed on ground (65% NF).
- FLT ..... Engine is commanded to run at flight nominal speed (100% NF or 102% NF dependent on the RPM selector switch (#8) on pilot collective).

3 OEI switch

- Central position ..... OEI training not engaged.
- #1 (magnetic retained) ..... Engine 1 selected in OEI training mode; Engine 2 simulated in OFF mode on PDF only.
- #2 (magnetic retained) ..... Engine 2 selected in OEI training mode; Engine 1 simulated in OFF mode on PDF only.

4 START push buttons

- Pressed and released ..... Commands the relevant engine to start in MANUAL mode. If pressed again during starting in manual mode, aborts the engine start.

5 ECL

- OFF ..... The relevant engine is shut down
- MIN ..... When in MANUAL mode sets the minimum fuel flow
- FLIGHT ..... Permits full range of engine speed to be automatically adjusted by the EEC in AUTO mode
- MAX ..... When in MANUAL mode sets the maximum fuel flow

6 CHIP DETECTOR push buttons

- Pressed ..... Tests the relevant engine chip detector test (caution ENG CHIP illuminated).

7 ENG 1 (2) GOV AUTO/MANUAL selector switches

- AUTO ..... Selects the relevant engine for Automatic operation controlled by the EEC: allows the automatic starting and the automatic control of the engine.
- MANUAL ..... Selects the relevant engine for Manual operation controlled by the ECL: allows the manual starting / wet motoring / dry motoring and the manual control of the engine.

8 RPM SELECTOR switch

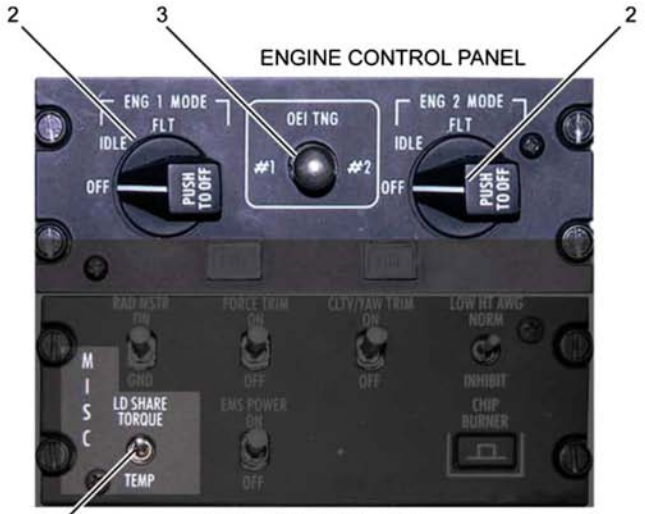
- 100% ..... Main rotor speed set at 100% RPM.
- 102% ..... Main rotor speed set at 102% RPM.

9 TQ LIM push buttons

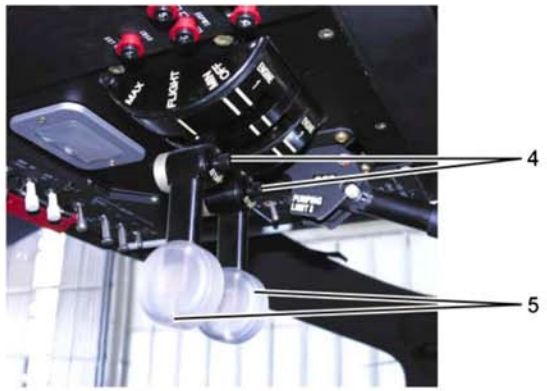
- Pressed ..... Engage/disengage torque limiting function.

10 ENGINE TRIM 1(2) SWITCH

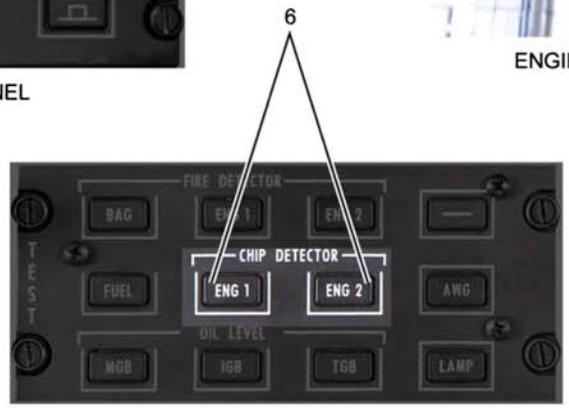
- + ... Moves the ECL towards MAX
- - ..... Moves the ECL towards MIN



MISCELLANEOUS PANEL



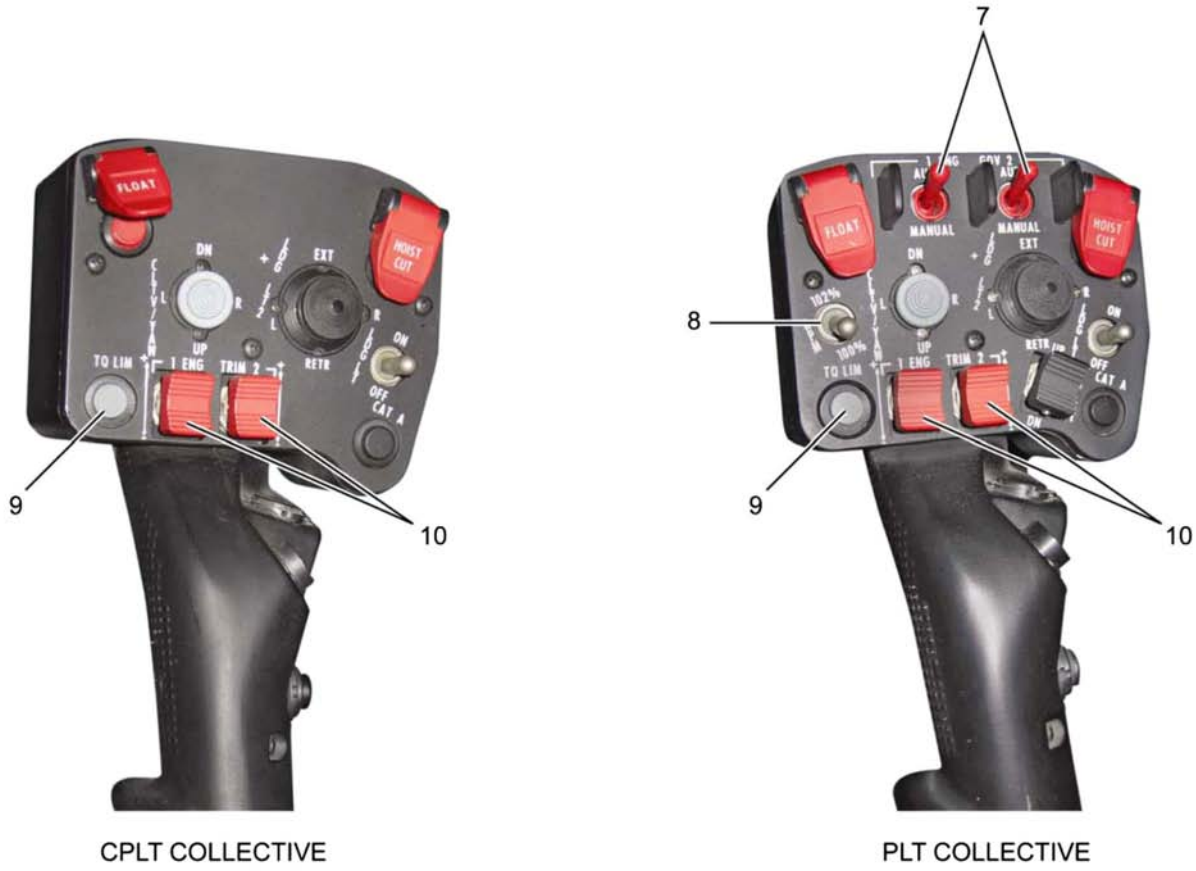
ENGINE CONTROL LEVERS



TEST CONTROL PANEL

ICN-39-A-710000-G-A0126-03181-A-01-1

ENGINE CONTROLS



CPLT COLLECTIVE

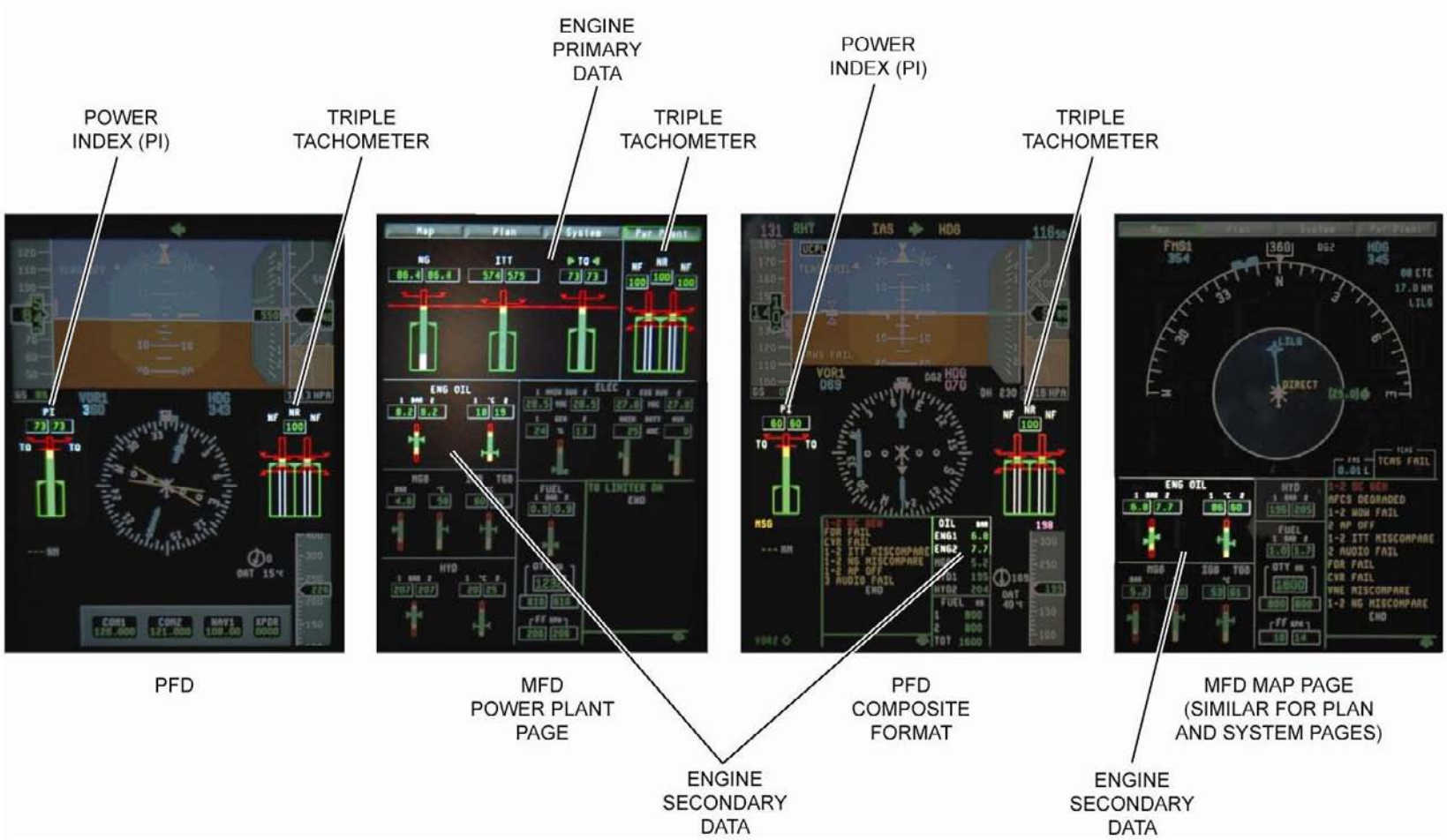
PLT COLLECTIVE

ICN-39-A-710000-G-A0126-03182-A-01-1

ENGINE CONTROLS

ENGINE INDICATION – GENERAL

The primary and secondary engine parameters indicators and a triple tachometers are displayed on the MFD, Power Plant page. Other engine parameters, Power Index and the Triple Tachometer indicators, are displayed on the PFD.
The NG, ITT, TQ and PI indicators have one single vertical scale with one analogue pointer and one digital readout per engine.
The Triple Tachometers combines NF and NR indications on a dual vertical scale with one analog pointer and one digital readout for each engine and for NR: the Triple Tachometer on the PFD does not have digital readouts for NF.
Secondary engine parameters (oil pressure and oil temperature) are displayed with a vertical scale with one analogue pointer and digital readout per engine.
In case MAP, PLAN or a synoptic page system is selected, primary engine parameters are removed and secondary engine parameters are displayed in the bottom half of the MFD.
In composite format only digital readouts of engine oil pressures are provided.
Primary engine data indicators (NG, ITT, TQ), Triple Tachometer and PI indicator automatically change their scale ranges in accordance with the power plant operating status: AEO, OEI, Power off (Autorotation).



ICN-39-A-710000-G-A0126-03183-A-01-1

ENGINE INDICATION – GENERAL

ENGINE – INDICATIONS

1 NG INDICATOR

- ..... Provides digital readout and analog indicator for engines 1 and 2 gas generator turbine speed, in percent of rpm.

2 ITT indicator

- ..... Provides digital readout and analog indicator for engines 1 and 2 inter turbine temperature, in °C or percentage.

3 TQ indicator

- ..... Provides digital readout and analog indicator for engines 1 and 2 torque, in percentage.

4 NF indicator

- ..... Provides digital readout and analog indicator for engines 1 and 2 free turbine rpm, in percent value.

5 NR indicator

- ..... Provides digital readout and analog indicator for main rotor speed, in percent of rpm.

6 NG Maximum transient

- ..... Marks the maximum transient value for NG (time limit = 5 seconds)

7 ITT

- ..... ITT maximum transient for 5 seconds.

8 TQ

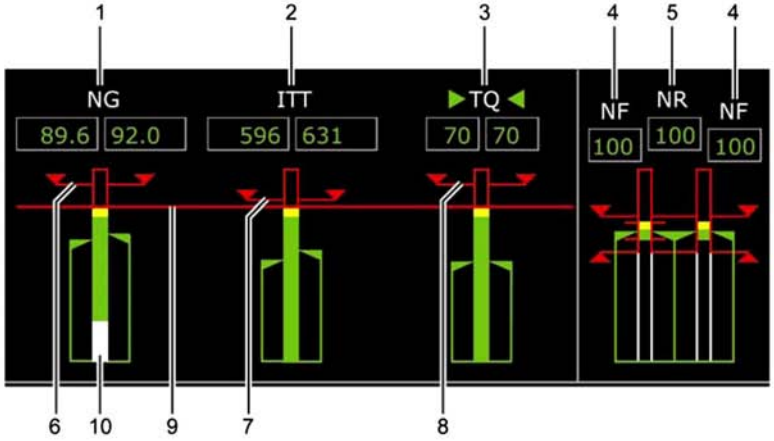
- ..... TQ maximum transient for 5 seconds.

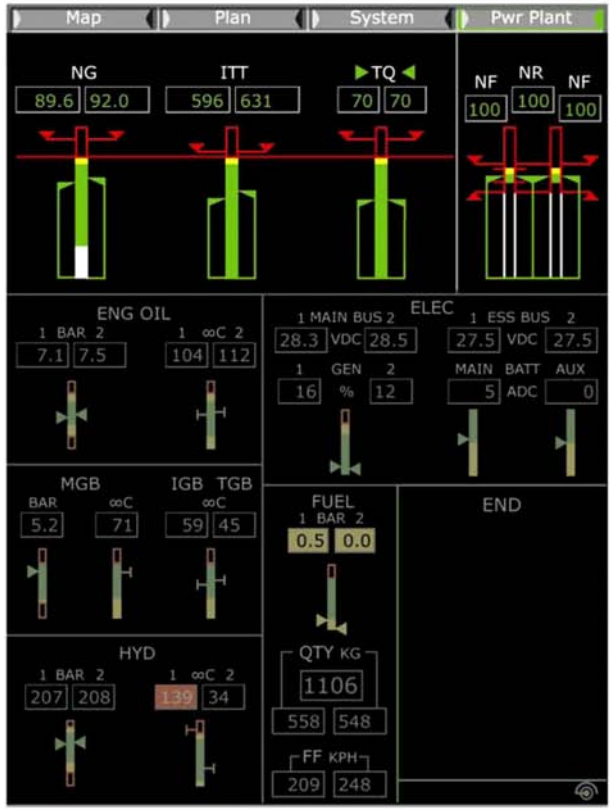
9 Maximum take off

- ..... Marks the maximum take off value for NG, ITT, TQ.

10 White range

- ..... Identifies engine acceleration/ deceleration range for NG.





MFD

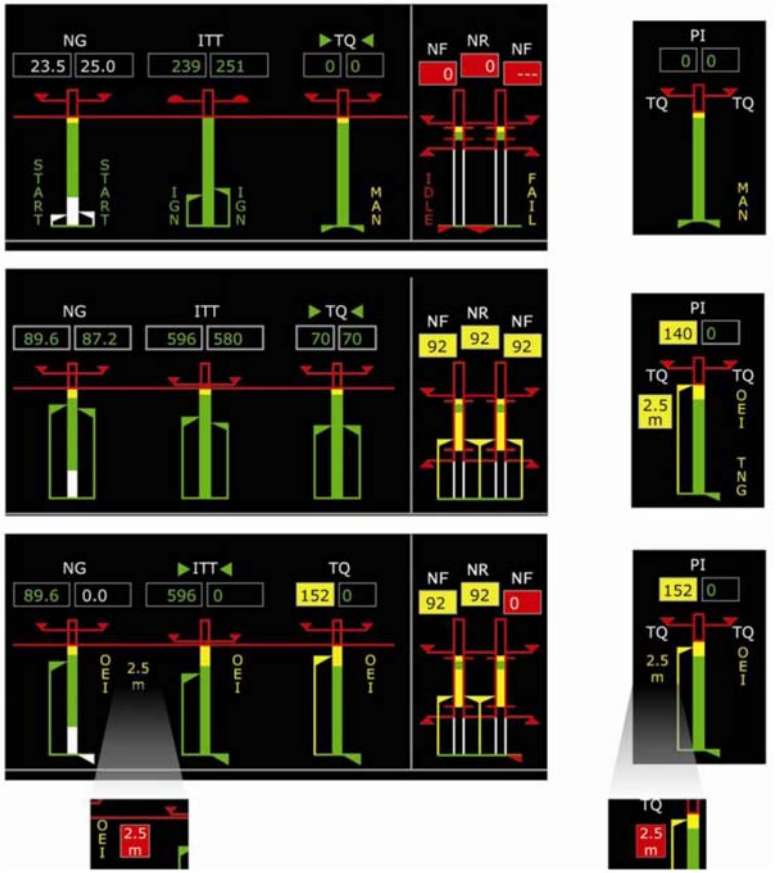
ICN-39-A-710000-G-A0126-03185-A-01-1

ENGINE INDICATION – PRIMARY DATA

The following legend related to engine operation can be displayed on the MFD Power Plant page and on the PFD:

FAIL FAIL	• Displayed on the left/right side of Triple Tachometer; indicates TQ/NF sensor failure.
OEI	• Displayed on the left/right side of PI, NG, ITT and TQ indicators in case of OEI conditions.
MAN	• Displayed on the left/right side of TQ and PI indicators indicates engine is in manual mode
OEI TNG	• Displayed on the left/right side PI indicators indicates engine is OEI training mode (simulator only)
2.5 m	• Displayed in steady amber on the left / right side of PI indicator and between the NG and ITT indicator when in OEI conditions the operationg engine is in the cautionary range (2.5 minutes time limit)
2.5 m	• Displayed in blinking red inverse video when the 2.5 m time is within 10 sec from expiration • Displayed is steady red inverse video when the 2.5 m time limit has expired; the 1(2) ENG LIM EXPIRE caution message is also displayed

	• Displayed on the left/right side of NG indicator during starting phase (0 to 49%)
	• Displayed on the left/right side of ITT indicator during starting phase (0 to 49% NG) to confirm ignition system is powered.
	• Displayed aside the legend of TQ or ITT indicators according to the load sharing switch position
  	• Displayed on the left/right side of the triple tachometer; indicates the ENG MODE switch is set to IDLE. • The IDLE legend matches the colour of corresponding NF pointer.



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ENGINE INDICATION – LEGENDS

ENGINE – POWER INDEX INDICATOR

1 PI Indicator

- ..... Provides digital readout and analog indicator in percent of torque equivalent.

2 PI maximum transient

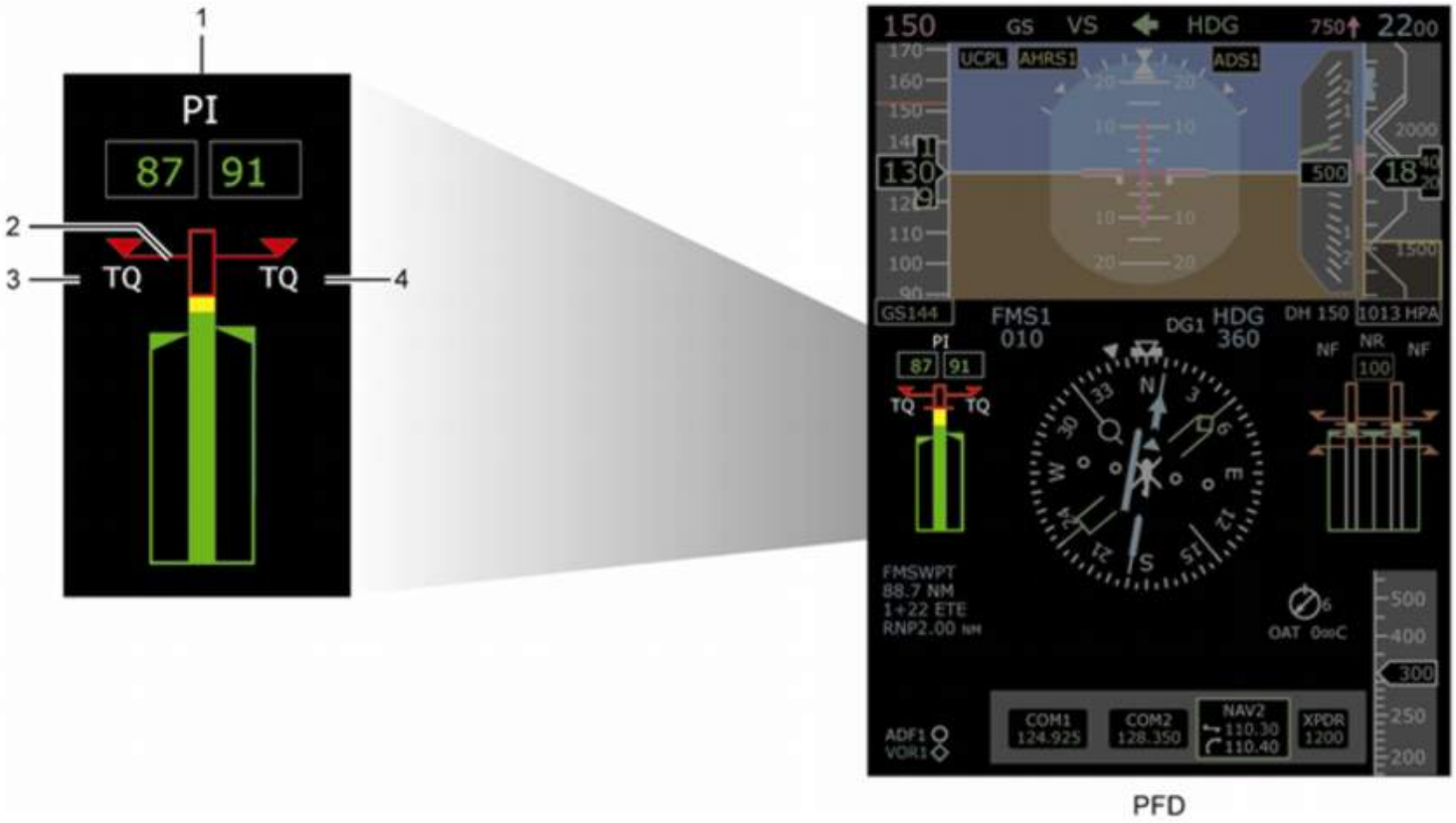
- ..... Marks the maximum transient5 value for PI (time limit 5 seconds)

3 Engine 1 limiting parameters legend

- ..... Engine 1 current limiting parameter legend (NG; ITT; TQ as applicable).

4 Engine 2 limiting parameters legend

- ..... Engine 2 current limiting parameter legend (NG; ITT; TQ as applicable).



PFD

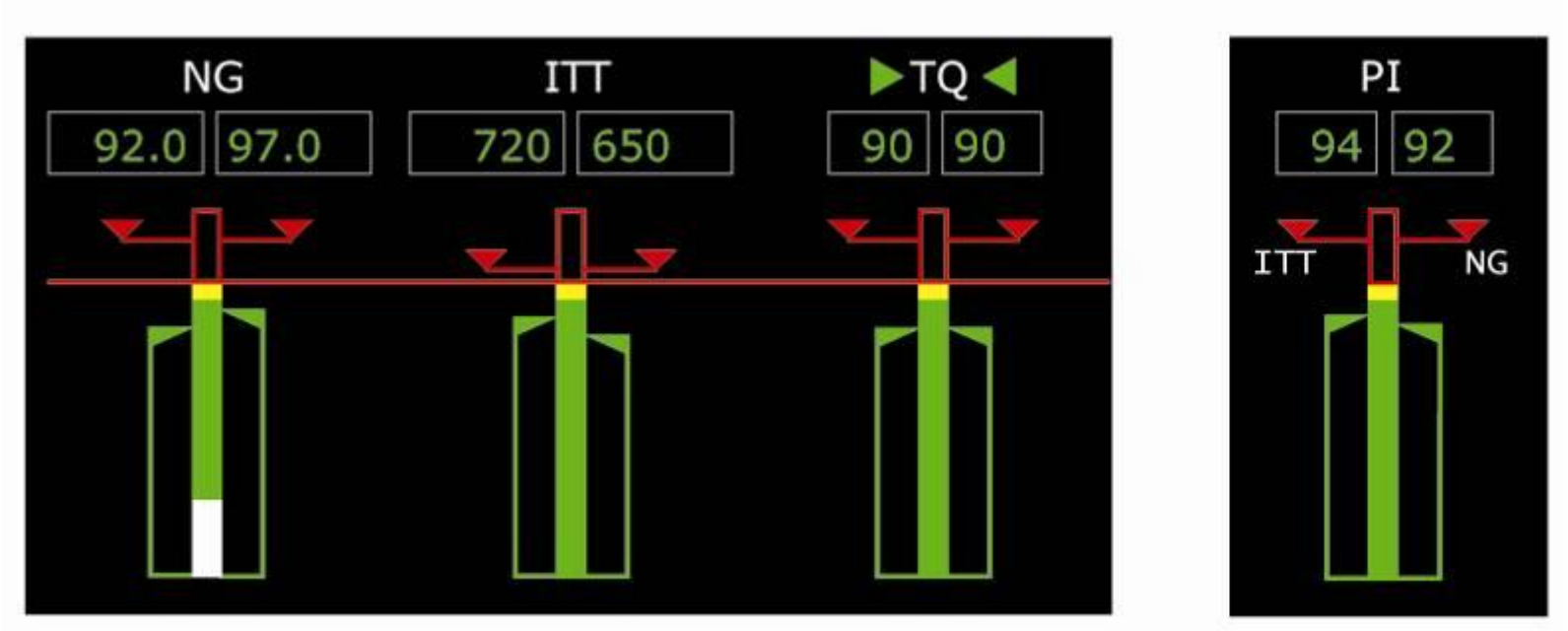
ICN-39-A-710000-G-A0126-03188-A-01-1

POWER INDEX

POWER INDEX INDICATOR (CONT'D)

The Power Index indicator (located on PFD) is a single scale instrument that provides the pilot with an overall indication of engine performance by displaying a composite of the three primary engine parameters (Ng, ITT, Tq) over a scale that is the same as the TQ indicator in Pwr Plant page of the MFD.
PI is an expression of “equivalent torque”; to achieve this, Ng and ITT values are re-scaled to percent of torque.
PI displays the power available at any instant and the parameter that is closest to reaching its operational limit.
The limiting parameter is forced to TQ when PI value is between 0 and 10%.
The PI indicator is automatically rescaled according to the power plant status: AEO, OEI, Power off. PI scale always reflects the TQ indicator scale when TQ is the limiting parameters.
The actual parameter values are always available in the Power Plant page on MFD.
PI indication for each engine is computed independently from each other.
The following figure shows a possible scenario as an example:

- the legend ITT is displayed on the left side of the PI scale meaning that engine 1 is closest to reaching the temperature limit;
- the legend NG is displayed on the right side of the PI scale meaning that engine 2 is closest to reaching the NG limit;
- primary engine data on MFD show the actual values, in particular actual torque is 90% for both engines.



ICN-39-A-710000-G-A0126-03259-A-01-1

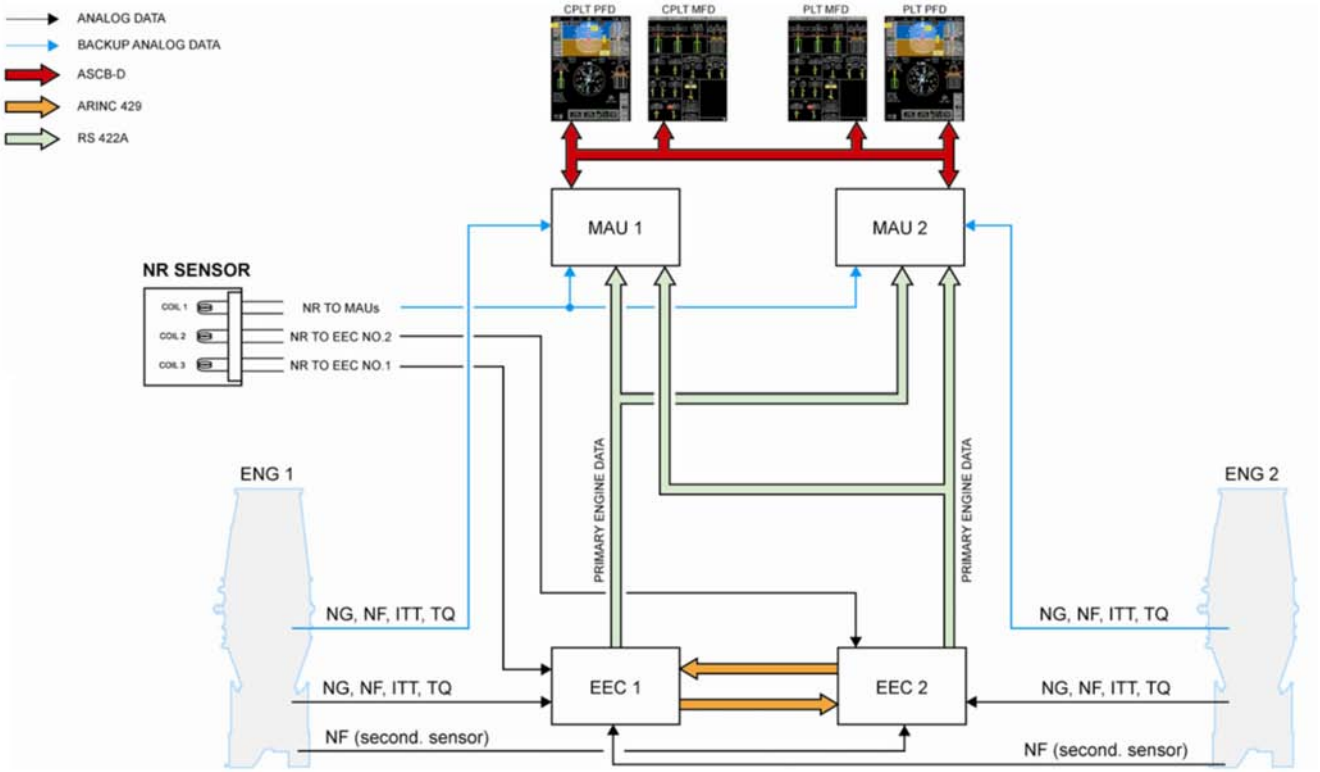
EXAMPLE OF POWER INDEX

PRIMARY DATA / BACKUP DATA INDICATIONS

The ITT thermocouples of each engine are directly connected to the on-side EEC and to the on-side MAU.
The NG and TQ/NF sensors of each engine have two independent coil pick-ups each and provide two independent analog output signals, one connected to the on-side EEC, the other to the on-side MAU. The ù NR sensor has similar interface (see chapter 62-00).
Each EEC outputs the primary engine data (NG, ITT, TQ, NF) and NR on a digital bus to both MAUs.
The digital data available from EEC are selected by default for display on the MFD (primary data source); in case any of the digital primary engine data is not available from the EEC, the corresponding analog input from the sensor to the MAU is automatically selected for display (backup data source). If the primary engine data digital bus communication is lost, the 1(2) EEC DATA caution message is displayed in the CAS.
The backup data can also be manually selected for display by checking the Analog check box in the Pwr Plant page menu.
When backup data source is selected for display, PI indication is not computed: PI pointer is removed and digital readout shows amber dashes.

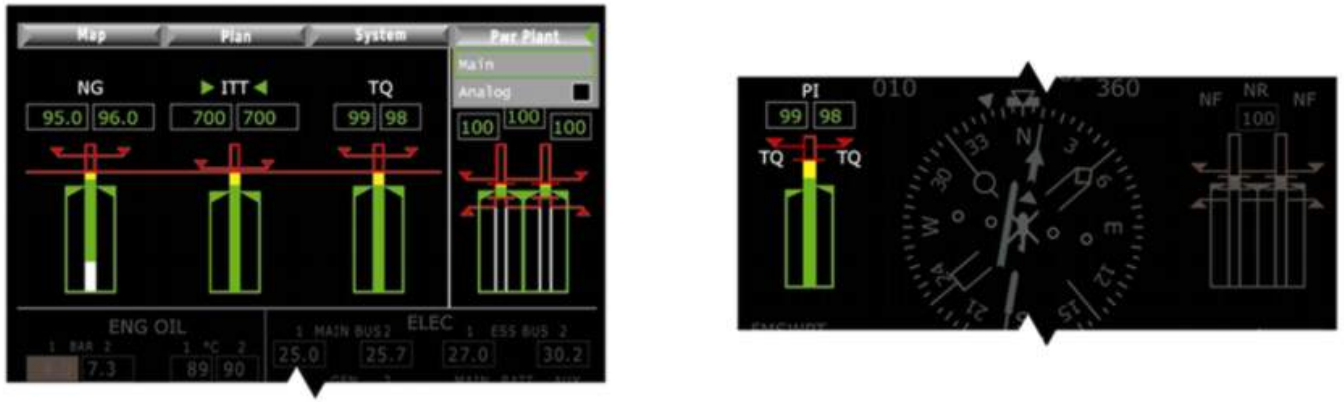
MAUS COMPARISON FUNCTION

In case of miscompare between analogue data and digital data, the caution message will be generated in the CAS window.
In case of failure of an analogue parameter, the caution message ENG ANALOG FAILURE will be generated in the CAS window.
On Power Plant page, digital data are displayed in green. When ANALOG is selected, the analogue read-out numbers change the colour in white and the displayed data (backup data) comes directly from the relative sensors.
In case of primary date is declared invalid by the EEC, the MAUs use the back-up sensor for display, monitor warning, etc., purpose. The analogue read-out numbers value is shown in “white”.



ICN-39-A-710000-G-A0126-03189-A-01-1

ENGINE DATA - GENERAL SCHEMATIC



ENGINE PRIMARY DATA SELECTED



BACKUP ENGINE DATA SELECTED (ANALOG)

ICN-39-A-710000-G-A0126-03190-A-01-1

ENGINE PRIMARY AND BACKUP ANALOG DATA INDICATIONS

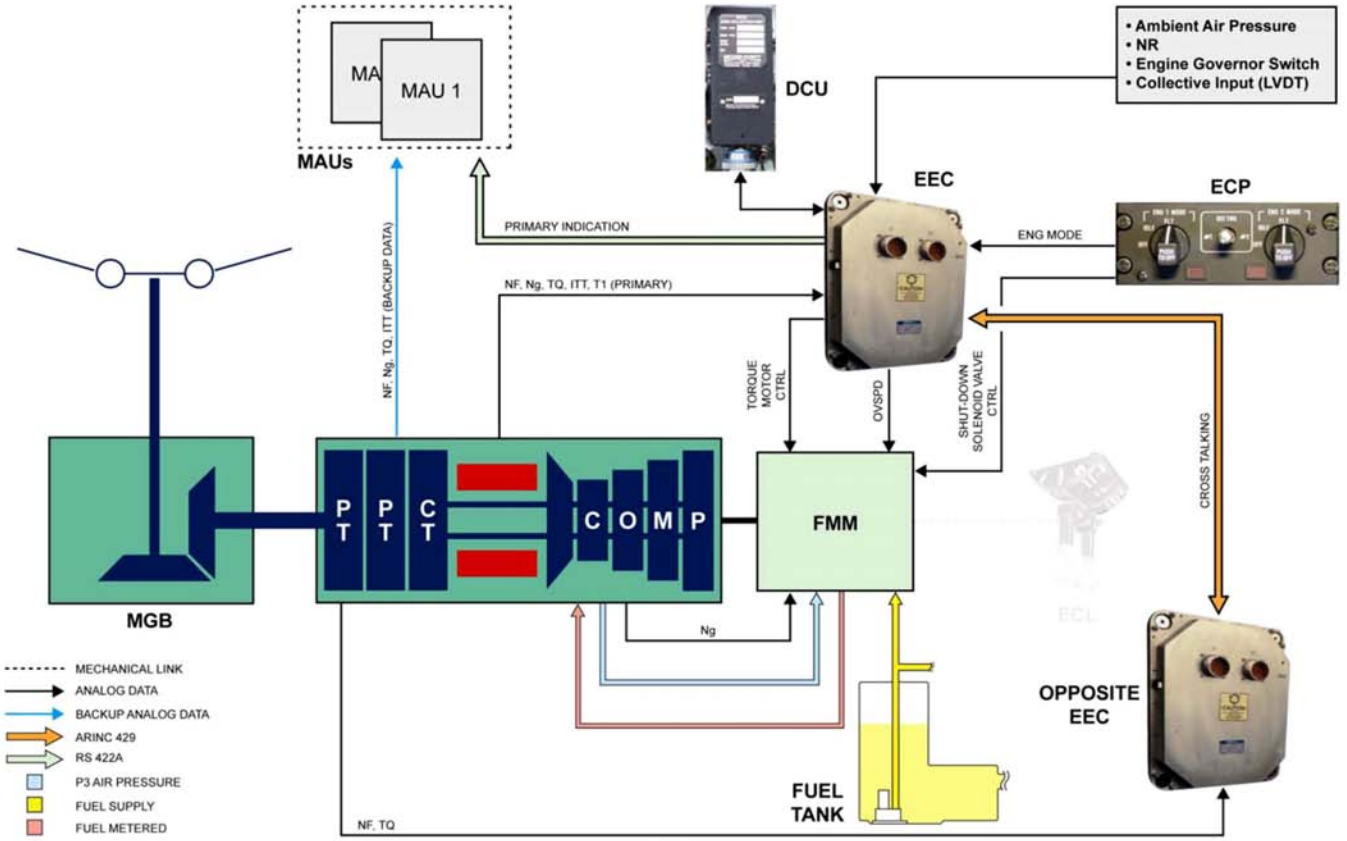
ENGINE-SYSTEM OPERATIONS

AUTOMATIC MODE

The normal automatic mode (AUTO) is the primary mode of operation achieved with the ENG MODE switch on FLIGHT position and ENG GOV switch on AUTO.
In automatic mode the EEC:

- provides automatic controls of the full operating envelope of the engine without exceeding limiting parameters
- computes the desired fuel flow and through a torque motor sets the position of the metering valve in the FMM to achieve a new NG reference point in accordance with variation of pressure altitude, OAT and collective position.

Engine starting and stopping is also automatically performed by acting on the ENG MODE switch on the Engine Control Panel.
In AUTO mode the two EECs intercommunicate via a cross talking digital bus so that they can match the engine setting either in ITT or torque (see Engine Load share).

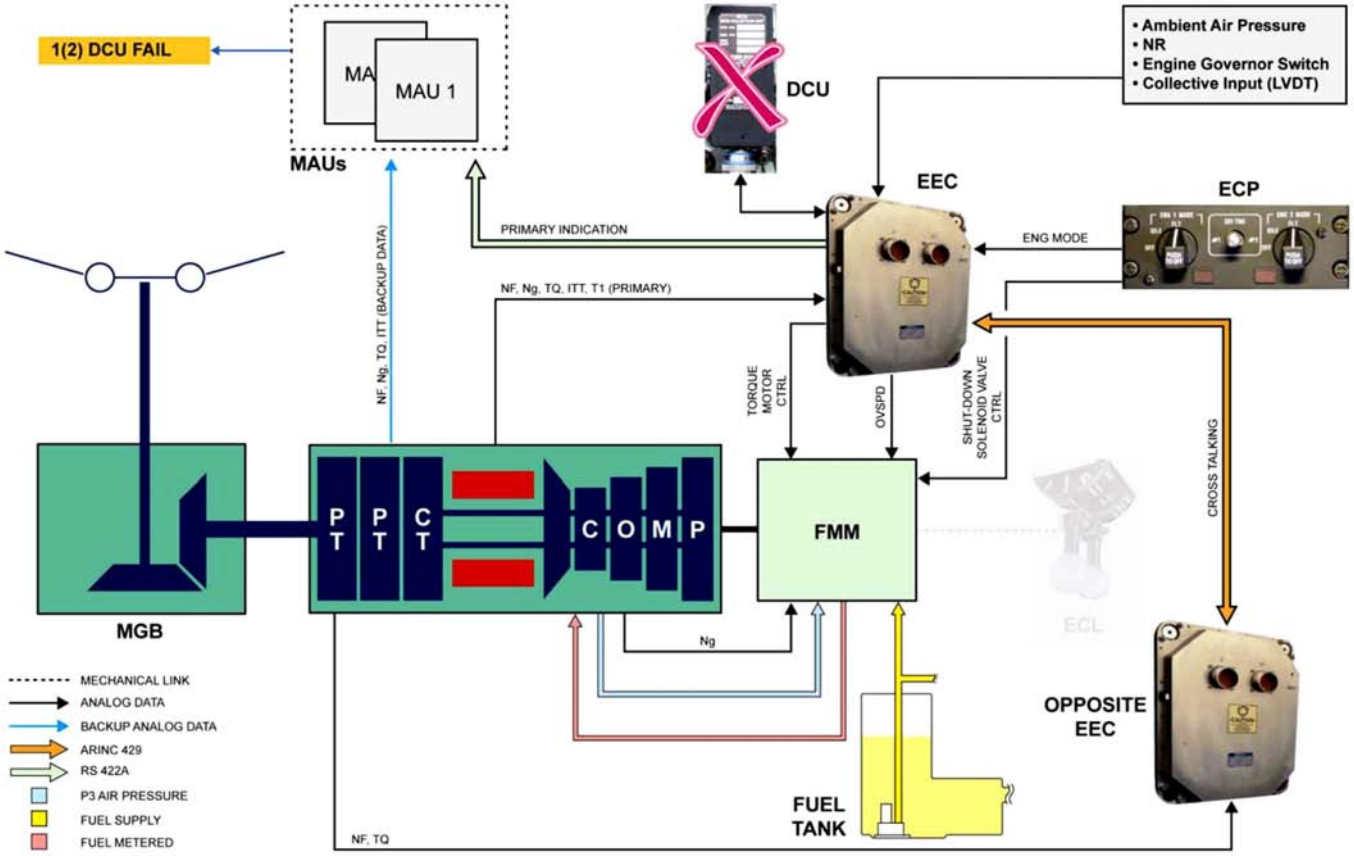


ICN-39-A-710000-G-A0126-03192-A-01-1

ENGINE SYSTEM OPERATION – AUTOMATIC MODE

AUTOMATIC MODE – DCU FAILURE

The 1(2) DCU caution message indicates that the link between EEC and DCU is lost and the engine control system is operating with a non-critical system fault which may result in degraded engine operation.
Full rotor governing is mantained since engine is still opeting in AUTO mode.

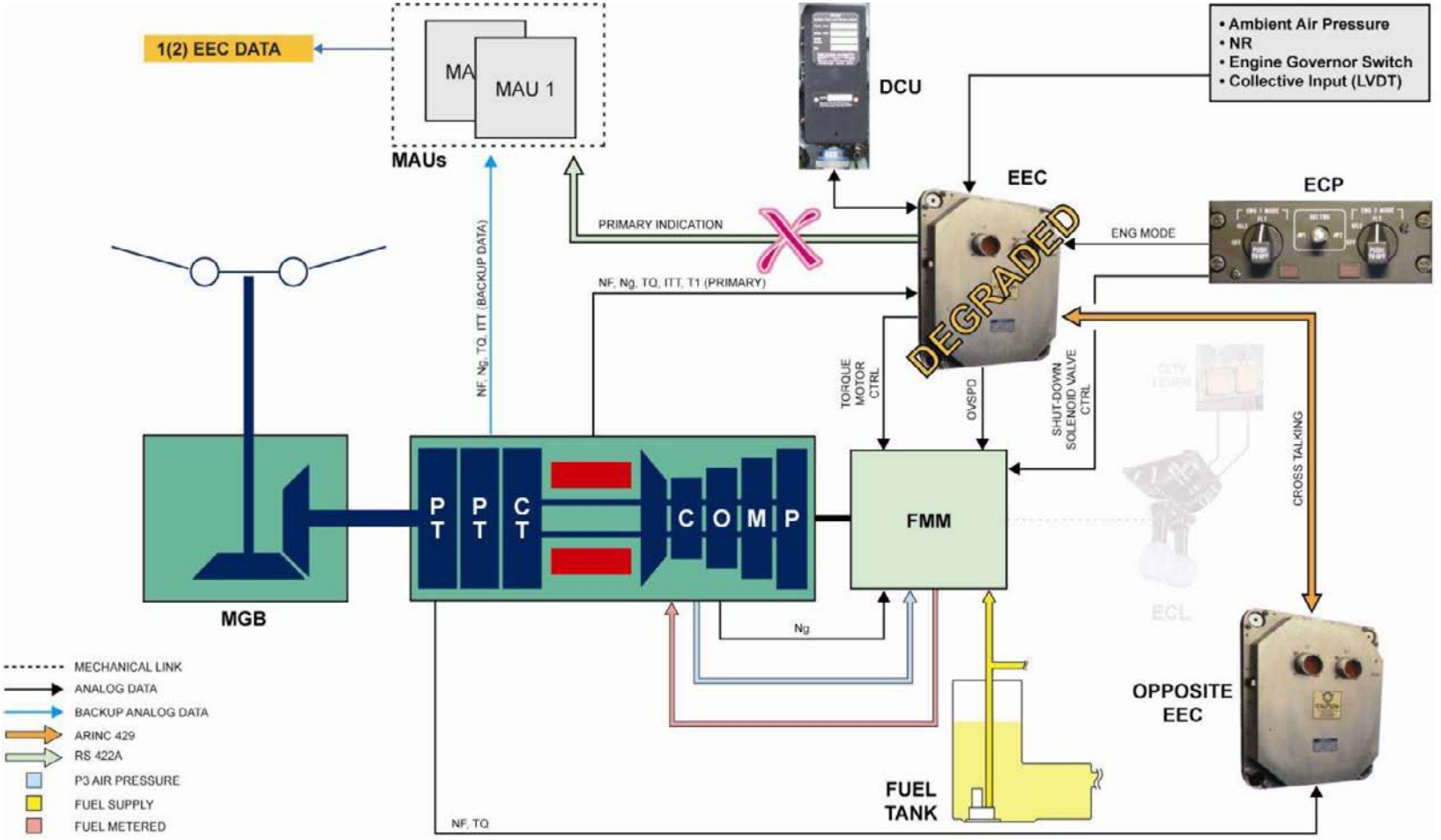


ICN-39-A-710000-G-A0126-03193-A-01-1

ENGINE SYSTEM OPERATION – AUTOMATIC MODE

AUTOMATIC MODE – EEC DEGRADED

The 1(2) EEC DATA caution message indicates that the digital link between EEC and MAU is lost and the engine control system is operating with a non-critical system fault which may result in degraded engine operation.
Full rotor governing is mantained since engine is still operating in AUTO mode.
In this case the analog backup data are automatically selected for display.

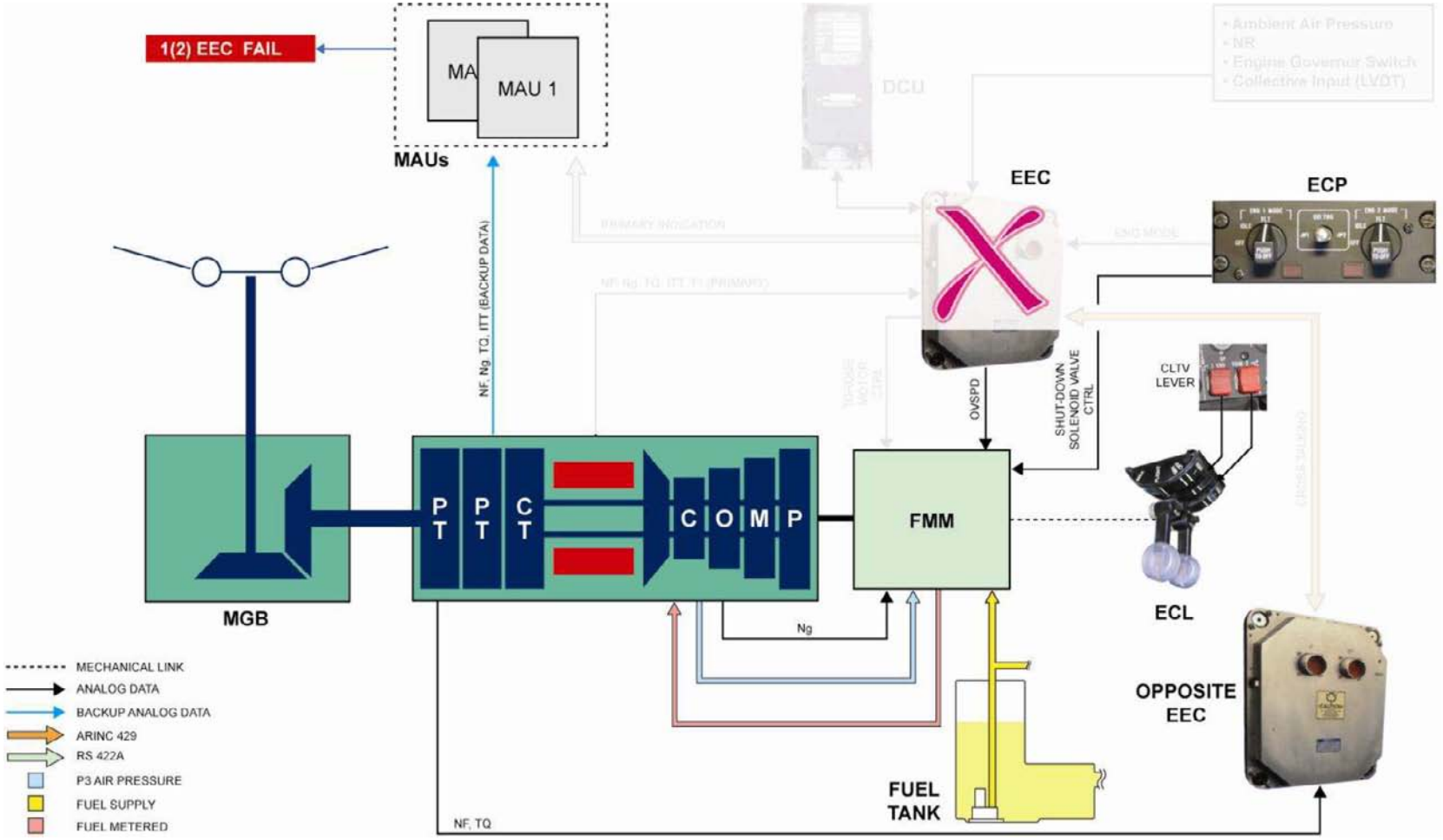


ICN-39-A-710000-G-A0126-03194-A-01-1

ENGINE SYSTEM OPERATION – AUTOMATIC MODE

MANUAL MODE - ECC FAIL

In case of a critical failure of the EEC, the 1(2) EEC FAIL warning message is displayed in the CAS (associated with aural warning message WARNING – WARNING).
The engine control system automatically enters into MANUAL mode: the FMM torque motor is disables and fuel flow can only be adjusted by moving the ECL.
ECL position can be remotely controlled by acting on the ENG TRIM switches on the collective lever.
ECL can also be manually moved by grasping it.
Upon changeover from AUTO to MANUAL mode with engine in steady state (power demand / NG = constant), no change in fuel flow is experienced (NG remains constant).
When in MANUAL mode NF limiter is still active and the function of the ENG MODE selector on the Engine Control Panel (ECP) is limited to the control of the shut-down solenoid valve (energized closed when the selector is in OFF position).
Pilot can voluntary select MANUAL mode by acting on the ENG GOV switches on the collective lever (pilot side only).

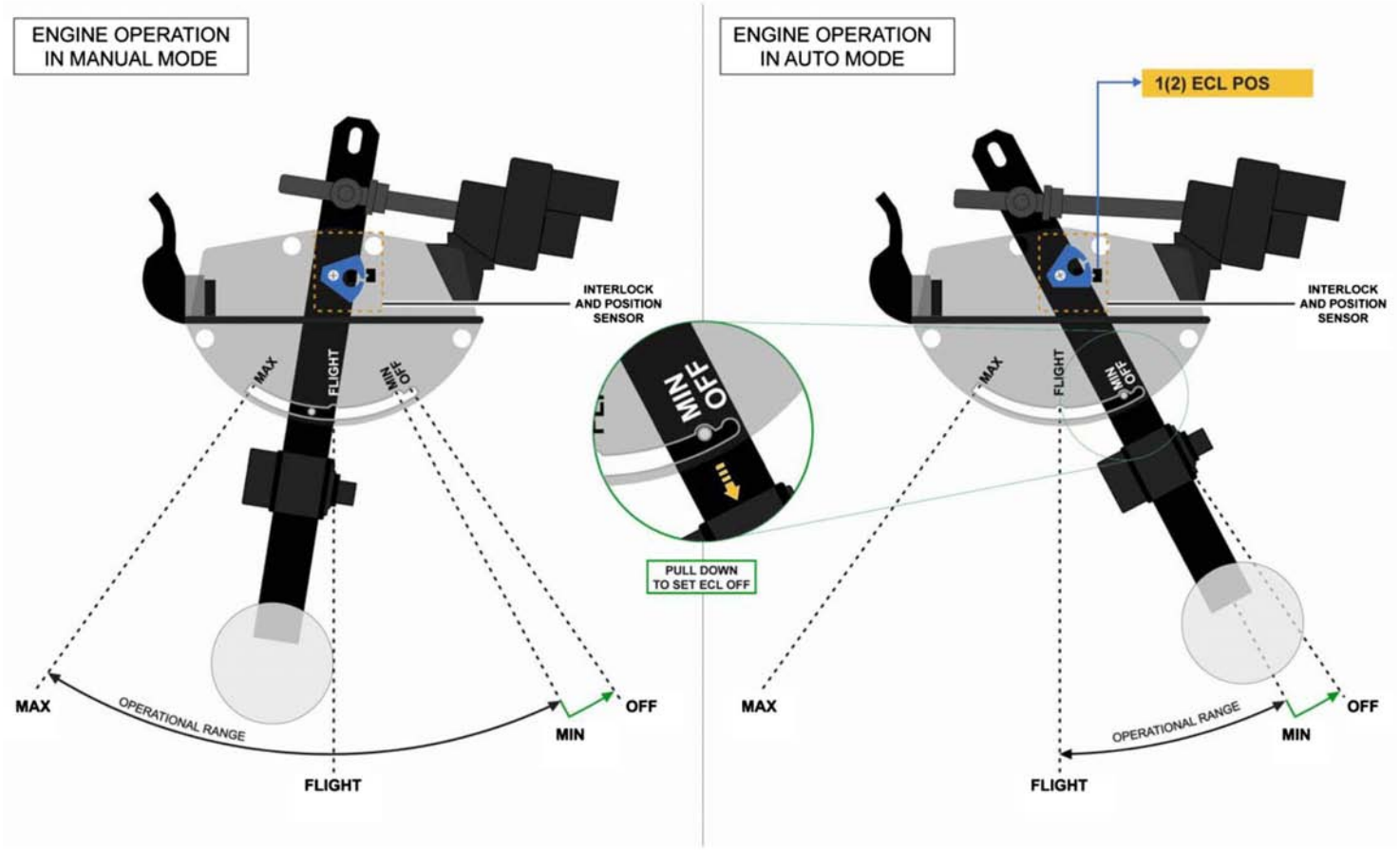


ICN-39-A-710000-G-A0126-03262-A-01-1

ENGINE SYSTEM OPERATION - MANUAL MODE

ECL – OPERATION

Each Engine Control Lever (ECL) is mechanically connected to the FMM via teleflex cables.
Whenever the ECL is at OFF, the manual shut-off valve in the FMM is closed and the engine, if running, shuts down. A mechanically detent at MIN position prevents accidental engine shutdown when retarding the ECL: to set the ECL to OFF position it must be pulled down and moved backwards.
The ECL is required to be in FLIGHT position for normal engine operation (AUTO mode). While engine is in AUTO mode, a position sensor inside the ECL controls the activation of the 1(2) ECL POS caution message in the CAS if the ECL is not at FLIGHT; the same device provides an interlock to prevent moving the ECL beyond FLIGHT position.
When the engine is operated in MANUAL mode, engine power can only be adjusted by moving the ECL: the interlock circuit is disabled (ECL can be moved through its full range from MIN to MAX) and the 1(2) ECL POS caution message is suppressed.
During manual operation, either for training or in an emergency, the pilot has full authority to increase or decrease the engine power by moving the ECL.
The amount of travel required for a certain amount of power, change is proportional and linear in the complete range of ECL.



ICN-39-A-710000-G-A0126-03195-A-01-1

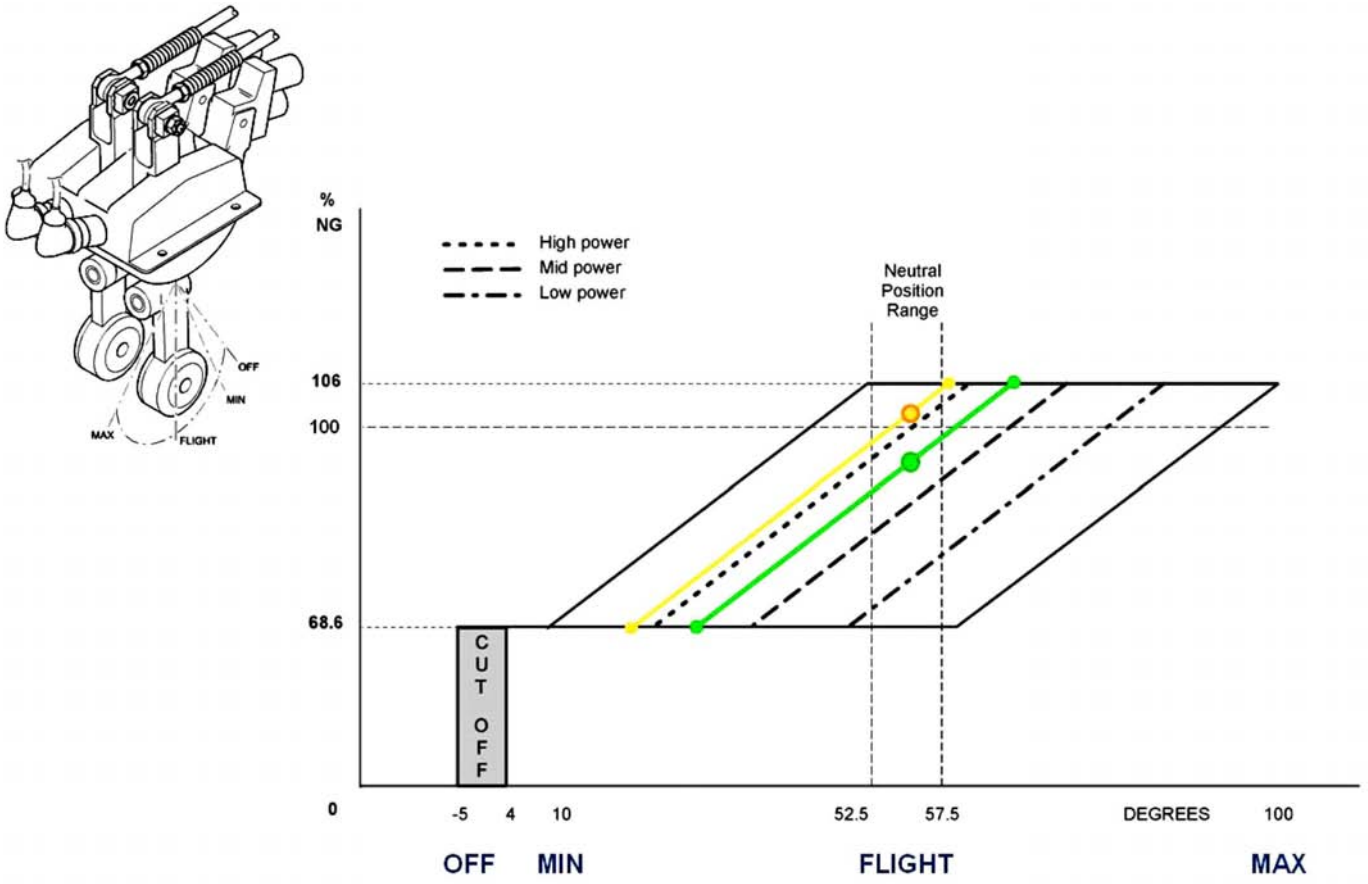
ECL – OPERATION

ECL - OPERATION (CONT'D)

While the engine is operating in AUTO mode and the ECL is in FLIGHT position, NG is adjusted by the EEC as a function of the power demand.
When the engine is switched to MANUAL mode, either because of EEC failure or by pilot selection through the ENG GOV switch and the engine is in steady state (NG = constant), no change in fuel flow is experienced.
The full range of NG values can then be achieved by moving the ECL:

- between MIN and FLIGHT if engine was operating at maximum power, or
- between FLIGHT and MAX if engine was operating at minimum power, or
- any intermediate range for any other initial power setting.

The amount of travel required for a certain amount of power change is the same in the complete range of ECL travel.

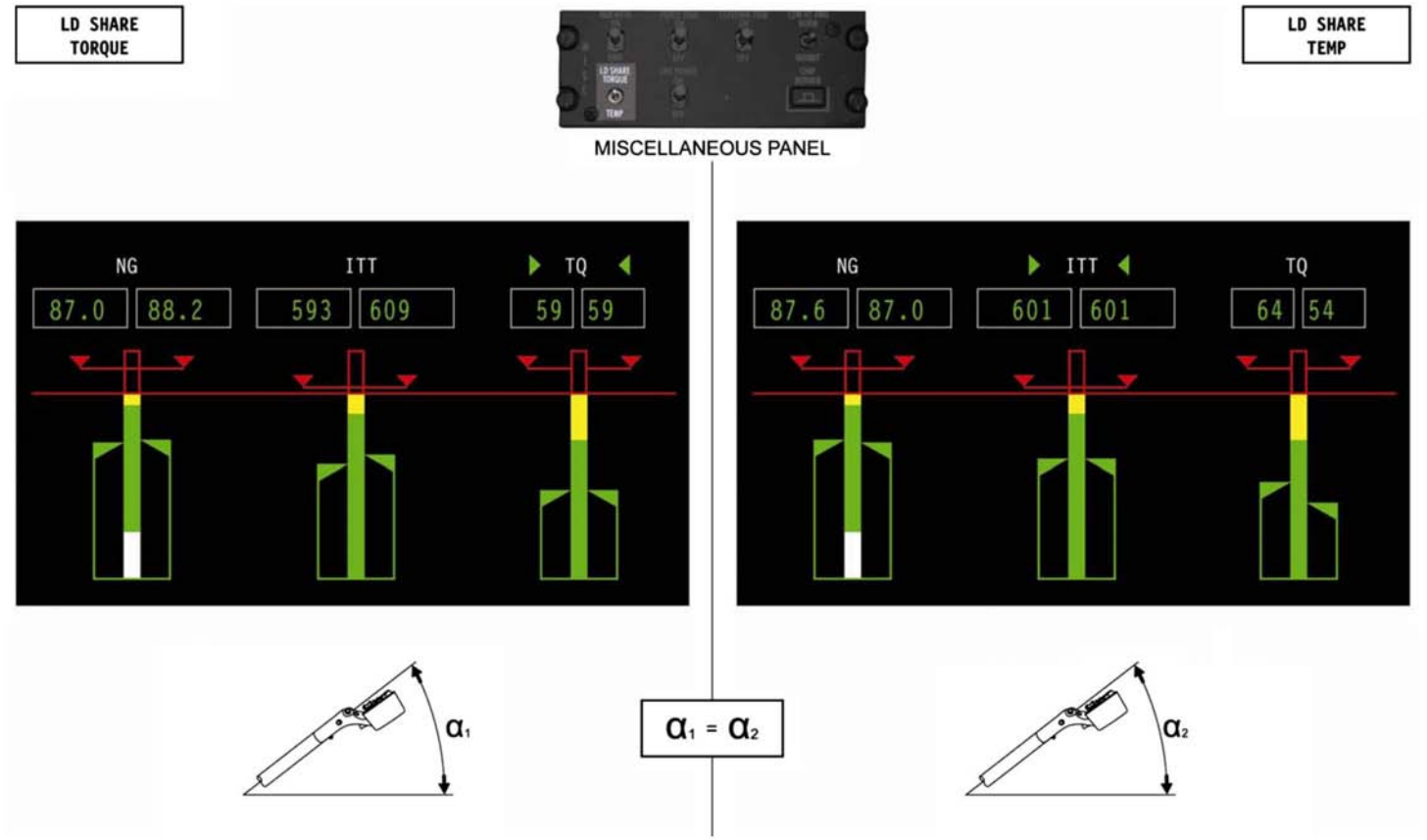


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ECL – OPERATION

ENGINE LOAD SHARE

When both engines are operating at FLIGHT in AUTO mode, the EECs adjust the engines so as to match their torque or ITT values.
Selection of the parameter to be matched is made via the LD SHARE switch on the MISC panel in the central console.
The selected parameter for engine load share is indicated by two green triangles pointing at the TQ or ITT legend of the corresponding indicator in the Pwr Plant page.
The principle of the ITT/TQ matching is to increase the power delivered by the engine that has the lower ITT.



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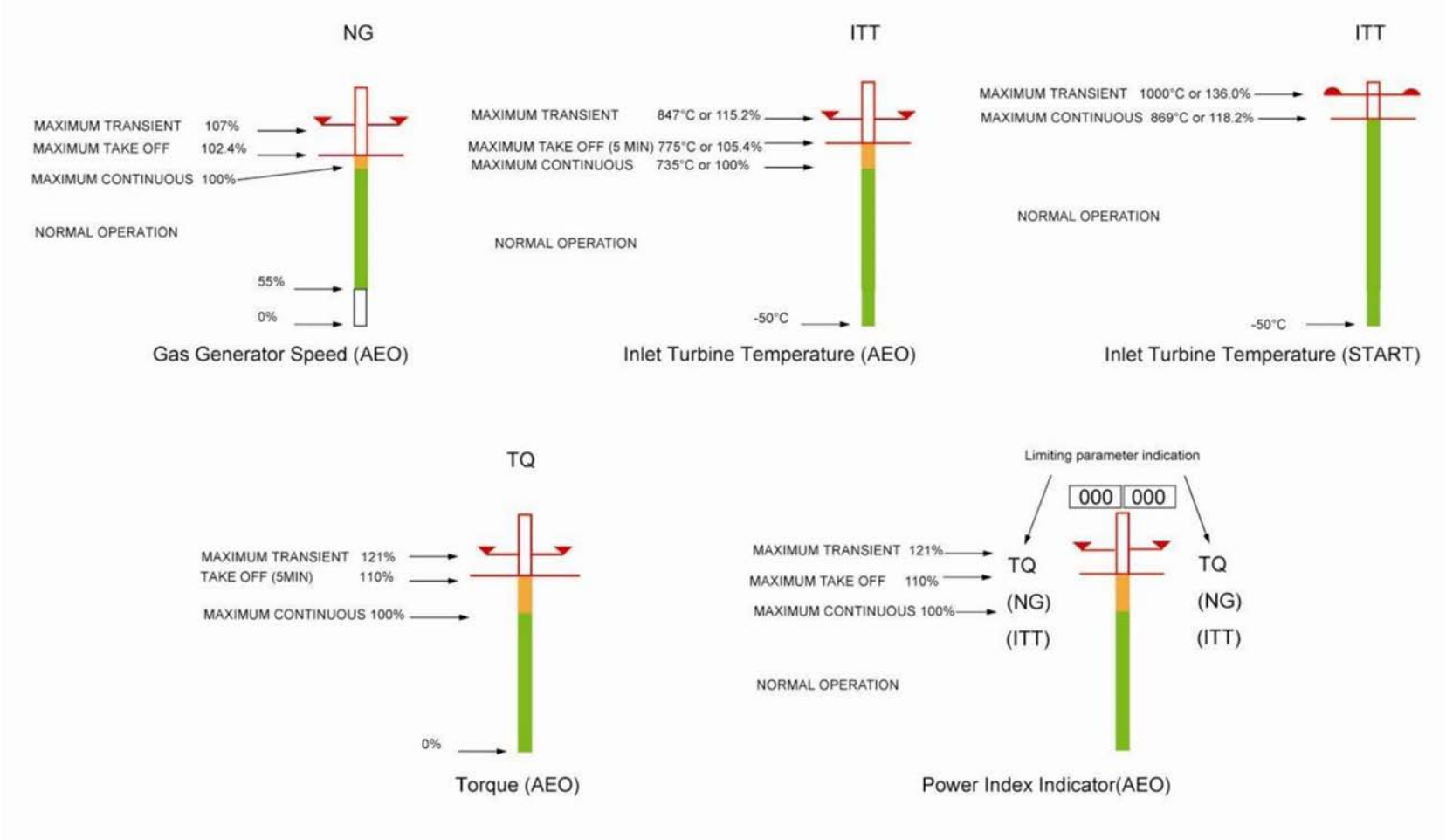
ENGINE LOAD SHARE

ENGINE LIMITATION

Engine intermediate limitations (maximum continuous AEO and OEI, take-off AEO) are intended to protect engine life only; no automatic control through the EEC is provided to respect limits on Ng, ITT and Tq. It is then pilot's responsibility not to exceed this limits (e.g. AEO maximum continuous and take-off power limit, AEO time at take-off power, OEI maximum continuous power limit and time at 2.5 minute range power).
In OEI conditions, if the engine exceeds the "2.5 minute time rating", the 1(2) ENG LIM EXPIRE caution message is displayed in the CAS.
The engine primary indicators and the triple tachometer automatically rescale according to the power plant status (AEO, OEI, Power off) in order to help the pilot respecting the engine and NR limits.

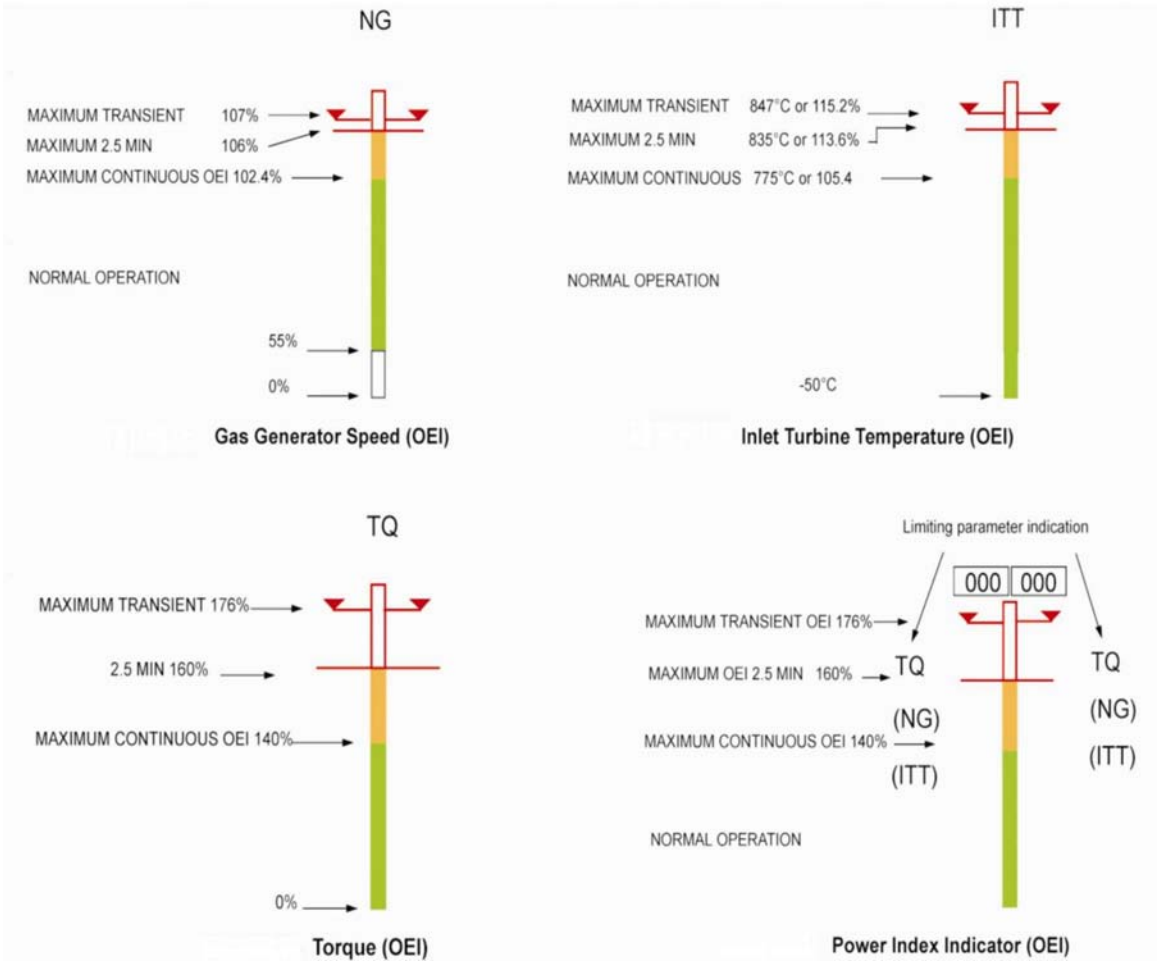
ITT LIMITER

Engine is protected for overtemperature by the ITT Limiter function of the EEC throughout its entire range of operation.
During engine starting, the maximum ITT limit value is displayed on the ITT indicator by a red round mark.
In case the ITT limit value for starting is exceeded, the HOT START caution message is displayed in the CAS and the exceedance is logged.
If the ITT Limiter function fails, the 1(2) ITT LIMITER caution message is displayed in the CAS.



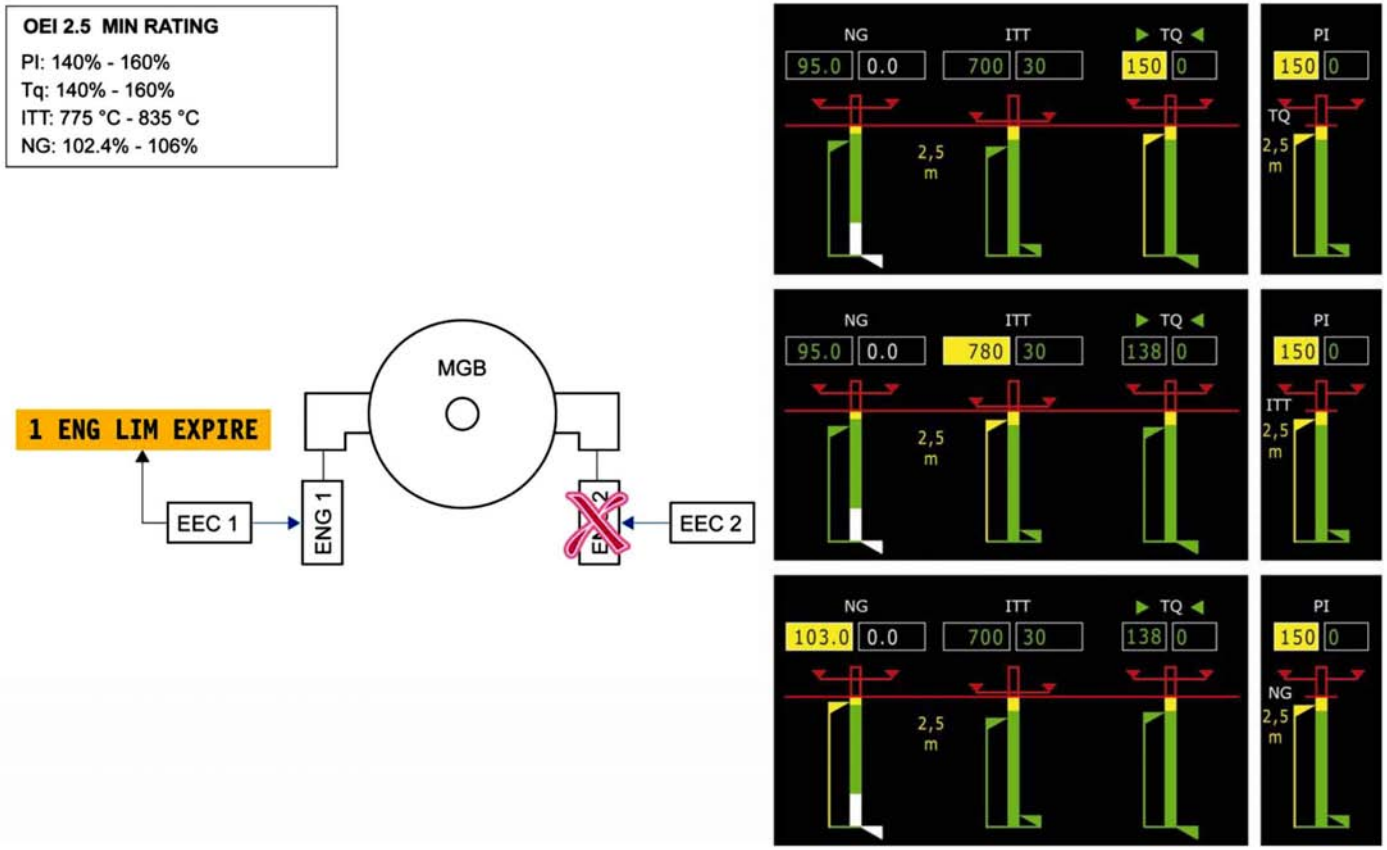
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AEO LIMITATIONS



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OEI LIMITATIONS

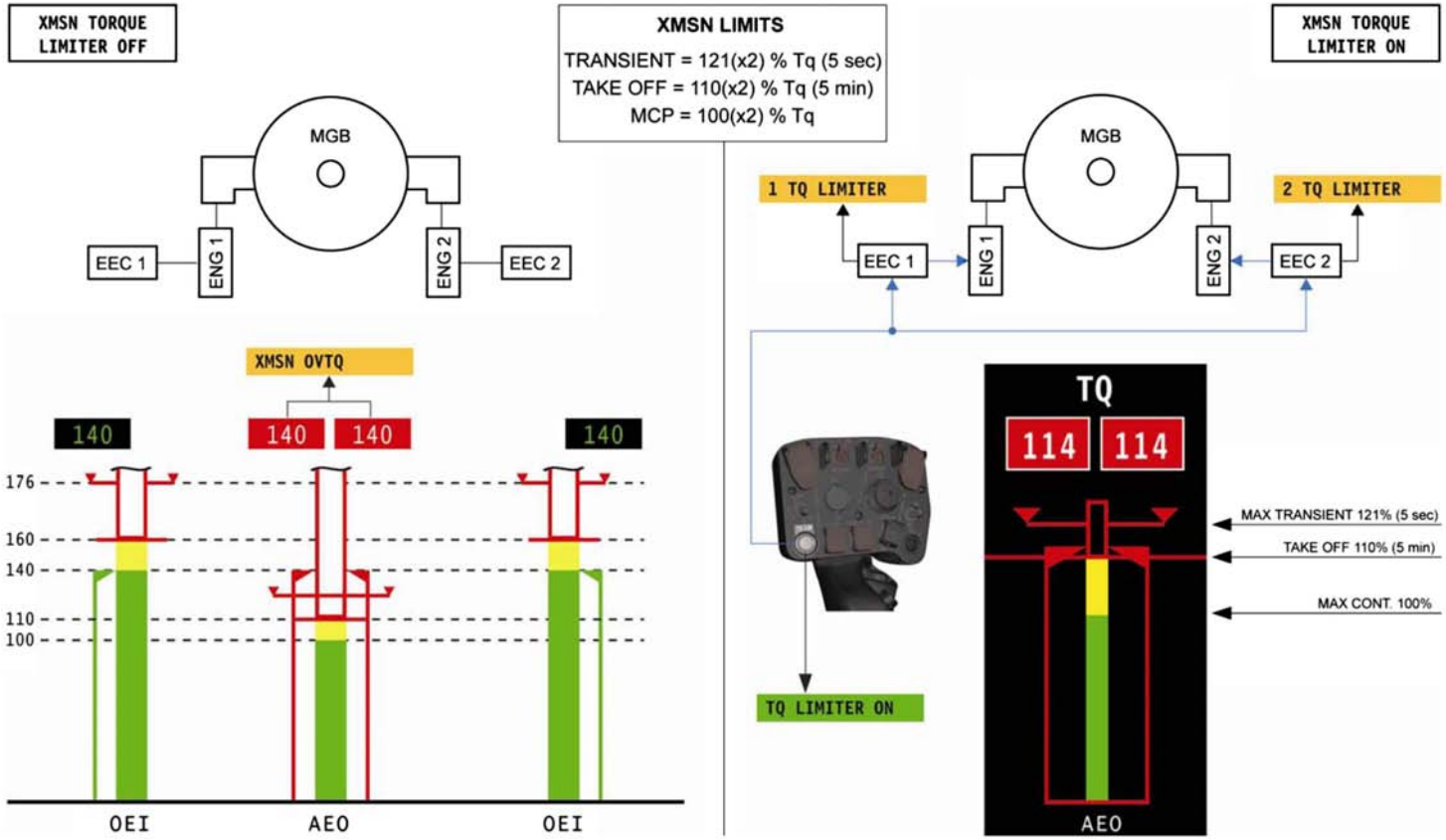


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OEI ENGINE LIMITS

TORQUE LIMITER

Engine power can be limited in order to protect the main gearbox when in AEO by engaging the Torque Limiter function.
This is achieved by pressing the TQ LIM pushbutton on the collective grip; when pressed the EECs control the total engine torque in order to not exceed 228% of total torque. Torque Limiter has no effect in single engine operation.
When Torque Limiter is engaged, the TQ LIMITER ON advisory message is displayed in the CAS. If the Torque Limiter function is failed, the 1(2) TQ LIMITER caution message is displayed in the CAS.
Torque is the only parameter involved in the limitation: Ng and ITT could reach the single engine limits even in AEO with TQ LIM engaged, if allowed by the environmental conditions.
Torque limitation is not available in MANUAL mode.
At power-up the Torque Limiter is OFF by default: it is therefore pilot's choice to manually engage it.



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TORQUE LIMITER

OEI TRAINING MODE

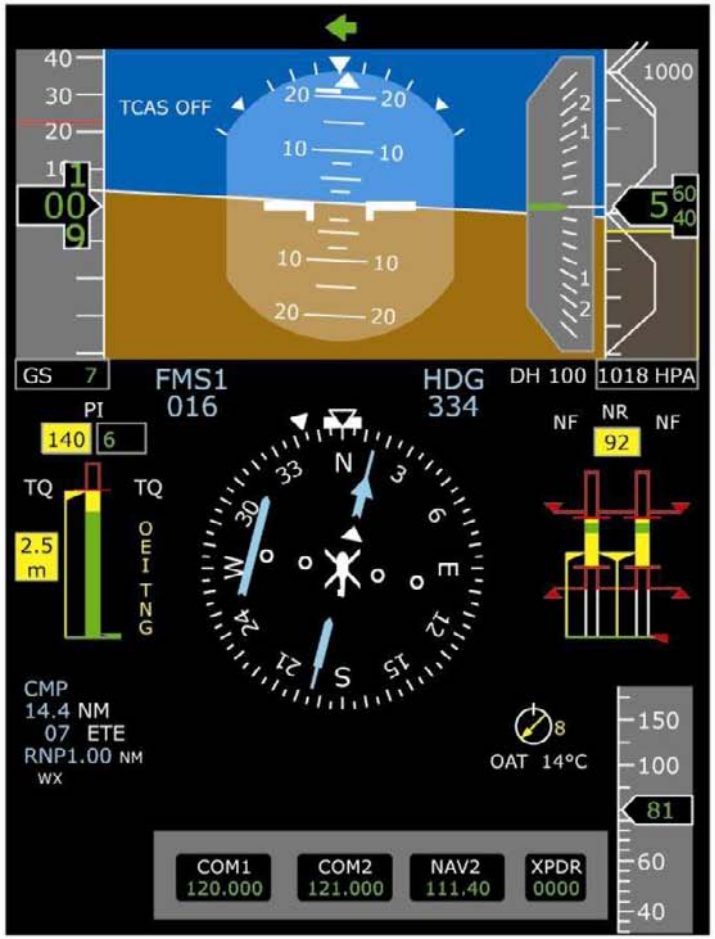
The OEI Training mode is an EEC function available when in AEO conditions in order to simulate an engine flame-out for training purposes.
The mode is selected by moving the OEI TNG switch on the Engine Control Panel to either #1 or #2 position, which selects the engine to be simulated in OEI conditions (the other engine is simulated OFF).
The OEI Training mode logic uses twin engine power to simulate an initial single engine transient to “maximum torque” and rotor droop. Then both engines are used but they are limited to a maximum total PI of 140%.
In order to simulate the transient following an engine failure the engine selected to OEI TNG will accelerate to a maximum of 110% to then reduce to approximately 70%, to while the inoperative engine will decelerate to a minimum of approximately 25% TQ then accelerate to around 70% TQ.
When the Training Mode is activated, the PI and NF displays on the PFD are artificially configured to show OEI condition for the engine not selected to OEI TNG.
On the MFD, the actual engine parameters are displayed while on the NR/NF indicator the coloured ranges are modified, from AEO to OEI, to allow NR/NF droop to 90% as required by the cat A procedure. When in Training Mode and within the OEI TNG 2.5 min range, the 2.5 m legend is dislayed along side the PI of the active engine:

- In amber while within the 2.5 min
- flashing inverse red when within 10 sec from expiry
- steady red inverse when the limit has been exceeded.

In OEI TNG mode the PFD presents the simulated OEI condition while the MFD, for safety reasons, presents the real AEO conditions.
The OEI training mode is disabled if:

- either engine is in MANUAL mode
- a critical or non-critical fault exists on either ECC
- either engine flames out
- the torque limiter switch is OFF
- the ENG switch is not in FLIGHT position on either engine
- the Nr speed drops below 87%

In case the above conditions apply when the OEI training mode is selected, the switch is reset to the centre (OFF) position.





ECP

PI TRN LIMITS	
MCP	130%
2.5 Min. Range	131%-140%
5 Sec. (Transient)	150%

MFD Display in OEI TNG



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OEI TRAINING MODE

CAS WARNING MESSAGES			
CAS CAPTION	**FAILURE DESCRIPTION**	**PROCEDURE NAME**	**AW139-RFM-4D**
1(2) ENG OUT	Associated engine NG less than 34.3% or rate of change outside predetermined limits NOTE A • if the ECC 1 (2) is faulty, the MAU detect the engine failure through the NG1 backup signal whose threshold point is active at NG <50% and inactive at NG >55% • if the NG1 (2) coming from the EEC 1 (2) is not valid, the relevant backup analog values is considered • the AWG provides for two pairs of tones + voice warning ENGINE 1 (2) OUT – ENGINE 1 (2) OUT repeated once	ENGINE OUT	Section 3 EMERGENCY AND MALFUNCTION PROCEDURES ENGINE

CAS WARNING MESSAGES			
CAS CAPTION	**FAILURE DESCRIPTION**	**PROCEDURE NAME**	**AW139-RFM-4D**
1(2) ENG IDLE	Associated engine in IDLE and collective being raised (Triggered on ground only) NOTE B • the AWG provides for the voice warning 1(2) ENG IDLE; message is continuosly repeated	ENGINE IDLE	
1(2) ENG OIL PRESS	Associated engine oil pressure below limit (less than 4.2 bar; resets above 4.7 bar) NOTE C • the AWG provides for the voice warning WARNING – WARNING repeated once	ENGINE OIL PRESSURE LOW	

CAS WARNING MESSAGES			
CAS CAPTION	**FAILURE DESCRIPTION**	**PROCEDURE NAME**	**AW139-RFM-4D**
1(2) EEC FAIL	EEC 1(2)is not operating due to a critical fault NOTE D • the EEC 1 (2) will revert to MANUAL mode automatically with the amber legend MAN in reverse video displayed on the TQ/PI scales. The critical fault is latching and the EEC 1 (2) will not resume engine control in AUTO mode even through signal becomes valid again, unless the AUTO/ MANUAL switch is cycled from AUTO to MANUAL back to AUTO or upon the EEC 1 (2) power reset • the AWG provides for the voice warning WARNING – WARNING repeated once	ENGINE EEC FAIL	

COMPRESSOR STALL

A compressor stall is typically recognized by an abnormal increase of ITT (or abnormal increase in PI indication with ITT as limiting parameter) and may be accompanied by an audible bang or pop and fluctuating NG and TQ (monitored on MFD PWR PLANT page). The compressor stall may be transient or steady.

UNUSUAL ENGINE NOISE

Compressor damage as a result of FOD may increase the engine noise level and is detectable by a high-pitched whining sound. The noise level of the high pitched whine should vary with NG (monitored on MFD PWR PLANT page) and should be significantly higher than the usual engine noise.

ENGINE OIL PRESSURE HIGH

If engine oil pressure operates continuously above oil pressure limit, see RFM – Section 3 for the malfunction procedure.

ENGINE MANUAL START

It is recommended that normal engine starts be made using the AUTO mode. Engine manual startingo, on the ground, should only be carried out if it is essential to move the aircraft, for example from a helideck to allow access to other aircraft. It is necessary, however, to have the other engine functioning in AUTO mode to control rotor speed.
For starting procedure in MANUAL mode refer to RFM – Section 3.
During starting in MANUAL mode the EEC is not in control: fuel flow is adjusted using the ECL and the starting sequence is ended by the GCU when the starter-generator speed sensor senses 50.5% NG.

CAS CAUTION MESSAGES			
CAS CAPTION	**FAILURE DESCRIPTION**	**PROCEDURE NAME**	**AW139-RFM-4D**
1(2) OVSPD	NF approx 110% and/or PI 0%, drive shaft failure on affected engine NF approx 110% and both Pi fluctuating, engine “run-up”	ENGINE POWER TURBINE OVER-SPEED	Section 3 EMERGENCY AND MALFUNC-TION PROCEDURES ENGINE
1(2) ENG LIM EXPIRE	associated engine has exceeded OEI 2.5 minute time rating	ENGINE LIMIT EXCEEDANCE	

CAS CAUTION MESSAGES			
CAS CAPTION	**FAILURE DESCRIPTION**	**PROCEDURE NAME**	**AW139-RFM-4D**
1(2) ENG OIL TEMP	associated engine oil temperature high (greater than 140°C; resets below 137°C)	ENGINE OIL TEMPERATURE	
1(2) ENG CHIP	chip detected in the associated engine oil lubricating system	ENGINE OIL CHIP DETECTOR	
1(2) ECL FAIL	associated ECL not responding to internal Built In Test or link failure between the trim switch and the electrical motor in the ECL	ENGINE CONTROL LEVER	
1(2) ECL POS	associated engine ECL out of FLIGHT position detent (only active when engine control in AUTO mode)	ENGINE CONTROL LEVER POSITION	
1(2) ENG MODE SEL	associated engine MODE SELECT switch failure	ENGINE MODE SELECT SWITCH	
1(2) OVSPD DET	associated engine NF in overspeed condition	ENGINE POWER TURBINE OVERSPEED	
1(2) EEC DATA	associated ECC link interface between ECC and MAU is lost (non critical faul)	ENGINE ELECTRONIC CONTROL DATA	
1(2) DCU	associated engine control function degraded	DEGRADATION OF ENGINE CONTROL FUNCTIONS	

CAS CAUTION MESSAGES			
CAS CAPTION	**FAILURE DESCRIPTION**	**PROCEDURE NAME**	**AW139-RFM-4D**
1(2) HOT START	associated engine ITT limits exceeded during start	ENGINE HOT START	
1(2) TQ LIMITER	associated engine torque limiter system not functioning	TORQUE LIMITER	
1(2) ITT LIMITER	associated engine ITT limiter not functioning	INTER TURBINE TEMPERATURE LIMITER	
RPM SELECT	RPM select switch malfunction	ROTOR SPEED SELECTOR	
1(2) ITT LIMITER	associated engine ITT limiter not functioning	INTER TURBINE TEMPERATURE LIMITER	
RPM SELECT	RPM select switch malfunction	ROTOR SPEED SELECTOR	
1(2) FUEL FILTER	associated fuel filter blockage, impending by-pass condition	FUEL FILTER BY-PASS	
1(2) FUEL ICING	associated fuel temperature less than 5° C, possible fuel heater malfunction and fuel icing	FUEL ICING	

CAS CAUTION MESSAGES			
CAS CAPTION	**FAILURE DESCRIPTION**	**PROCEDURE NAME**	**AW139-RFM-4D**
1(2) FUEL HEATER	associated fuel temperature greater than 74°C or fuel temperature less than 5°C with associated engine oil temperature greater than 82°C. Possible fuel heater malfunction. (Caution only active with associated engine running → NG 1 (2) above 60%)	FUEL HEATER	
AVIONIC FAULT	loss of communications to a single MAU is detected for EEC 1 and EEC 2 NOTE E • The maintenance message in the CMC is A429/RS422 BUS	AVIONIC	

CAS CAUTION MESSAGES			
CAS CAPTION	**FAILURE DESCRIPTION**	**PROCEDURE NAME**	**AW139-RFM-4D**
1(2) NF MISCOMPARE **OR** **1(2) NG MISCOMPARE** **OR** **1(2) ITT MISCOMPARE** **OR** **1(2) TQ MISCOMPARE** **OR** **NR MISCOMPARE**	associated parameter EEC and analogue backup data comparison discrepancy. The MISCOMPARE caution is generated when comparison with backup parameter exceeds the following values: NF 3% NG 3% ITT 50°C TQ 5% NR 3% NOTE F - the analogue sensors are selected from MFD PWR PLANT page, menu selection using Cursor Control Device (CCD)	ENGINE AND ROTOR PARAMETERS MISCOMPARE	

CAS CAUTION MESSAGES			
CAS CAPTION	**FAILURE DESCRIPTION**	**PROCEDURE NAME**	**AW139-RFM-4D**
ENG ANALOG FAILURE	engine analogue monitoring systems failed NOTE G • the analogue sensors are selected from MFD PWR PLANT page, menu selection using Cursor Control Device (CCD)	ENGINE ANALOGUE SENSOR FAILURE	

CAS CAUTION MESSAGES			
CAS CAPTION	**FAILURE DESCRIPTION**	**PROCEDURE NAME**	**AW139-RFM-4D**
1(2) MAU OVHT	Associated MAU overheat If MAU1 fails The secondary engine parameter 1 FUEL PUMP is not valid and the following CAS cautions are not available: • 1 ENG OIL TEMP • 1 ECL FAIL • 1 ECL POS • 1 FUEL HEATER • 1 FUEL ICING If MAU2 fails The secondary engine parameter 2 FUEL PUMP is not valid and the following CAS cautions are not available: • 2 FUEL HEATER • 2 FUEL ICING	MODULAR AVIONICS UNIT OVER-HEAT / FAIL	

CAS ADVISORY MESSAGES			
CAS CAPTION	**FAILURE DESCRIPTION**		**AW139-RFM-4D**
TQ LIMITER ON	engine torque limiter activated		NORMAL PROCEDURES ADVISORY CAPTIONS DEFINITIONS

ENGINE LIMITATIONS

Refer to AW139-RFM-4D Section 1 for

- engine limitations
- NG limitations
- ITT limitations
- TQ limitations
- NR/NF limitations
- POWER INDEX limitations

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IGNITION

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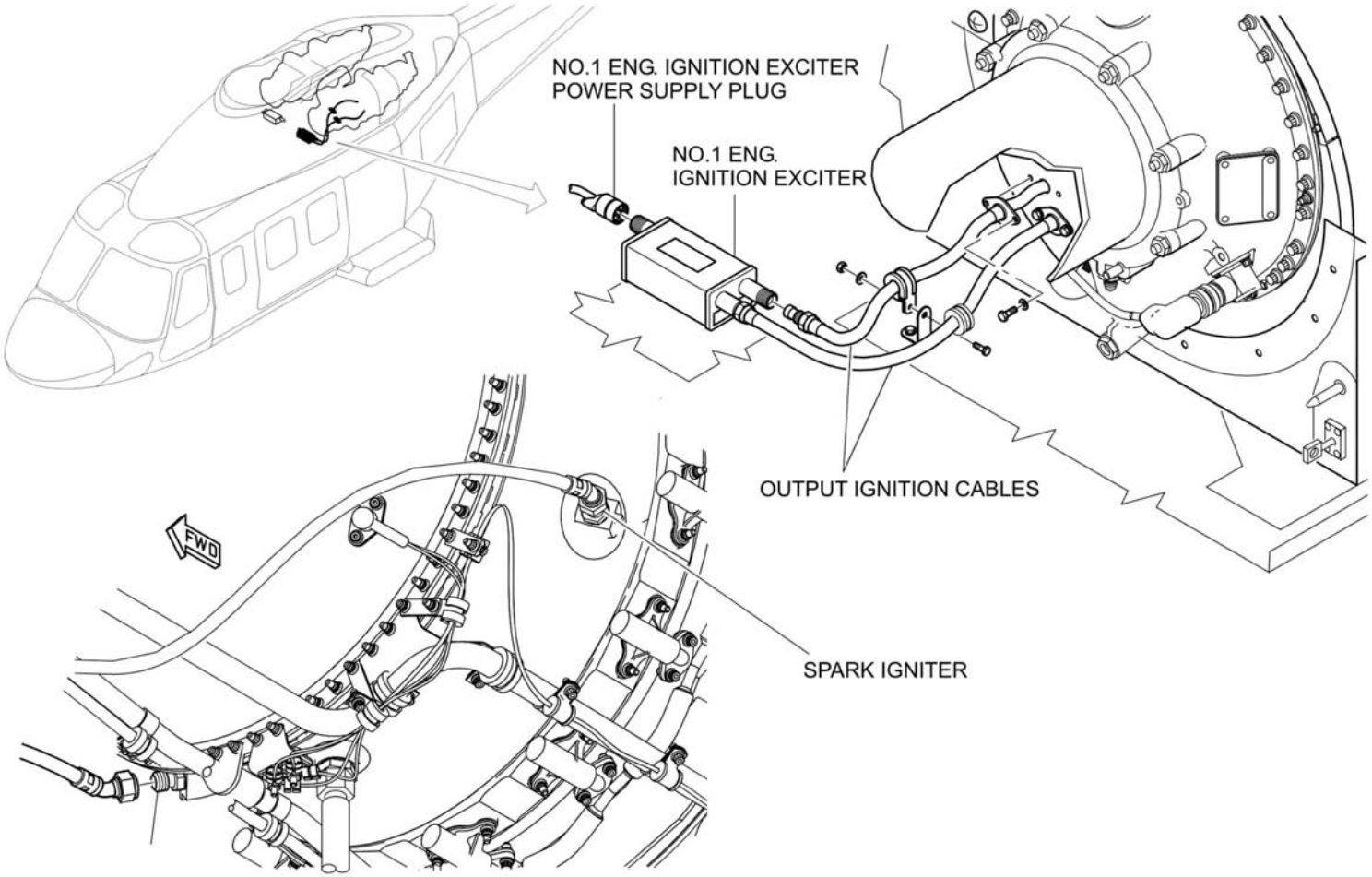
SECTION 10 - ELECTRICAL POWER SUPPLY

DESCRIPTION

The ignition provides the initial spark to ignite the fuel/air mixture.
The system consists of a dual output ignition exciter, two high-tension cable assemblies and two spark igniters.
The unit generates sparks in the 10 to 30 VDC range and is located on the upper deck, just after the main transmission.

OPERATION

The ignition exciter is a sealed unit containing electronic circuitry encased in an epoxy resin.
It transforms the DC input into a pulsed high voltage output.
It is a dual channel capacitor discharge type that has the capacity of storing 1.25 Joule of energy at an output voltage of 8 kW.
The ignition can be performed in two ways: automatically or manually (according to the starting type) selected by the AUTO/MANUAL switch on the pilot collective stick panel.
The automatic ignition is started by setting the ENG 1 or ENG 2 MODE selector switch of the ENGINE CONTROL PANEL to IDLE or FLT.
During this phase, the MFD will display the green legend IGN aside the ITT scale on the primary and secondary VMS information page.
It is disengaged when the EEC senses that the engine speed value is at 50 ± 1% NG.
In case the automatic mode cannot be performed, the MANUAL mode is selected via the AUTO/MANUAL switch of the pilot collective stick panel and the starting sequence is initiated by pressing the START button on the ENGINE CONTROL PANEL.
As the engine speed reaches the 49 ± 1% NG, the ignition sequence is stopped automatically by the relative GCU.
Also during the manual start the legend IGN will be displayed.



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N.1 IGNITION EXCITER COMPONENT LOCATION

CONTROLS AND INDICATORS

The ENGINE CONTROL PANEL housed on the overhead panel and the pilot collective levers are instead involved in the ignition-starting system when the automatic mode is performed.
The ENGINE CONTROL LEVER housed on the overhead panel and the pilot collective levers are instead involved in the ignition-starting system when the manual mode is performed.
The MFD provides the only green legend IGN during the ignition.
The ENGINE CONTROL PANEL provides the following functions:

1 ENG 1 (2) MODE selector switch:

- OFF: ..... when the N. 1 (2) engine is not running or it shuts down the engine or during the N. 1 (2) engine dry motoring procedure when Auto or Manual is selected
- ON: ..... allows the N. 1 (2) engine starting, the minimum on ground (65% Npt) and the N. 1 (2) engine stabilization for a period of 60 sec. during the shut down phase or during the N. 1 (2) engine wet motoring procedure both Auto or Manual mode

The pilot collective stick provides the following functions:

2 ENG 1 (2) GOV AUTO/MANUAL selector switch:

- AUTO: ..... allows the N. 1 (2) engine automatic starting and provides the N. 1 (2) engine automatic control
- MANUAL: ..... allows the N. 1 (2) engine manual starting / wet motoring / dry motoring and provides the N. 1 (2) engine manual control

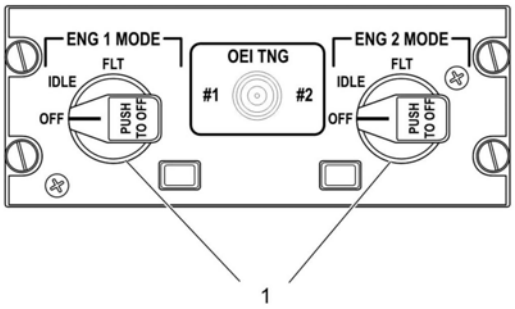
NOTE A

When in MANUAL mode, the amber legend MAN in reverse video is displayed on the left side of TQ scale (MFD Main and Cruise pages) and PI scale (PFDs).

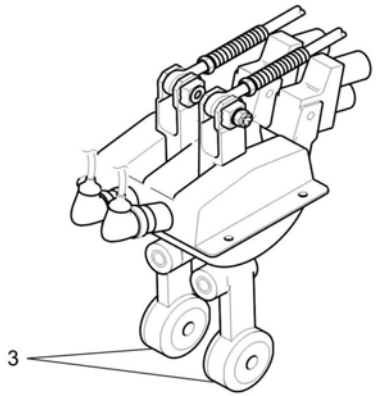
The ENGINE CONTROL LEVER provides the following functions:

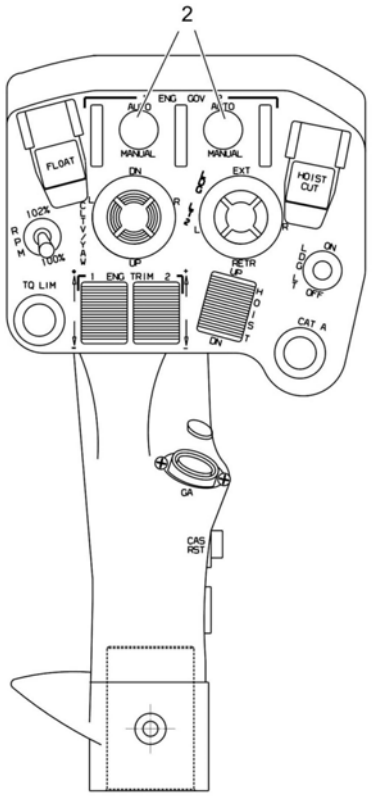
3 ENG 1 (2) START push-button switch: ..... when pressed, the N. 1 (2) engine manual starting / dry motoring / wet motoring is engaged in Manual mode.

ENGINE CONTROL PANEL



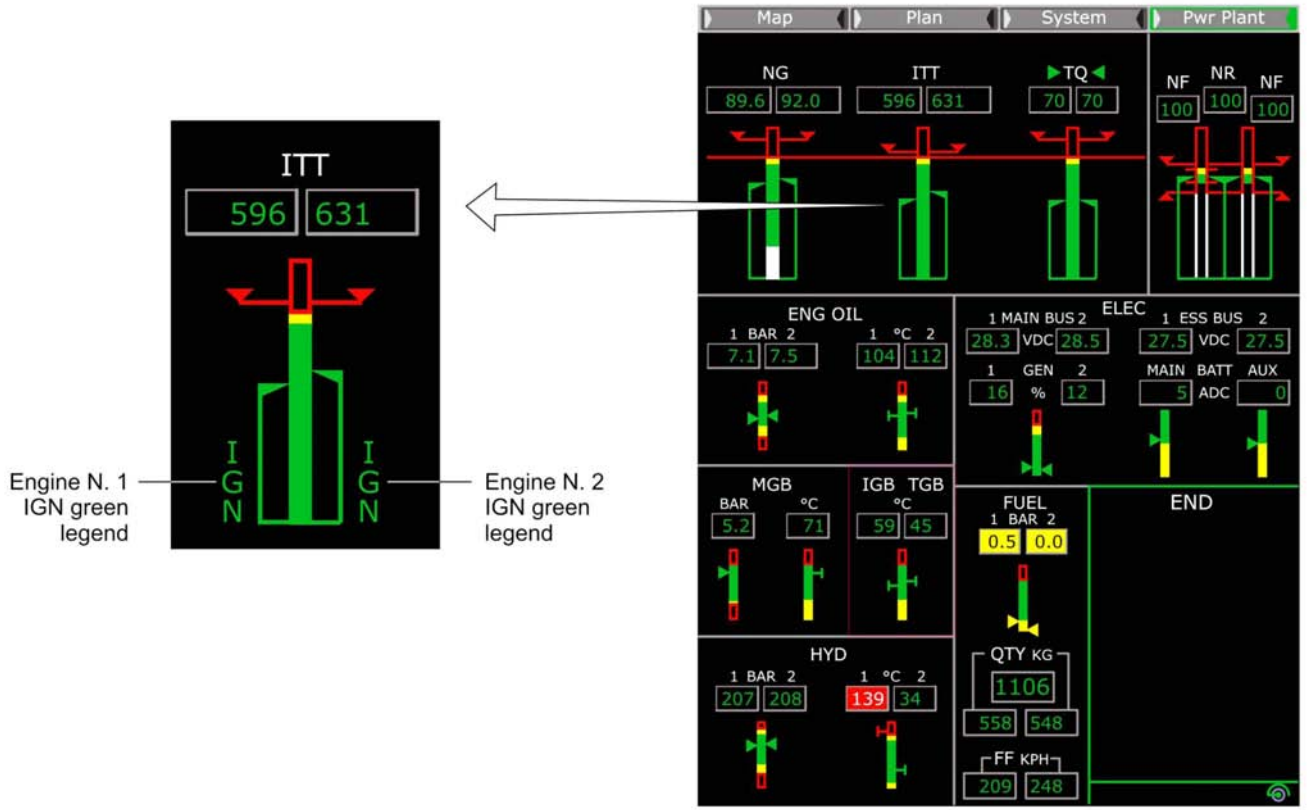
ENGINE CONTROL LEVER





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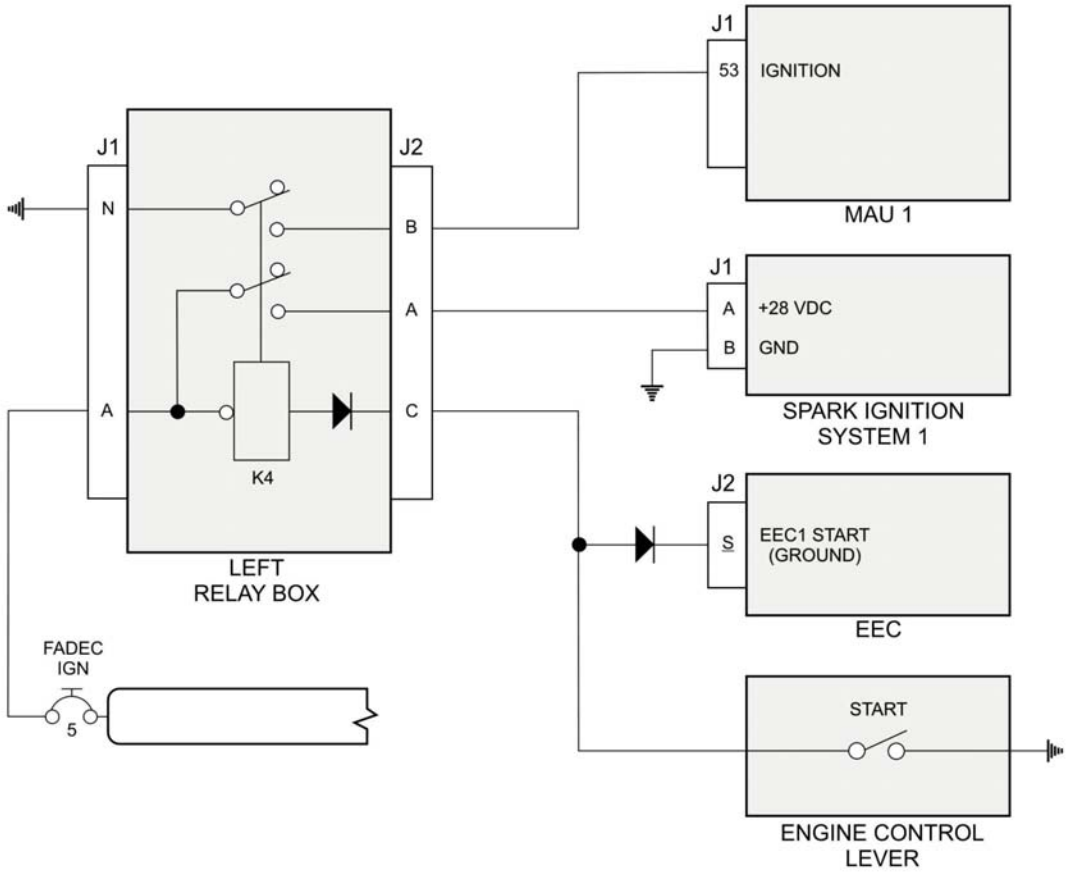
IGNITION SYSTEM CONTROLS



MULTIFUNCTION DISPLAY (MFD)

ICN-39-A-741000-G-A0126-01149-A-01-1

IGNITION SYSTEM INDICATING



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ENGINE IGNITION SYSTEM – SCHEMATIC DIAGRAM

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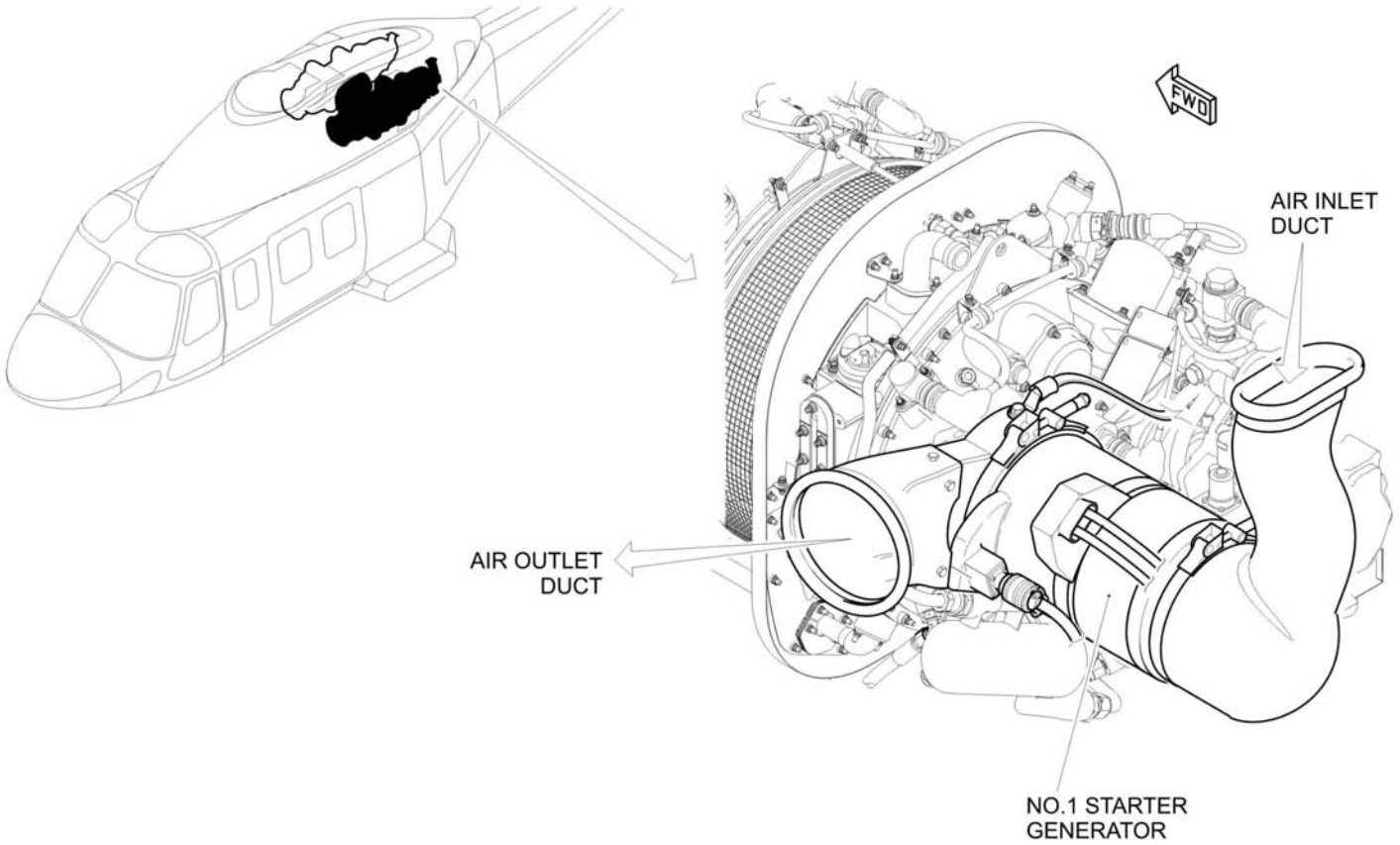
STARTING

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SECTION 10 - CRANKING

DESCRIPTION

The engine starting can be done automatically or manually via an electrical starter/generator motor. It is recommended that normal engine starts be made using the auto mode. Either engine may be started first. The starting is controlled through the EEC (AUTO mode) or the GCU (MANUAL mode).
The normal speed range for the start cycle is from 0 to 5500 rpm and the normal speed for the generator mode is from 6800 rpm to 11078 rpm.
At all times during the start sequence, pilot monitoring of engine limitations is required to prevent engine deterioration due to an abnormal starting condition.
Following an engine shutdown due to flameout or by an automatic or manual shutdown initiated by the pilot, the normal starting procedure should follow the starting requirements.
During the starting phase, the ignition system is automatically controlled and it is turned on at the beginning of the start sequence.
The GCU incorporates an automatic starter cut-out feature, such that the GCU will open the starter-generator line contactor at a predetermined speed of approx. 5500 rpm to terminate the start cycle.



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N.1 STARTER – GENERATOR LOCATION

OPERATION

To start the engines in AUTO mode, the pilot uses the Engine Control Panel.
To start the engines in MANUAL mode, the pilot uses both the Engine Control Lever and the Engine Control Panel.
The engine starting AUTO or MANUAL mode is selected via the 1 (2) ENG GOV switch on the pilot collective grip.
Besides the engine start, cranking is required to perform the dry motoring run for engine washing/ventilation and the wet motoring run to check for fuel leak or engine preservation/depreservation during maintenance operation.
To start the engine in AUTO mode proceed as follows:

- set the 1 (2) ENG GOV switch on the pilot collective grip to AUTO
- set the Engine Control Lever 1 (2) to FLIGHT position to allow the engine fuel flowing
- set the FUEL PUMP 1 (2) switch of the FUEL CONTROL PANEL to ON
- set the FUEL SHUT-OFF VALVE 1 (2) of the FUEL CONTROL PANEL to ON
- set the ENG 1 (2) MODE selector switch of the ECP to IDLE/FLIGHT position to open the shutdown solenoid and so to allow the engine fuel flowing and to provide the electrical power to start and spark igniters
- the green legends START/IGN are displayed on left/ right side of NG and ITT scale respectively
- the EEC de-energizes the start-ignition relays at 50 ±1% NG.

NOTE A

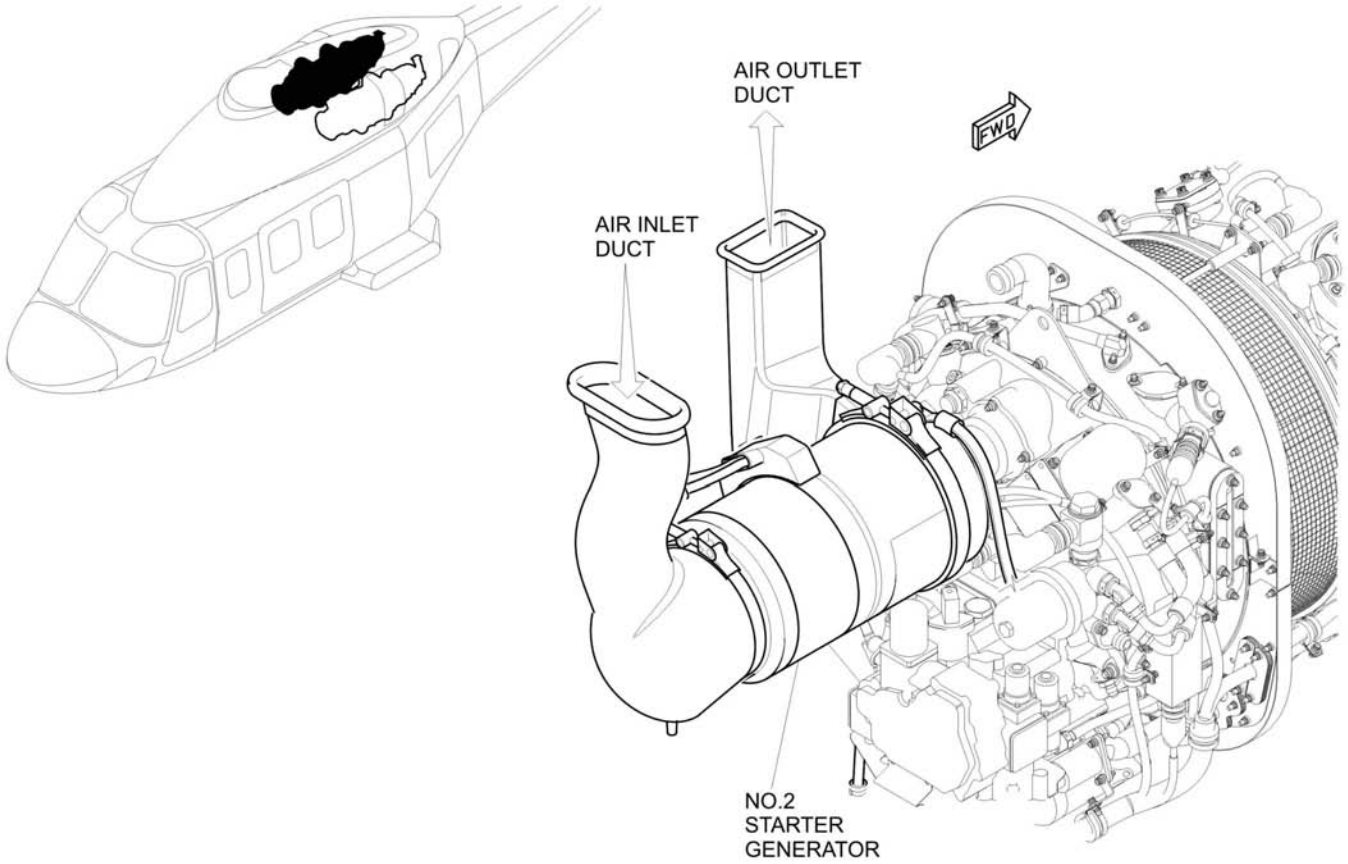
The quick engine start procedure (AUTO mode) may be carried out to speed up the take off. Set the ENG 1 (2) MODE selector switch to FLIGHT directly, then the second engine to FLT when NG No. 1 (2) is above 25%.

NOTE B

The engine start duty cycle is the following:

- 45 seconds on, 1 minute off
- 45 seconds on, 1 minute off
- 45 seconds on, 30 minutes off

- To start the engine in MANUAL mode proceed as follows:
- set the 1 (2) ENG GOV slector switch on the pilot collective grip to MANUAL and note the amber legend MAN in view on TQ/PI scales
- set the ENGINE CONTROL LEVER 1 (2) to OFF position
- set the FUEL PUMP 1 (2) switch of the FUEL CONTROL PANEL to ON
- set the FUEL SHUT-OFF VALVE 1 (2) of the FUEL CONTROL PANEL to ON
- set the ENG 1 (2) MODE selector switch of the ENGINE CONTROL PANEL to IDLE position to open the fuel shutdown solenoid to allow the subsequent engine fuel flowing
- press the START push-button switch on the ENGINE CONTROL LEVER 1 (2) and advance the ECL as necessary to allow the engine fuel to flow
- the green legends START/IGN are displayed on left/ right side of NG/ITT scales respectively.



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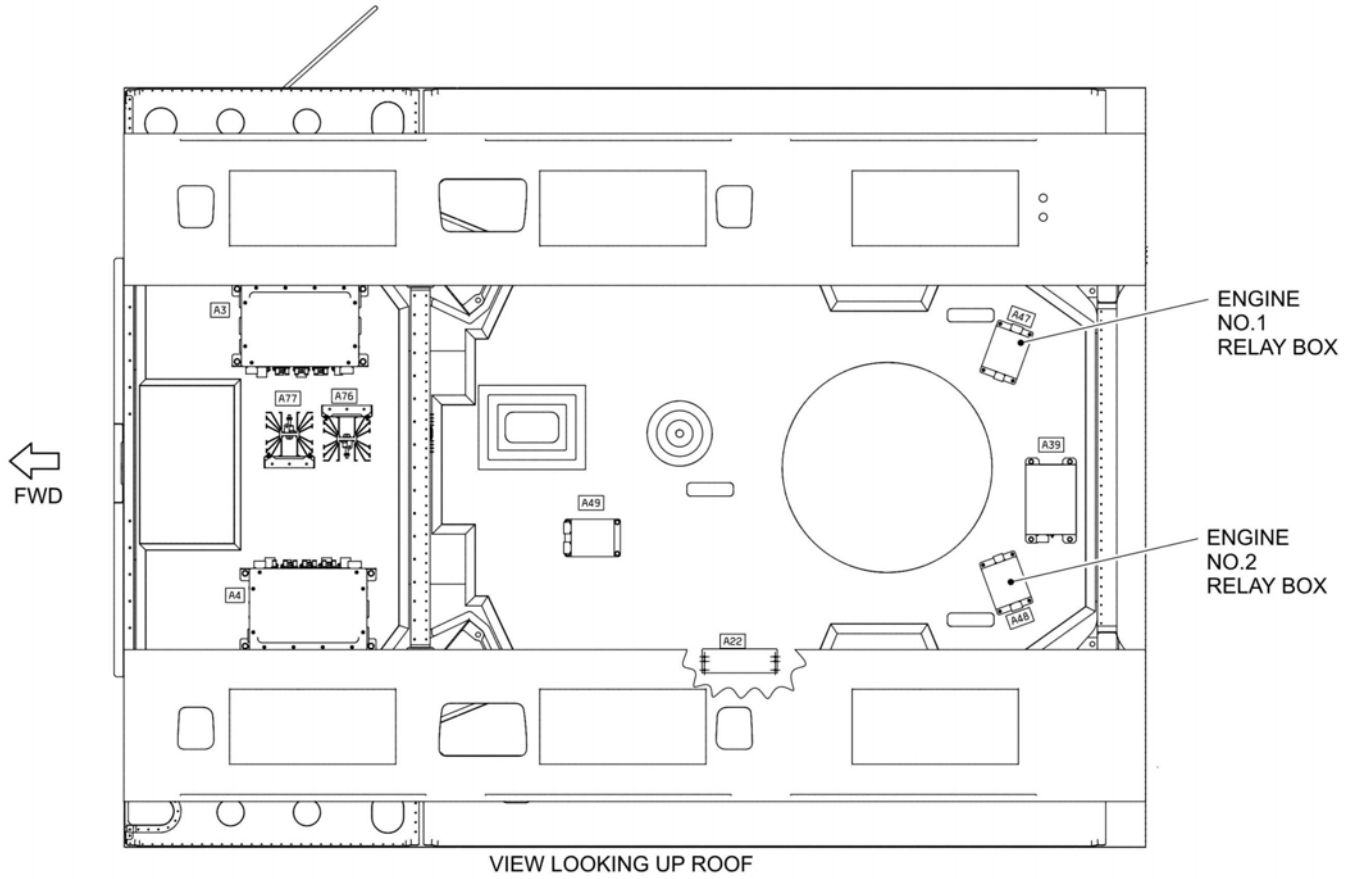
N.2 STARTER – GENERATOR LOCATION

once the engine reaches the 49 ±1% NG, the GCU disengages automatically the starter-ignition relays.
To start the engine for the dry motoring run, following an aborted start shutdown, allow a 30 sec fuel drain period before re-starting.
The procedure is used to clear internally trapped fuel and vapor or if there is evidence of a fire within the engine. Observe the starter-generator duty cycle limitations during the re-start attempts.
Proceed as follows:

- set the 1 (2) ENG GOV selector switch either to AUTO or MANUAL
- set the ECL 1 (2) to OFF either to AUTO or MANUAL
- set the FUEL PUMP/VALVE 1/2 switches to OFF
- pop the IGN 1 (2) circuit breaker out
- set the ENG 1 (2) MODE selector switch to IDLE if AUTO otherwise OFF
- press the start push-button switch if MANUAL
- press the start push-button switch a second time to stop the dry motoring run sequence if MANUAL
- set the ENG 1 (2) MODE selector switch to OFF to stop the dry motoring run sequence if AUTO.

To start the engine for the wet motoring run proceed as follows:

- set the 1 (2) ENG GOV selector either to AUTO or MANUAL
- set the ECL 1 (2) to FLIGHT either to AUTO or MANUAL
- set the FUEL PUMP/VALVE 1 (2) switches to ON
- pop the IGN 1 (2) circuit breaker out
- set the ENG 1 (2) MODE selector switch to IDLE
- press the start push-button switch if MANUAL. The fuel flow is delivered to the engine as fuel pump speed increases
- press the start push-button switch a second time to stop the wet motoring run sequence if MANUAL
- set the ENG 1 (2) MODE selector switch to OFF to stop the wet motoring run sequence if AUTO.



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CRANKING COMPONENTS LOCATION

CONTROL AND INDICATORS

The ECP housed on the console and the pilot collective levers are involved in the starting when the auto mode is performed.
The ECL housed on the overhead panel, the ECP and the pilot collective levers are instead involved in the starting when the manual mode is performed.
The MFD provides the green legend START during the starting.
The malfunctions are displayed in the CAS message window #1.
The ECP provides the following functions:

1 ENG 1(2) MODE selector switch:

- OFF: ..... when the N. 1(2) engine not running or it shuts down the engine or during the N.1(2) engine dry motoring procedure when Manual selected.
- IDLE: ..... allows the N. 1(2) engine starting, the minimum on ground (65% NF) and the N. 1(2) engine stabilization for a period of 60 sec during the shut down phase or during the N. 1(2) engine wet motoring procedure both Auto and Manual mode.
- FLT: ..... allows the N. 1(2) engine quick starting at the flight nominal speed (100% NF).

NOTE C

The MFD and PFDs display the green legend IDLE on the left side of NF scale of the triple tachometers. The amber legend FAIL is displayed on the triplo tachometer in case of TQ/NF sensors failure. It replaces the green legend IDLE.

NOTE D

During the starting phase, the NG digital readout with the associated pointer are white until the 49% NG is reached.
The two green legend START/IGN are removed at the same percentage.

The pilot collective stick provides the following functions:

2 ENG 1(2) GOV AUTO/MANUAL selector switch:

- AUTO: ..... allows the N. 1(2) engine automatic starting and provides the N. 1(2) engine automatic control.
- MANUAL: ..... allows the N. 1(2) engine manual starting / wet motoring / dry motoring and provides the N. 1(2) engine manual control.

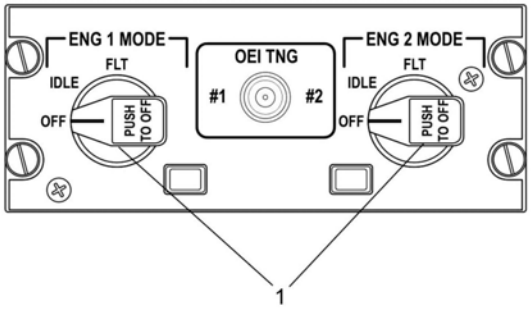
NOTE E

When in MANUAL, the amber legend MAN in reverse video is displayed on the left side of TQ scale (MFD Main and Cruise pages) and PI scale (PFDs).

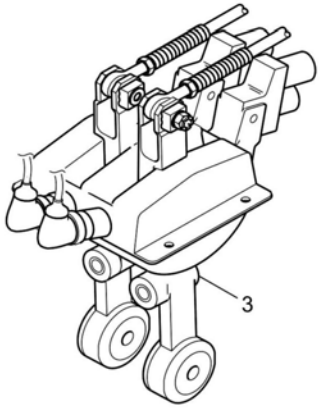
The ECL provides the following functions:

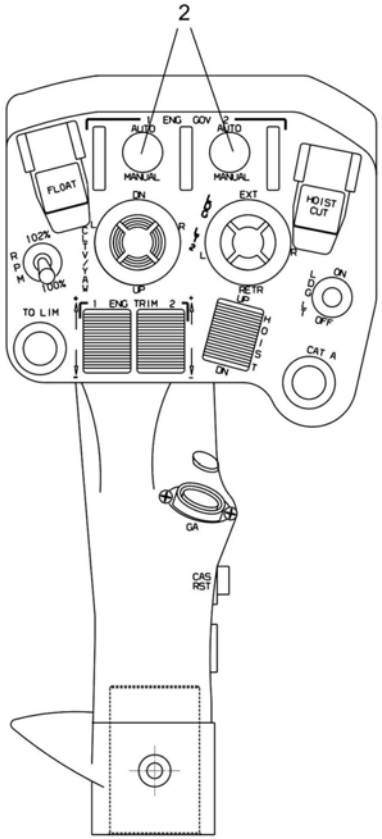
3 ENG 1(2) START push-button switch: ..... when pressed the N. 1(2) engine manual starting / wet motoring 7 dry motoring is engaged in Manual mode.

ENGINE CONTROL PANEL



ENGINE CONTROL LEVER





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CRANKING CONTROLS